ADEPT

SIRION Pathfinder-2 Orbital Debris Assessment Report (ODAR)

SIRION Pathfinder-2 - ODAR-1.0

This report is presented as compliance with NASA-STD-8719.14, APPENDIX A. Report Version: 1.0, 7/29/2018

Sirion Global Pty Ltd.

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DAS Software Version Used In Analysis: v2.0.2

Revision Record				
Revision:	Date:	Affected Pages:	Changes:	Author(s):
1.0	7/23/2018	All –Initial	DAS Software Results Orbit Lifetime Analysis	E. Butte

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 Sirion Global Pty Ltd. (“Sirion”), an Australian company that is a wholly-owned subsidiary of Helios Wire Corporation (“Helios Wire”), a Canadian company. This is not a primary mission of the satellite manufacturer, Astro Digital US, Inc (“Astro Digital”). All other portions of the launch composite are not the responsibility of Astro Digital and the SIRION PATHFINDER-2 Program (defined below) is not the lead launch organization.

Assessment Report Format:

ODAR Technical Sections Format Requirements:

Sirion has an ownership interest in the SIRION PATHFINDER-2 satellite (“SIRION PATHFINDER-2” or the “Satellite”). Sirion has an office located at PO Box 127, Crow's Nest NSW 1585. 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 Satellite. Sections 9 through 14 apply to the launch vehicle ODAR and are not covered here.

ODAR Section 1: Program Management and Mission Overview

Relationship between Helios, Sirion, and Astro Digital: Sirion has an ownership interest in the Satellite. Sirion acquired this interest pursuant to an Assignment Agreement dated as of July 30, 2018 between Sirion and its parent Helios of a Satellite Procurement Agreement dated January 31, 2018 between Helios and Astro Digital (the Satellite manufacturer). Pursuant to the assigned Satellite Procurement

Agreement, title to the Satellite shall pass from Astro Digital to Sirion upon Astro Digital's delivery of the Satellite to SpaceFlight, Inc., the US-based launch integrator that is arranging the launch of the Satellite on board a Falcon-9 launch vehicle.

Program/project manager: Eric Butte (Helios Wire), Brian Cooper (Astro Digital)

Senior Management: Scott Larson (Helios Wire), Chris Biddy (Astro Digital), Stephe Wilks (Sirion Global)

Foreign government or space agency participation: Australia.
Summary of NASA's responsibility under the governing agreement(s): N/A

Schedule of upcoming mission milestones:

- Shipment of one (1) spacecraft to Seattle, USA (SpaceFlight's facility): 10 August 2018
- Launch: 15-21 November 2018

Mission Overview: SIRION Pathfinder-2 is a telecommunication satellite designed to collect sensor data from machine-to-machine (M2M) and Internet of Things (IoT) tags through our S-band NGSO ITU filing through Australia ACMA. It will be launched into a sun-synchronous, Low Earth Orbit (LEO) inside a 6U Cubesat deployer as a part of a QUADPACK device developed by Holland-based Innovative Space Logistics (ISL). The deployer is to be included on-board a Falcon 9 launch vehicle, planned for launch during the window 15-21 November 2018. The spacecraft carries two software defined radios in the S-band and C-band frequencies and communicates to user terminal access points with frequent revisit times through the S-band radio and communicates to our Gateways through the C-band radio. The data that is collected from M2M and IoT tags are uplinked to the payload and processed on-board and then downlinked over a miniaturized C-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: Falcon 9, Vandenberg AFB

Proposed Launch Date: 15-21 November 2018

Mission Duration: The anticipated lifetime of the spacecraft (pl.) is ≥ 5 years in LEO.

Launch and deployment profile, including all parking, transfer, and operational orbits with apogee, perigee, and inclination: The Falcon 9 launch vehicle will transport multiple mission payloads to orbit. SIRION Pathfinder-2 will be deployed into an approximately sun synchronous near-circular 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. SIRION Pathfinder-2 will deploy a UHF antenna and one solar panel after 30 minutes once deployed from a QUADPACK deployer from ISL. The spacecraft will be launched into a sun synchronous orbit with the following parameters.

