Astro Digital Corvus-BC3 Orbital Debris Assessment Report (ODAR)

CORVUSBC3-ODAR-1.3

This report is presented as compliance with NASA-STD-8719.14, APPENDIX A.
Report Version: 1.3, 9/21/2017

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DAS Software Version Used In Analysis: v2.0.2

Revision Record				
Revision:	Date:	Affected Pages:	Changes:	Author(s):
1.0	9/7/2015	All –Initial	DAS Software Results Orbit Lifetime Analysis	B. Cooper
1.1	11/12/2015		Minor formatting	B. Cooper
1.3	9/21/2017		Updated orbit for launch on PSLV	B. Cooper

Table of Contents

Self-assessment of the ODAR using the format in Appendix A.2 of NASA-STD-8719.14:

A self assessment is provided below in accordance with the assessment format provided in Appendix A.2 of NASA-STD-8719.14.

Requirement #	Launch Vehicle				Spacecraft			Comments
	Compliant	Not Compliant	Incomplete	Standard Non Compliant	Compliant	Not Compliant	Incomplete	
4.3-1.a	☐	☐	☒	☐	☒	☐	☐	No Debris Released in LEO. See note 1.
4.3-1.b	☐	☐	☒	☐	☒	☐	☐	No Debris Released in LEO. See note 1.
4.3-2	☐	☐	☒	☐	☒	☐	☐	No Debris Released in GEO. See note 1.
4.4-1	☐	☐	☒	☐	☒	☐	☐	See note 1.
4.4-2	☐	☐	☒	☐	☒	☐	☐	See note 1.
4.4-3	☐	☐	☒	☐	☒	☐	☐	No planned breakups. See note 1.
4.4-4	☐	☐	☒	☐	☒	☐	☐	No planned breakups. See note 1.
4.5-1	☐	☐	☒	☐	☒	☐	☐	See note 1.
4.5-2					☒	☐	☐	No critical subsystems needed for EOM disposal
4.6-1(a)	☐	☐	☒	☐	☒	☐	☐	See note 1.
4.6-1(b)	☐	☐	☒	☐	☒	☐	☐	See note 1.
4.6-1(c)	☐	☐	☒	☐	☒	☐	☐	See note 1.
4.6-2	☐	☐	☒	☐	☒	☐	☐	See note 1.
4.6-3	☐	☐	☒	☐	☒	☐	☐	See note 1.
4.6-4	☐	☐	☒	☐	☒	☐	☐	See note 1.
4.6-5	☐	☐	☒	☐	☒	☐	☐	See note 1.
4.7-1	☐	☐	☒	☐	☒	☐	☐	See note 1.
4.8-1					☒	☐	☐	No tethers used.

Note 1: The primary payloads for this mission belong to other organizations. This is not a primary mission of Aquila Space. All other portions of the launch composite are not the responsibility of Aquila Space and the Corvus Program is not the lead launch organization.

<u>Assessment Report Format</u>:

ODAR Technical Sections Format Requirements:

Aquila Space, Inc. is a US company. This ODAR follows the format in NASA-STD-8719.14, Appendix A.1 and includes the content indicated as a minimum, in each of sections 2 through 8 below for the Corvus-BC3 satellite. Sections 9 through 14 apply to the launch vehicle ODAR and are not covered here.

<u>Corvus-BC3 Space Mission Program</u>:

<u>ODAR Section 1</u>: Program Management and Mission Overview

Program/project manager: Brian Cooper

Senior Management: Chris Biddy

Foreign government or space agency participation: None.

Summary of NASA's responsibility under the governing agreement(s): N/A

Schedule of upcoming mission milestones:

- Shipment of one (1) spacecraft to Spaceflight Services, Seattle, WA: 30 January 2016
- Launch: 1 April 2016

Mission Overview: Corvus-BC3 is a remote sensing satellite designed to collect multi-spectral imagery data at 22 meters resolution. It will be launched into a sun-synchronous, Low Earth Orbit (LEO) inside a 6U Cubesat deployer device developed by ISIS, Inc. The deployer is to be included on-board a PSLV launch vehicle, planned for launch on 15 December 2017. The spacecraft carries three separate cameras to gather imagery in the Red, Green, and Near-Infrared spectral bands with frequent revisit times. This imagery is processed on-board and then downlinked over a miniaturized high-speed Ka-band transmitter. The satellite bus uses reaction wheels, magnetic torque coils, a star tracker, magnetometers, sun sensors, and gyroscopes to enable precision 3-axis pointing without the use of propellant.

