Astro Digital Ignis Orbital Debris Assessment Report (ODAR)

ASTRO-DIGITAL-IGNIS-ODAR-1.0

This report is presented as compliance with NASA-STD-8719.14, APPENDIX A. Report Version: 1.0, 11/12/2015

Astro Digital US, Inc.

3171 Jay St,
Santa Clara, CA 95054

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DAS Software Version Used In Analysis: v2.0.2

Astro Digital Ignis Orbital Debris Assessment Report
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APPROVAL:

Chris Biddy
CEO, Astro Digital

DocuSigned by:
Chris Biddy
A0AAC0F78C324C3...
4/19/2019

Jan A. King
CTO, Astro Digital

DocuSigned by:
4C541E8F6F63477...
4/19/2019

Patrick Shannon
Program Manager
Ignis Program

DocuSigned by:
Patrick Shannon
5DBEBF7971A04F9...
4/19/2019

Revision Record				
Revision:	Date:	Affected Pages:	Changes:	Author(s):
1.0	4/11/2019	All –Initial	DAS Software Results Orbit Lifetime Analysis	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 all launch missions belong to other organizations. This is not a primary mission of Astro Digital. All other portions of the launch composite are not the responsibility of Astro Digital and the Ignis Program is not the lead launch organization.

Assessment Report Format:

ODAR Technical Sections Format Requirements:

Astro Digital US, 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 Ignis satellite. Sections 9 through 14 apply to the launch vehicle ODAR and are not covered here.

Ignis Space Mission Program:

ODAR Section 1: Program Management and Mission Overview

Program Manager: Patrick Shannon
Mission 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 Spacecraft: Q3 2019
- First Launch: Q3 2019

Mission Overview: The Ignis spacecraft is a technology demonstration spacecraft built to the 6U CubeSat standard. It includes the Apollo Constellation Engine (ACE), a Hall Thruster which ionizes and expels a high density proprietary propellant. The spacecraft will be launched aboard a LauncherOne rocket built by Virgin Orbit.

The spacecraft bus is the Corvus-6 design. The common satellite bus uses reaction wheels, magnetic torque coils, star trackers, magnetometers, sun sensors, and gyroscopes to enable precision 3-axis pointing without the use of propellant.

Launch Vehicles and Launch Sites: *LauncherOne and Cosmic Girl*, Guam International Airport, Guam, United States.

Proposed Initial Launch Date: Q3 2019

Mission Duration: The anticipated lifetime of the spacecraft is ≤ 3 years in LEO.

Launch and deployment profile, including all parking, transfer, and operational orbits with apogee, perigee, and inclination: The selected launch vehicle will transport multiple mission payloads to orbit. The Ignis spacecraft will be deployed into a medium inclination low Earth orbit. Once the final stage has burned out, the satellite payloads will be automatically dispensed. The Ignis spacecraft will deploy a UHF antenna and solar panel once released from the LauncherOne upper stage. The spacecraft will decay naturally from operational orbits within the following orbital parameters:

Nominal Orbital Altitude: 500 km

Eccentricity: 0.0000 to 0.0033

Inclination: 44.5° to 45.5°

The Ignis propulsion system is considered to be experimental, and as such is not being relied on as a viable deorbit method. The spacecraft will be launched into an orbit that will result in a natural orbital decay in less than 5 years.

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

Physical description of the spacecraft:

The Ignis spacecraft uses the standard Corvus-6 bus, which is based on the 6U CubeSat form factor. Basic physical dimensions are 32 cm x 21 cm x 11 cm with a mass of no more than 12 kg. The superstructure is comprised of 6 outer panels with multiple subsystems serving as internal supporting structures bridging between the panels. There are L rails along each of the 32 cm edges which accommodate the deployment of the satellite from the deployer. The bus electronics provide additional internal stability to the structure. The ACE thruster nozzle is located on the -X face of the spacecraft with the thrust axis passing through the center of mass of the spacecraft.