Apogee: 575 ± 10 km
Perigee: 575 ± 10 km
Inclination: 97.7° ± 0.1°

The spacecraft will then begin an orbit raising campaign to increase its orbital altitude to have semi-major axis of 650 km. After communication services have completed, the spacecraft will lower its orbit to a maximum altitude of 620 km. The spacecraft carries more than enough propellant to reach even lower altitudes, and this will be attempted to reduce orbital lifetime. However, 620 km is low enough to guarantee a passive reentry within 25 years, so this is the worst-case assumption that will be used for analysis.

ODAR Section 2: Spacecraft Description:

Physical description of the spacecraft: SIRION Pathfinder-2 is based on the 6U Cubesat form factor. Basic physical dimensions are 366 mm x 226 mm x 116 mm with a mass of approximately 9.5 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 Communication Payload, the Attitude Control Module, the Propulsion Module and the Data and Power Module. The design includes a spring-loaded UHF antenna and one solar panel 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.

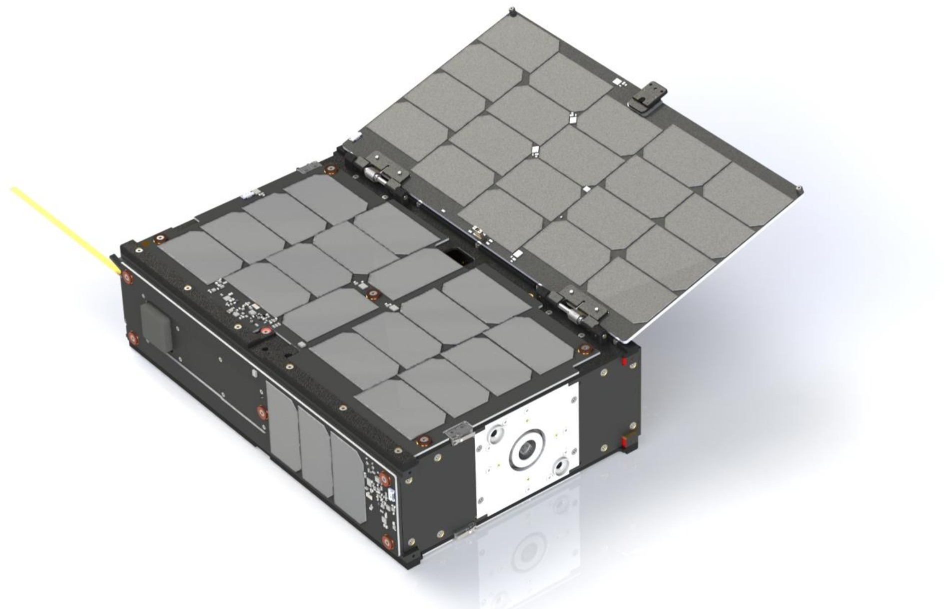

Figure 1: SIRION Pathfinder-2 *Spacecraft*

Total satellite mass at launch, including all propellants and fluids: 9.5 kg.

Dry mass of satellites at launch: 9.3 kg.

Description of all propulsion systems (cold gas, mono-propellant, bi-propellant, electric, nuclear): The Indium FEEP (Field Emission Electric Propulsion) Motor (IFM) is an electric thruster. The IFM Nano Thruster is a mature technology, developed under ESA contracts for 15 years. During which, more than 100 emitters have been tested as well as an ongoing lifetime test that has demonstrated more than 18,000 hours of firing without degradation of the emitter performance. The IFM has dynamic precise thrust control. The thrust can be controlled through the electrode voltages, providing excellent controllability over the full thrust range and a low thrust noise.
The IFM Nano Thruster contains no moving parts and the propellant is in its solid state at room temperature. Avoiding any liquid and reactive propellants and pressurized tanks, which significantly simplifies handling, integration, and launch procedures, and it's a safe and inert system compliant with all launcher requirements. The thruster includes 200 grams of solid inert indium propellant that is slowly ablated and used as propellant during operations.