Launch Vehicle and Launch Site: PSLV, Satish Dhawan Space Centre, India

Proposed Launch Date: 15 December 2017

Mission Duration: The anticipated lifetime of the spacecraft (pl.) is ≥ 1 year in LEO.

Launch and deployment profile, including all parking, transfer, and operational orbits with apogee, perigee, and inclination: The Falcon 9 launch vehicle will launch will transport multiple mission payloads to orbit. Corvus-BC3 will be deployed into an approximately sun synchronous elliptical low Earth orbit. Once the final stage has burned out, the primary payloads will be dispensed. After the primary payloads are clear, the secondary payload will separate. Corvus-BC3 will deploy a UHF antenna and two solar panels once deployed from a QuadPack deployer from ISIS. The spacecraft will decay naturally from an operational circular orbit defined as follows:

Apogee: 505 ± 20 km

Perigee: 505 ± 20 km

Inclination: 97.4° ± 1°

Corvus-BC3 has no on-board propulsion and therefore does not actively change its orbit. There is no parking or transfer orbit.

<u>ODAR Section 2</u>: Spacecraft Description:

Physical description of the spacecraft: Corvus-BC3 is based on the 6U Cubesat form factor. Basic physical dimensions are 366 mm x 239 mm x 113 mm with a mass of approximately 11.5 kg. The superstructure is comprised of six rectangular plates forming the sides of the structure with interior stiffening members. There are L rails along each of the 366 mm corner edges. These accommodate the deployment of the satellite from the deployer. Additional stiffness is provided by various major module components mounted within the spacecraft structure. These include the Imaging Payload, the Ka-Band transmitter, the Attitude Control Module, and the Data and Power Module. The design includes a spring-loaded UHF and two solar panels that are deployed after jettison from the deployer by two independent burn wires controlled by software timers via the flight computer. Power is locked away from all spacecraft platform and payload components by means of redundant series separation switches. These switches cannot be activated until the spacecraft separates from the deployer structure. The spacecraft is depicted in Figure 1.

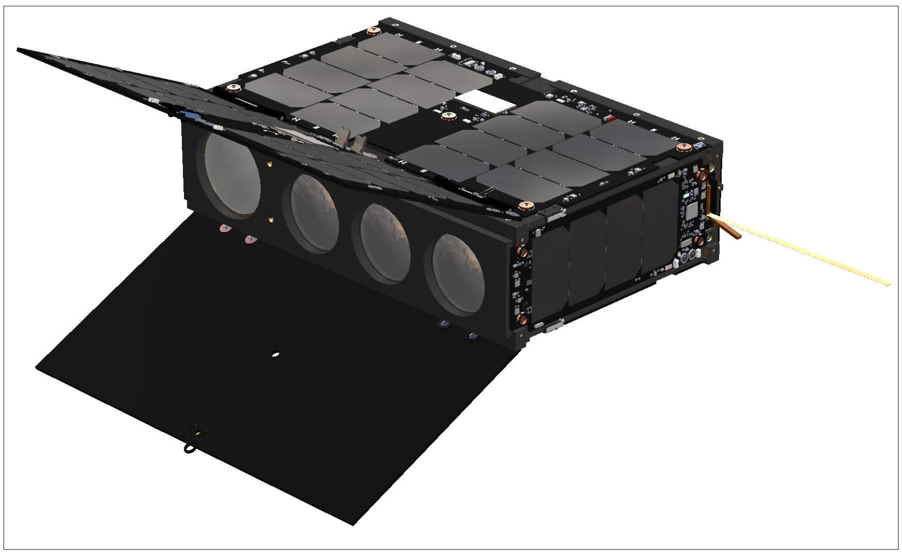

Figure 1: Corvus-BC3 Spacecraft

Total satellite mass at launch, including all propellants and fluids: 11.5 kg.