The spacecraft bus includes a spring-loaded UHF antenna and deployable solar panel which are deployed after jettison from the deployer by a burn wire controlled by a software timer 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: Ignis Spacecraft

Total satellite mass at launch, including all propellants and fluids:
Ignis: 12.0 kg +0.0 kg/-1.0 kg

Dry mass of satellites at launch:
Ignis: 10.9 kg +0.0 kg/-1.0 kg

Description of all propulsion systems (cold gas, mono-propellant, bi-propellant, electric, nuclear): The ACE propulsion system ionizes a high density proprietary inert propellant (see Proprietary information Exhibit for details). The ionized propellant is expelled out of the thruster using the Hall effect at an Isp exceeding traditional chemical propulsion systems. The expected thrust and Isp will vary with power input levels. Thrust will not exceed 25 mN maximum. The Ignis propulsion system includes 1.1 kg of propellant. The propulsion system as designed has approximately 12,000 Ns total impulse available.

The propulsion system is pressurized using inert nitrogen. The beginning-of-life pressure is 15 psi, or approximately atmospheric pressure. The tank is manufactured from steel to prevent any lifetime cycling issues from causing an unintended rupture and release of propellants.

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:
Up to 1.1 kilograms of propriety propellant and less than 10 grams of gaseous nitrogen pressurant gas. These fluids will be loaded prior to integration of the spacecraft into the standard CubeSat deployer. There will be approximately zero gauge pressure at the time of propellant loading through launch. The atmospheric pressure present will result in the tank pressurizing to approximately 15 psia during launch. The pressure vessel has been tested to 3x the proof pressure and is qualified for transportation by the DOT, as it is not a pressure vessel while in the atmosphere.

Fluids in Pressurized Batteries: None

The Corvus-6 satellite bus design uses four unpressurized standard Lithium-Ion battery cells in each spacecraft. The energy capacity of each cell is 17.5 W-hrs. The total capacity energy capacity per spacecraft is 70 W-hrs.

There is an additional payload battery system which uses nine unpressurized standard Lithium-ion batteries wired in series to provide a 36 volt output to the ACE thruster. Each cell has a capacity of 17.5 W-hrs for a total battery capacity of 157 W-hrs.

Description of attitude control system and indication of the normal attitude of the spacecraft with respect to the velocity vector: The Ignis spacecraft will be initially controlled by magnetic torque coils embedded in the fixed solar panels of the spacecraft. These will be used to detumble the spacecraft to a low enough rate such that the reaction wheels can take over and provide precision 3-axis attitude control. There are four primary attitude modes.

- A *sun pointing mode* that is optimized for solar power generation from the satellite. The spacecraft's large fixed panels 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 *vector tracking mode*, which will allow the thrust axis to be pointed in any direction in inertial space. This mode will make use of reaction wheels, gyroscope, and star tracker to orient the spacecraft.
- An *ACS idle mode,* which will allow the spacecraft to spin up in a predictable manner while firing the thruster for long durations without the reaction wheels providing any torque.
- A *detumble mode* that the spacecraft will enter after deploying from the launch vehicle or after spinning up during a long duration thruster firing.

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

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 satellite's orbit. The bus batteries are operated in an "all-parallel" arrangement that results in increased safety thanks to natural voltage balancing between cells. The payload battery uses a commercial battery management system to ensure all cells in the string are at the same state of charge, preventing overcharging events. A series of Triple Junction Solar Cells generate a maximum on-orbit power of approximately 36 watts at the end-of-life of the mission (5 years for calculation purposes). Typical bus operations consume 8 watts of power on average. The thruster can consume up to 300 watts in short bursts. 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.

ODAR Section 3: Assessment of Spacecraft Debris Released during Normal Operations:

Identification of any object (>1 mm) expected to be released from the spacecraft any time after launch, including object dimensions, mass, and material: 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

ODAR Section 4: Assessment of Spacecraft Intentional Breakups and Potential for Explosions.

Potential causes of spacecraft breakup during deployment and mission operations:

There are three potential scenarios that could potentially lead to a breakup of the satellite. In order of credibility:

1) Rupture of a propellant tank (N2 pressurant gas and propellant); and
2) Lithium-ion battery cell failure.