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 SIRION Pathfinder-2 satellite uses four unpressurized standard COTS Lithium-Ion battery cells in each spacecraft. The energy capacity of each battery cell 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 SIRION Pathfinder-2 spacecraft attitude will be controlled initially by 4 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 C and S-Band antennas to be directed at any location on the Earth's surface. This mode is used for aligning the communication antennas and for downlinking payload data to a C-band ground station. This mode will make use of reaction wheels, sun sensors, gyroscope, and reaction wheels 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 12 watts of power on average. The charge/discharge cycle is managed by a power management system overseen by the Flight Computer. During thruster firings, the spacecraft can momentarily consume up to 40 watts for short periods of time

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 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 – Disabling of input photovoltaic converters. This will result in a gradual discharge and full shutdown of all 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: SIRION Pathfinder-2 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.
SIRION Pathfinder-2 includes an electric propulsion system with a solid ingot of indium used as a propellant. Since the IFM Nano thruster doesn't have any pressure vessels, there is no hazardous stored energy in the propulsion system.

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.

ODAR Section 5: Assessment of Spacecraft Potential for On-Orbit Collisions

The flight profile for the satellite begins with initial checkouts of the satellite subsystems after orbit insertion at 575 +/- 10 km. Once checkouts are complete. Ground operators will commence the orbit raising procedure. This consists of firing the thruster for 20 minutes centered around the maximum latitude of the orbit (closest to the north pole) every orbit for a period of 2.5 months. The apogee of the orbit increases towards the direction of the south pole with each firing, but as the orbit raising is very gradual, the apogee precesses 180 degrees during the orbit raising campaign. Therefore, the ultimate orbit is a circular orbit with a semi-major axis of 650 km. After the primary communications mission has been completed, the spacecraft initiates a de-orbit campaign to lower the semi-major axis to below 620 km. This is accomplished in reverse of the orbit raising campaign; by firing at the south pole over the course of a minimum of 25 days. The orbit will be lowered even further if possible to speed the reentry of the spacecraft.

Orbit transfers are designed to control the semi-major axis and eccentricity, which keeps the satellite within 1 km of desired altitude and maintains a near-circular orbit. Each 20 minute firing raises the opposite side of the orbit by approximately 200 meters. Ground operators will use the on-board GPS receiver for precise tracking and propulsion maneuvers assessment. Ground operators will also use Two-Line Elements (TLEs) from JSpOC for back up orbit determination.

Ground operators will interface closely with the Joint Space Operations Center (JSpOC) to assess the collision risk of trajectories resulting from propulsive maneuvers and to share data on maneuver plans with other operators, following the processes outlined in the JSpOC's Spaceflight Safety Handbook for Satellite Operators. Specifically, operators will submit an early orbit maneuver plan to JSpOC and request an Early Orbit Conjunction Assessment (CA) before launch. Ground operators will then provide ephemeris to the JSpOC after the firing. During operation, ground operators will share maneuver plans with the JSpOC before executing each maneuver. Operators will also request CA screenings from the JSpOC before ceasing operations. This will allow the JSpOC to coordinate with NASA to ensure the deorbiting spacecraft will safely descend through the International Space Station's orbit.

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: 0.000003; 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

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. The propulsion system will be utilized to place the spacecraft in an orbit such that the orbital lifetime before passive reentry is less than 25 years.

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: 9.5 kg

Cross-sectional Area: 0.142 m^2

Area to mass ratio: 0.142/9.5 = 0.0149 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 SIRION- Pathfinder-2 satellite method of disposal is COMPLIANT using method "a." The spacecraft will be left in a 620 x 620 km near-circular orbit, reentering in approximately 5970 days (16.35 years) after end of operations with orbit history as shown in Figure 2 (analysis assumes an approximate random tumbling behavior).