Dry mass of satellites at launch: 11.5 kg. (No propellants exist)

Description of all propulsion systems (cold gas, mono-propellant, bi-propellant, electric, nuclear): None.

Identification, including mass and pressure, of all fluids (liquids and gases) planned to be on board and a description of the fluid loading plan or strategies, excluding fluids in sealed heat pipes: None

Fluids in Pressurized Batteries: None

The Corvus-BC3 satellite uses four unpressurized standard COTS Lithium-Ion battery cells in each spacecraft. The energy capacity of each battery is 12 W-Hrs. The total capacity energy capacity per spacecraft is 48 W-Hrs.

Description of attitude control system and indication of the normal attitude of the spacecraft with respect to the velocity vector: The Corvus-BC3 spacecraft attitude will be controlled initially by 5 magnetorquer coils embedded in the solar arrays, which will allow the satellite to be aligned relative to the Earth's magnetic field. These will allow the satellite detumble and align with the magnetic field.

- An *inertial mode* that is optimized for solar power generation from the satellite. The spacecraft's large fixed panel and deployable panel will be oriented towards the sun. This mode will make use of magnetometers, sun sensors, reaction wheels, and magnetic torquers to orient the spacecraft correctly.
- A *targeted tracking mode*, which will allow the Imager or Ka-Band antenna to be directed at any location on the Earth's surface. This mode is used for taking multi-spectral imagery and for downlinking payload data to a Ka-band ground station. This mode will make use of reaction wheels and a star tracker to orient the spacecraft.

Description of any range safety or other pyrotechnic devices: None. The spacecraft deploys its antenna and panels using a burn wire system. System power is locked off during launch by two series and two parallel deployment switches but, the QUADPACK prevents any form of premature deployment, in any case. The antenna and panel spring constants are very low.

Description of the electrical generation and storage system: Standard COTS Lithium-Ion battery cells are charged before payload integration and provide electrical energy during the eclipse portion of the satellites' orbit. A series of Triple Junction Solar Cells generate a maximum on-orbit power of approximately 34 watts at the end-of-life of the mission (5 years for calculation purposes). Typical operational mode for the satellite consumes 17 watts of power on average. The charge/discharge cycle is managed by a power management system overseen by the Flight Computer.

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

Identification of any radioactive materials on board: None

<u>ODAR Section 3</u>: Assessment of Spacecraft Debris Released during Normal Operations:

Identification of any object (>1 mm) expected to be released from the spacecraft any time after launch, including object dimensions, mass, and material: None.

Rationale/necessity for release of each object: N/A.

Time of release of each object, relative to launch time: N/A.

Release velocity of each object with respect to spacecraft: N/A.
Expected orbital parameters (apogee, perigee, and inclination) of each object after release: N/A.

Calculated orbital lifetime of each object, including time spent in Low Earth Orbit (LEO): N/A.

Assessment of spacecraft compliance with Requirements 4.3-1 and 4.3-2 (per DAS v2.0.2)
4.3-1, Mission Related Debris Passing Through LEO: COMPLIANT
4.3-2, Mission Related Debris Passing Near GEO: COMPLIANT

<u>ODAR Section 4</u>: Assessment of Spacecraft Intentional Breakups and Potential for Explosions.

Potential causes of spacecraft breakup during deployment and mission operations: There is no credible scenario that would result in spacecraft breakup during normal deployment and operations.

Summary of failure modes and effects analyses of all credible failure modes which may lead to an accidental explosion: The in-orbit failure of a battery cell protection circuit could lead to a short circuit resulting in overheating and a very remote possibility of battery cell explosion. The battery safety systems discussed in the FMEA (see requirement 4.4-1 below) describe the combined faults that must occur for any of seven (7) independent, mutually exclusive failure modes to lead to such 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 (EOM) including method of passivation and amount which cannot be passivated:
Four (4) 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:

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 spacecraft due to the lack of penetration energy to the multiple enclosures surrounding the batteries.