Summary of failure modes and effects analyses of all credible failure modes which may lead to an accidental explosion: 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: Thirteen (13) 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, Ignis: 0.0000

Supporting Rationale and FMEA details:

Pressure Tank Explosion:

Effect: A rupture of the propellant tank would release gaseous nitrogen and up to 1.1 kg of liquid propellant. Due to the low pressure (15 psia), the penetrating energy of any debris would be relatively low. The propellant tank is enclosed in the solid aluminum structural panels of the spacecraft. These aluminum walls would contain any solid debris within the body of the spacecraft. Droplets of propellant would be expected to leak out of vent holes in the spacecraft body, but would reenter quickly due to their high area to mass ratio.

Probability: Very low. A structural failure of the tank would need to occur, and the mechanisms by which these failures occur are very well understood. Cubesats are typically volume-limited as opposed to mass-limited. This means that it is very easy to add mass to a given structure to protect against failure, and structural strength margins can be very high. This approach is employed in the design of the pressure vessels for the Ignis spacecraft. Additionally, the tank design will be certified by US DoT for commercial transporation. Whereas typical aerospace components would have a margin of safety under 2, all structures on the Corvus satellite designs have strength to failure margins of 3 or greater.

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: The Ignis satellite includes the ability to fully disconnect the Lithium Ion cells from the charging current of the solar arrays. At End-Of-Life, this feature can be used to completely passivate the batteries by removing all energy from them. In the unlikely event that a battery cell does explosively rupture, the small size, mass, and potential energy, of these small batteries is such that while the spacecraft could be expected to vent gases, the debris from the battery rupture should be contained within the spacecraft due to the lack of penetration energy to the multiple enclosures surrounding the batteries.

The thruster propellant will not be actively vented due to the desire to prevent solidified propellant droplets from causing a debris hazard at survivable orbits. Instead, the thruster will be fired until all propellants and pressurants are expelled.

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: 0.00001; 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: 0.0000; COMPLIANT

Identification of all systems or components required to accomplish any post-mission disposal operation, including passivation and maneuvering: None

ODAR Section 6: 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.

6.2 Plan for any spacecraft maneuvers required to accomplish post-mission disposal: No maneuvers are required to accomplish post-mission disposal. The experimental thruster will be used to attempt to lower the orbit, but this is not a requirement to achieve the described disposal plan.

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

Spacecraft Mass: 12.0 kg (selected as worst case mass)
Cross-sectional Area: 0.124 m^2 (average tumbling)
(Calculated by DAS 2.1.1). Area to mass ratio: 0.0103 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 Ignis spacecraft will follow a concept of operations to ensure a safe disposal within 5 years of the end of the mission. The spacecraft is launched into a circular 500 km altitude orbit. Even if the thruster is unable to provide any impulse, the spacecraft will passively reenter due to atmospheric drag within 3.0 years of the launch date. Any thrust maneuvers will be planned such that the orbit altitude is reduced below 500 km, thereby accelerating the reentry time.

This analysis was performed with the NASA Debris Assessment Software 2.1.1. Figure 2 shows the output data from this analysis.