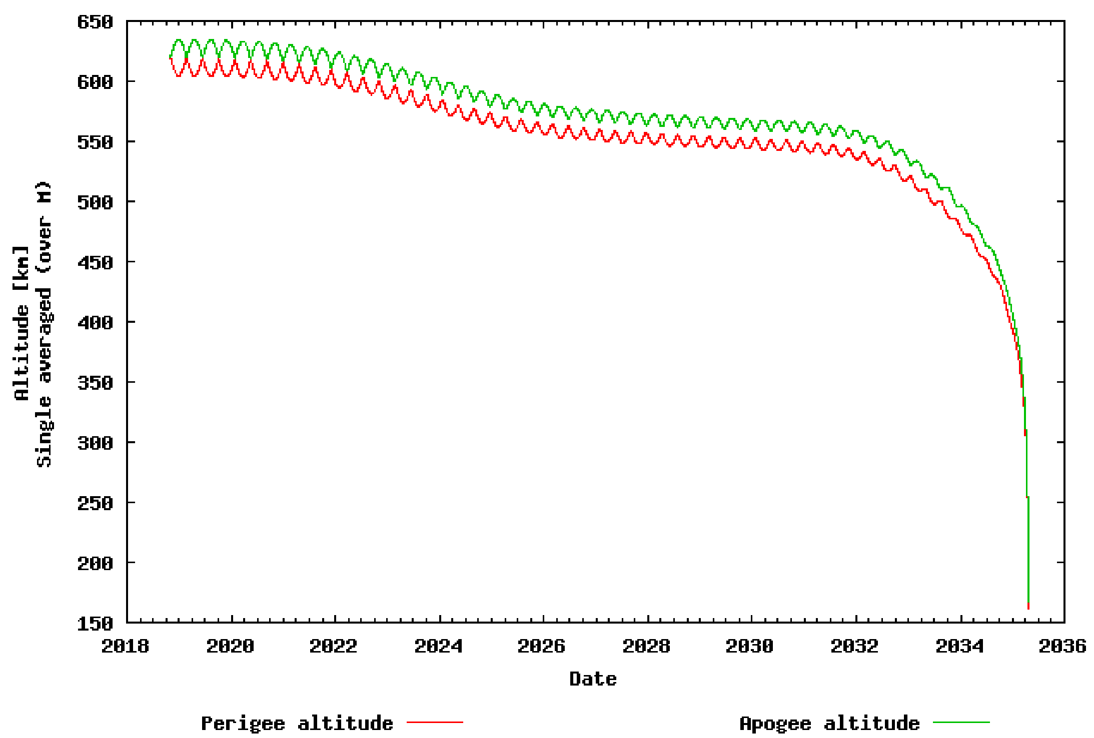

Figure 2: SIRION Pathfinder-2 Orbit History

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.

The satellite will reenter with solar panel and antenna deployed, increasing its cross-sectional area. This solar panel deployment system has been tested to GEVS qualification levels and has functioned perfectly in three separate satellite launches. The deployment system uses a very simple spectra braid to hold the panel closed with a Nickel-Chrome wire to melt the spectra braid. A similar mechanism has been used to deploy antennas for the Astro Digital satellites and has deployed successfully 10 times on-orbit with zero failures.