Probability: Extremely Low. It is believed 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: Protoflight level sine burst, sine and random vibration in three axes of both spacecraft, thermal vacuum cycling of both spacecraft and extensive functional testing followed by maximum system rate-limited charge and discharge cycles were performed to prove that no internal short circuit sensitivity exists. Additional environmental and functional testing of the batteries at the power subsystem vendor facilities were also conducted on the batteries at the component level.

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 if the feasibility of an out-of-control thermal rise in the cell. Cells were 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 device failure or terminal contact with conductors not at battery voltage levels (due to abrasion or inadequate proximity separation).

Mitigation 3: This failure mode is negated by:

a) qualification tested short circuit protection on each external circuit,

b) design of battery packs and insulators such that no contact with nearby board traces is possible without being caused by some other mechanical failure,

c) 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 effects 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. There are no moving parts in the proximity of the batteries.

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

Failure Mode 6: Low level current leakage or short-circuit through battery pack case or due to moisture-based degradation of insulators.

Mitigation 6: These modes are negated by:

a) battery holder/case design made of non-conductive plastic, and

b) operation in vacuum such that no moisture can affect insulators.

Combined faults required for realized failure: Abrasion or piercing failure of circuit board coating or wire insulators **AND** dislocation of battery packs **AND** failure of battery terminal insulators **AND** failure to detect such failures in environmental tests must occur to result in this failure mode.

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 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 post-mission 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: Corvus-BC 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, most 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.

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.

<u>ODAR Section 5</u>: Assessment of Spacecraft Potential for On-Orbit Collisions

Assessment of spacecraft compliance with Requirements 4.5-1 and 4.5-2 (per DAS v2.0.2, and calculation methods provided in NASA-STD-8719.14, section 4.5.4):

Requirement 4.5-1. Limiting debris generated by collisions with large objects when operating in Earth orbit:

"For each spacecraft and launch vehicle orbital stage in or passing through LEO, the program or project shall demonstrate that, during the orbital lifetime of each spacecraft and orbital stage, the probability of accidental collision with space objects larger than 10 cm in diameter is less than 0.001 (Requirement 56506)."

Large Object Impact and Debris Generation Probability: 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 post-mission disposal operation, including passivation and maneuvering: None

<u>ODAR Section 6</u>: Assessment of Spacecraft Post-mission Disposal Plans and Procedures

6.1 Description of spacecraft disposal option selected: The satellite will de-orbit naturally by atmospheric re-entry. There is no propulsion system.

6.2 Plan for any spacecraft maneuvers required to accomplish post-mission disposal: None

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

Spacecraft Mass: 11.5 kg

Cross-sectional Area: 0.124 m^2

(Calculated by DAS 2.0.2). Area to mass ratio: 0.124/11.5 = 0.0108 m^2/kg

6.4 Assessment of spacecraft compliance with Requirements 4.6-1 through 4.6-5 (per DAS v 2.0.2 and NASA-STD-8719.14 section): Requirement 4.6-1. Disposal for space structures passing through LEO:

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

a. Atmospheric reentry option: Leave the space structure in an orbit in which natural forces will lead to atmospheric reentry within 25 years after the completion of mission but no more than 30 years after launch; or Maneuver the space structure into a controlled de-orbit trajectory as soon as practical after completion of mission.

b. Storage orbit option: Maneuver the space structure into an orbit with perigee altitude greater than 2000 km and apogee less than GEO - 500 km.

c. Direct retrieval: Retrieve the space structure and remove it from orbit within 10 years after completion of mission."

Analysis: The Corvus-BC3 satellite method of disposal is COMPLIANT using method "a." The spacecraft will be left in a 505 km near-circular orbit, reentering in approximately 5.0 years after launch with orbit history as shown in Figure 2 (analysis assumes an approximate random tumbling behavior).