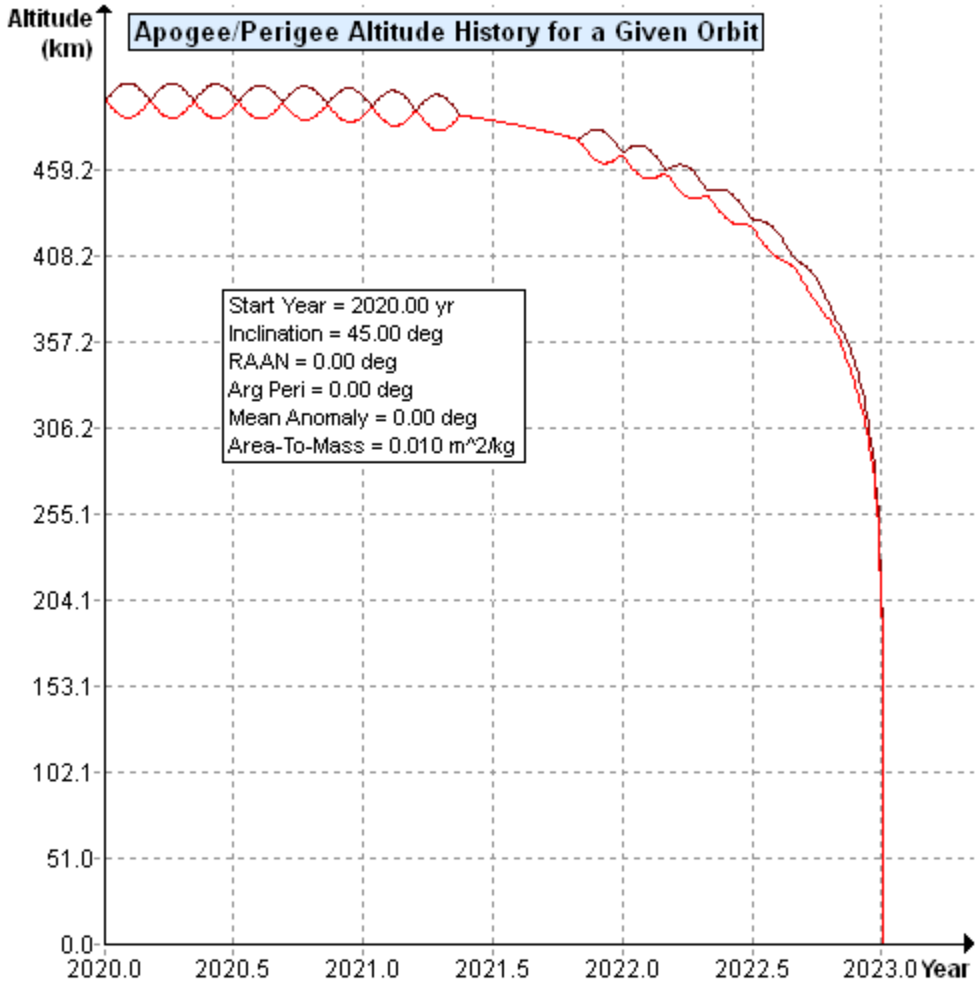

Figure 2: Ignis Orbit History (500 km passive deorbit)

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 v2.1.1 reports that Ignis is COMPLIANT with the requirement. The maximum values reported by the DAS software are:

- Demise Altitude = 0.0 km
- Debris Casualty Area = 0.61 m^2
- Impact Kinetic Energy = 12 Joules
- Risk of Human Casualty = 1:100,000,000

This is expected to represent the absolute maximum casualty risk, as calculated with DAS's modeling capability.

Requirements 4.7-1b, and 4.7-1c:
These requirements are non-applicable requirements because the spacecraft 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 Ignis. The satellite does not use controlled reentry.

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. The satellite does not use controlled reentry.

<u>ODAR Section 8</u>: Assessment for Tether Missions

Not applicable. There are no tethers used on the Ignis mission.

END of ODAR for Ignis.

The raw DAS report as follows for Ignis:

=============== End of Requirement 4.3-1 ===============
04 05 2019; 17:00:33PM   Processing Requirement 4.3-2: Return Status : Passed

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

=============== End of Requirement 4.3-2 ===============
04 05 2019; 17:00:37PM   Requirement 4.4-3: Compliant

=============== End of Requirement 4.4-3 ===============
04 05 2019; 17:03:36PM   Processing Requirement 4.5-1:   Return Status : Passed

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

****INPUT****

Space Structure Name = Ignis
Space Structure Type = Payload
Perigee Altitude = 500.000000 (km)
Apogee Altitude = 500.000000 (km)
Inclination = 45.000000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Mean Anomaly = 0.000000 (deg)
Final Area-To-Mass Ratio = 0.010300 (m^2/kg)
Start Year = 2020.000000 (yr)
Initial Mass = 12.000000 (kg)
Final Mass = 12.000000 (kg)
Duration = 1.000000 (yr)
Station-Kept = False
Abandoned = True
PMD Perigee Altitude = -1.000000 (km)
PMD Apogee Altitude = -1.000000 (km)
PMD Inclination = 0.000000 (deg)
PMD RAAN = 0.000000 (deg)
PMD Argument of Perigee = 0.000000 (deg)
PMD Mean Anomaly = 0.000000 (deg)