ODAR Section 7: 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 SIRION Pathfinder-2 is COMPLIANT with the requirement. The critical values reported by the DAS software are:

- Demise Altitude = 55.6 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:

```
10 17 2017; 07:09:54AM     DAS Application Started
10 19 2017; 06:52:28AM     Processing Requirement 4.3-1:
     Return Status :  Not Run

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

=============== End of Requirement 4.3-1 ===============
10 19 2017; 06:52:31AM     Processing Requirement 4.3-2: Return
Status : Passed

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

=============== End of Requirement 4.3-2 ===============
10 19 2017; 06:52:36AM     Requirement 4.4-3:  Compliant

=============== End of Requirement 4.4-3 ===============
10 19 2017; 06:52:55AM     Processing Requirement 4.5-1:
     Return Status :  Passed

==============
Run Data
==============
```

```
**INPUT**

      Space Structure Name = SIRION Pathfinder-2
      Space Structure Type = Payload
      Perigee Altitude = 600.000000 (km)
      Apogee Altitude = 600.000000 (km)
      Inclination = 97.750000 (deg)
      RAAN = 0.000000 (deg)
      Argument of Perigee = 0.000000 (deg)
      Mean Anomaly = 0.000000 (deg)
      Final Area-To-Mass Ratio = 0.006620 (m^2/kg)
      Start Year = 2017.000000 (yr)
      Initial Mass = 10.000000 (kg)
      Final Mass = 10.000000 (kg)
      Duration = 2.000000 (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 = 0.000004
      Returned Error Message: Normal Processing
      Date Range Error Message: Normal Date Range
      Status = Pass

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

=============== End of Requirement 4.5-1 ===============
10 19 2017; 06:53:01AM     Processing Requirement 4.6 Return
Status :  Passed

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

**INPUT**

      Space Structure Name = SIRION Pathfinder-2
      Space Structure Type = Payload

      Perigee Altitude = 600.000000 (km)
      Apogee Altitude = 600.000000 (km)
      Inclination = 97.750000 (deg)
      RAAN = 0.000000 (deg)
      Argument of Perigee = 0.000000 (deg)
```

```
      Mean Anomaly = 0.000000 (deg)
      Area-To-Mass Ratio = 0.006620 (m^2/kg)
      Start Year = 2017.000000 (yr)
      Initial Mass = 10.000000 (kg)
      Final Mass = 10.000000 (kg)
      Duration = 2.000000 (yr)
      Station Kept = False
      Abandoned = True
      PMD Perigee Altitude = 599.264521 (km)
      PMD Apogee Altitude = 599.264521 (km)
      PMD Inclination = 97.789963 (deg)
      PMD RAAN = 355.932180 (deg)
      PMD Argument of Perigee = 8.944930 (deg)
      PMD Mean Anomaly = 0.000000 (deg)

**OUTPUT**

      Suggested Perigee Altitude = 599.264521 (km)
      Suggested Apogee Altitude = 599.264521 (km)
      Returned Error Message = Passes LEO reentry orbit criteria.

      Released Year = 2038 (yr)
      Requirement = 61
      Compliance Status = Pass

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

=============== End of Requirement 4.6 ===============
10 19 2017; 06:55:36AM     *********Processing Requirement 4.7-1
      Return Status :  Passed

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

name = SIRION Pathfinder-2
quantity = 1
parent = 0
materialID = 5
type = Box
Aero Mass = 10.000000
Thermal Mass = 10.000000
Diameter/Width = 0.230000
Length = 0.350000
Height = 0.110000

name = SIRION Pathfinder-2
quantity = 1
parent = 1
materialID = 5
type = Box
Aero Mass = 10.000000
Thermal Mass = 10.000000
```

```
Diameter/Width = 0.230000
Length = 0.350000
Height = 0.110000

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

name = SIRION Pathfinder-2
Demise Altitude = 77.999683
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

*************************************
name = SIRION Pathfinder-2
Demise Altitude = 55.640999
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 SIRION Pathfinder-2 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 SIRION Pathfinder-2. 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 SIRION Pathfinder-2. It does not use controlled reentry and no debris is expected to survive.

ODAR Section 8: Assessment for Tether Missions

Not applicable. There are no tethers used in the SIRION Pathfinder-2 mission.

END of ODAR for SIRION Pathfinder-2

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