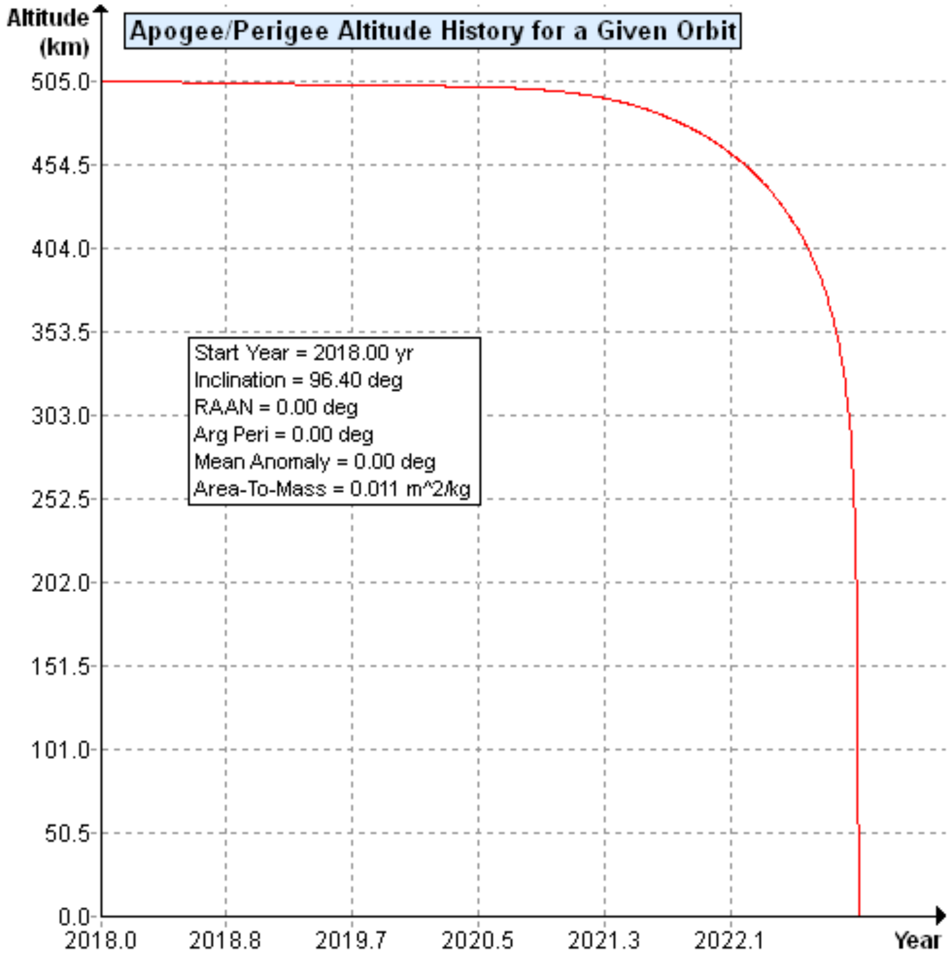

Figure 2: Corvus-BC3 Orbit History

Requirement 4.6-2. Disposal for space structures near GEO:
Analysis is not applicable.

Requirement 4.6-3. Disposal for space structures between LEO and GEO:
Analysis is not applicable.

Requirement 4.6-4. Reliability of Post-mission Disposal Operations:
Analysis is not applicable. The satellite will reenter passively without post mission disposal operations within the allowable timeframe.

<u>ODAR Section 7</u>: Assessment of Spacecraft Reentry Hazards:

Assessment of spacecraft compliance with Requirement 4.7-1: Requirement 4.7-1. Limit the risk of human casualty:

"The potential for human casualty is assumed for any object with an impacting kinetic energy in excess of 15 joules:
a) For uncontrolled reentry, the risk of human casualty from surviving debris shall not exceed 0.0001 (1:10,000) (Requirement 56626)."
Summary Analysis Results: DAS v2.0.2 reports that Corvus-BC1 and Corvus-BC2 are COMPLIANT with the requirement. The critical values reported by the DAS software are:

- Demise Altitude = 77.9 km
- Debris Casualty Area = 0.000000
- Impact Kinetic Energy = 0.000000

This is expected to represent the absolute maximum casualty risk, as calculated with DAS's limited modeling capability. The DAS Output Summary Follows:

=============== End of Requirement 4.3-1 ===============
09 19 2017; 18:58:14PM Processing Requirement 4.3-2: Return Status : Passed