****OUTPUT****

Collision Probability = 0.000000
Returned Error Message: Normal Processing
Date Range Error Message: Normal Date Range
Status = Pass

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

=============== End of Requirement 4.5-1 ===============
04 05 2019; 17:09:05PM Requirement 4.5-2: Compliant
04 05 2019; 17:09:12PM Processing Requirement 4.6 Return Status : Passed

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

****INPUT****

Space Structure Name = Ignis
Space Structure Type = Payload

Perigee Altitude = 500.000000 (km)
Apogee Altitude = 500.000000 (km)
Inclination = 45.000000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Mean Anomaly = 0.000000 (deg)
Area-To-Mass Ratio = 0.010300 (m^2/kg)
Start Year = 2020.000000 (yr)
Initial Mass = 12.000000 (kg)
Final Mass = 12.000000 (kg)
Duration = 1.000000 (yr)
Station Kept = False
Abandoned = True
PMD Perigee Altitude = 491.221054 (km)
PMD Apogee Altitude = 501.948986 (km)
PMD Inclination = 44.996381 (deg)
PMD RAAN = 178.806045 (deg)
PMD Argument of Perigee = 149.519091 (deg)
PMD Mean Anomaly = 0.000000 (deg)

****OUTPUT****

Suggested Perigee Altitude = 491.221054 (km)
Suggested Apogee Altitude = 501.948986 (km)

Returned Error Message = Passes LEO reentry orbit criteria.

Released Year = 2023 (yr)
Requirement = 61
Compliance Status = Pass

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

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

04 05 2019; 17:24:06PM   *********Processing Requirement 4.7-1
Return Status : Passed

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

name = Ignis
quantity = 1
parent = 0
materialID = 5
type = Box
Aero Mass = 12.000000
Thermal Mass = 12.000000
Diameter/Width = 0.210000
Length = 0.320000
Height = 0.110000

name = Thruster Structure
quantity = 1
parent = 1
materialID = 65
type = Flat Plate
Aero Mass = 0.154000
Thermal Mass = 0.154000
Diameter/Width = 0.140000
Length = 0.240000

name = Tank
quantity = 1
parent = 1
materialID = 59
type = Cylinder
Aero Mass = 0.400000
Thermal Mass = 0.400000
Diameter/Width = 0.100000

Length = 0.040000

name = AC Component
quantity = 1
parent = 1
materialID = -1
type = Box
Aero Mass = 0.050000
Thermal Mass = 0.050000
Diameter/Width = 0.050000
Length = 0.050000
Height = 0.050000

name = ZC Component
quantity = 1
parent = 1
materialID = -2
type = Box
Aero Mass = 0.015000
Thermal Mass = 0.015000
Diameter/Width = 0.015000
Length = 0.015000
Height = 0.015000

name = TAN Component
quantity = 1
parent = 1
materialID = -3
type = Box
Aero Mass = 0.010000
Thermal Mass = 0.010000
Diameter/Width = 0.010000
Length = 0.010000
Height = 0.007000

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

name = Ignis
Demise Altitude = 77.999115
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = Thruster Structure
Demise Altitude = 0.000000

Debris Casualty Area = 0.613564
Impact Kinetic Energy = 11.517694

****
name = Tank
Demise Altitude = 68.662743
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = AC Component
Demise Altitude = 0.000000
Debris Casualty Area = 0.422500
Impact Kinetic Energy = 7.511644

****
name = ZC Component
Demise Altitude = 0.000000
Debris Casualty Area = 0.378225
Impact Kinetic Energy = 7.542167

****
name = TAN Component
Demise Altitude = 0.000000
Debris Casualty Area = 0.371148
Impact Kinetic Energy = 9.459887

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

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