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

=============== End of Requirement 4.3-2 ===============
09 19 2017; 18:58:19PM Requirement 4.4-3: Compliant

=============== End of Requirement 4.4-3 ===============
09 19 2017; 19:12:20PM Processing Requirement 4.5-1: Return Status : Passed

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

INPUT

Space Structure Name = CorvusBC3
Space Structure Type = Payload
Perigee Altitude = 505.000000 (km)
Apogee Altitude = 505.000000 (km)
Inclination = 96.400000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Mean Anomaly = 0.000000 (deg)
Final Area-To-Mass Ratio = 0.010800 (m^2/kg)

Start Year = 2018.000000 (yr)
Initial Mass = 11.500000 (kg)
Final Mass = 11.500000 (kg)
Duration = 5.000000 (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 = 0.000001
Returned Error Message: Normal Processing
Date Range Error Message: Normal Date Range
Status = Pass

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

=============== End of Requirement 4.5-1 ===============
09 19 2017; 19:13:04PM Requirement 4.5-2: Compliant
09 19 2017; 19:13:07PM Processing Requirement 4.6Return Status : Passed

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

INPUT

Space Structure Name = CorvusBC3
Space Structure Type = Payload

Perigee Altitude = 505.000000 (km)
Apogee Altitude = 505.000000 (km)
Inclination = 96.400000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Mean Anomaly = 0.000000 (deg)
Area-To-Mass Ratio = 0.010800 (m^2/kg)
Start Year = 2018.000000 (yr)
Initial Mass = 11.500000 (kg)
Final Mass = 11.500000 (kg)
Duration = 5.000000 (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

Suggested Perigee Altitude = 505.000000 (km)
Suggested Apogee Altitude = 505.000000 (km)
Returned Error Message = Reentry during mission (no PMD req.).

Released Year = 2022 (yr)
Requirement = 61
Compliance Status = Pass

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

=============== End of Requirement 4.6 ===============
09 20 2017; 18:27:36PM *********Processing Requirement 4.7-1
Return Status : Passed

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

name = CorvusBC3
quantity = 1
parent = 0
materialID = 5
type = Box
Aero Mass = 11.500000
Thermal Mass = 11.500000
Diameter/Width = 0.200000
Length = 0.350000
Height = 0.100000

name = Lenses
quantity = 3
parent = 1
materialID = -1
type = Cylinder
Aero Mass = 0.085000
Thermal Mass = 0.085000

Diameter/Width = 0.070000
Length = 0.040000

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

name = CorvusBC3
Demise Altitude = 77.998566
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = Lenses
Demise Altitude = 75.758469
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****

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

****

Requirements 4.7-1b, and 4.7-1c:
These requirements are non-applicable requirements because Corvus-BC3 does not use controlled reentry.

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)."*

Not applicable to Corvus-BC3. The spacecraft does not use controlled reentry and no debris is expected to survive.

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)."*
Not applicable to Corvus-BC3. It does not use controlled reentry and no debris is expected to survive.

<u>ODAR Section 8</u>: Assessment for Tether Missions
Not applicable. There are no tethers used in the Corvus-BC3 mission.

END of ODAR for Corvus-BC3

Appendix A: Acronyms

Arg peri	Argument of Perigee
CDR	Critical Design Review
Cm	centimeter
COTS	Commercial Off-The-Shelf (items)
DAS	Debris Assessment Software
EOM	End Of Mission
FRR	Flight Readiness Review
GEO	Geosynchronous Earth Orbit
ITAR	International Traffic In Arms Regulations
Kg	kilogram
Km	kilometer
LEO	Low Earth Orbit
Li-Ion	Lithium Ion
m^2	Meters squared
ml	milliliter
mm	millimeter
N/A	Not Applicable.
NET	Not Earlier Than
ODAR	Orbital Debris Assessment Report
OSMA	Office of Safety and Mission Assurance
PDR	Preliminary Design Review
PL	Payload
ISIPOD	ISIS CubeSat Deployer
PSIa	Pounds Per Square Inch, absolute
RAAN	Right Ascension of the Ascending Node
SMA	Safety and Mission Assurance
Ti	Titanium
Yr	year