Orbital Debris Assessment Report for Space AI SAI-2 Mission per NASA-STD-8719.14B

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1 INTRODUCTION

1.1 Objective

The purpose of this report is to satisfy the orbital debris requirements listed in NPR 8715.6B, NASA Procedural Requirements for Limiting Orbital Debris and Evaluating the Meteoroid and Orbital Debris Environments for SAI-2 Mission. This report addresses the requirements stated on NASA-STD-8719.14B, Process for Limiting Orbital Debris covering sections 2 to 8 as indicated, sections 9 to 14 falls under the requirements for the primary mission responsible and for that are not presented here.

1.2 About Space AI

Space AI is a company based in Silicon Valley, California USA that has developed Node 1 as its core product. Node 1 will be the base infrastructure for the 4th industrial revolution. Node 1 is the next generation of SDR and the 5 senses of smart devices. Capable to deliver information data at the fastest speed, including the most advanced sensors for nano accuracy geolocation in the market; allowing innovators to connect and share resources with Distributed Computing Systems, Networks or Collaborative Systems and enhance their own devices, unmanned vehicles, robot systems or spacecrafts through standard connectors.

The company has sites in USA, Argentina, Mexico and Israel with a highly qualified team capable to deal with any kind of project. Space AI is currently working in different countries and sectors. Among them: agribusiness, transportation, space industry, solar energy and autonomous vehicles.

1.3 Self-assessment of the ODAR

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

NASA-STD-8719.14B – 2019-04-25 Orbital Debris Assessment Report Evaluation: SAI-2 Mission		
Requirement	**Status**	**Comments**
4.3-1a	Compliant	No planned debris release in LEO.
4.3-1b	Compliant	No planned debris release in LEO.
4.3-2	Compliant	NO planned debris release in near GEO for normal operations
4.4-1	Compliant	On board energy source (batteries) incapable of debris producing failure
4.4-2	Compliant	On board energy source (batteries) incapable of debris producing failure
4.4-3	Not applicable	No planned breakups
4.4-4	Not applicable	No planned breakups
4.5-1	Compliant	Probability well below 0.001
4.5-2	Not applicable	No post-mission disposal operation
4.6-1(a)	Compliant	
4.6-1(b)	Not applicable	
4.6-1(c)	Not applicable	
4.6-2	Not applicable	Non GEO Mission
4.6-3	Not applicable	Non MEO Mission
4.6-4	Compliant	Passive disposal
4.7-1	Compliant	Non-credible risk of human casualty
4.8-1	Not applicable	No tether mission

2 ODAR Section 1: Program Management and Mission Overview

2.1 Program Management

Program Manager: David Garcia
Mission Manager: Enrique Pacheco

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: 1st week of November 2021
- Launch: Window start 1st week of December 2021

2.2 Mission Summary Description

The overall goal of the SAI-2 mission is to test and operate a prototype Space AI supercomputer and advanced communications system in a space environment, to provide space heritage, and performance feedback for the design.

SAI-2 is a 6U CubeSat with an overall dimension of 10 cm X 20 cm X 30 cm, fully compliant with the 6U CubeSat Design Specification Revision 1.0 (CP-6UCDS-1.0 California Polytechnic State University). It is intended as a technology demonstrator for advanced communications and supercomputer architectures. Node I is an advanced SDR communications card designed to be used on CubeSats missions. The main purpose for using the Node I, is to test the main operational aspects of the card (radios, sensors) and the transmission performance in the broad range of operation for the radio, in order to test as much as possible the radio performance, both for Space-to-Earth and Earth-to-Space operations. The satellite will carry a Space AI MCC supercomputer and COTS AGP and supercomputer cards in order to test different high-computing intensive algorithms under space environment.

2.3 Launch Vehicle and Launch Site

The satellite will be delivered to Maverick Space Systems, Inc., that is acting as the Launch Service Provider to Space AI, no later than October 28th and they will deliver no later than November 5th to Space X to be integrated to a Falcon 9 rocket.

2.4 Launch Date and Mission Duration

The experimental SAI-2 mission is a free flying mission and is schedule to flight no later than the end of December 2021. The intended mission duration is 1 year.

2.5 Launch Profile

The Falcon 9 launch vehicle will transport multiple mission payloads to orbit, and SAI-2 spacecraft will be deployed into in a circular LEO orbit of 550KM at 97 degrees. The launch vehicle will determine the moment for the satellite to be dispensed. The SAI-2 spacecraft will be activated based on the deploy switches, this will activate the computer and a safety timer, once the timer reaches the end, the satellite will initiate the first set of operations. The satellite will activate the transmitter to indicate it's alive and operational.

Nominal Orbital Altitude: 500 km
Eccentricity: expected to be as close to 0.0000
Inclination: 97º

2.6 Spacecraft Maneuver Capability

The SAI-2 will not have any active device for operational manoeuvres; neither have any kind of propellant. The spacecraft will decay naturally from operational orbits within the stated orbital parameters in a natural orbital decay in less than 3 years.

2.7 Reason for selection of operational orbits

The SAI-2 Mission will fly as a secondary payload on a Space X Falcon 9 flight where the primary payloads belong to other organizations. This is not a primary mission of Space AI. All other portions of the launch vehicle are not the responsibility of Space AI and is not the lead launch organization. The orbital parameters were provided by the launcher organization, the selection was based purely on the soonest launch opportunity available.

2.8 Identification of interactions

The SAI-2 Mission will not expect to have or produce any physical interference with other operational spacecraft.

3 ODAR Section 2: Spacecraft Description

3.1 SAI – 2 Spacecraft Description

SAI-2 is a 6U CubeSat with an overall dimension of 10 cm X 20 cm X 30 cm, fully compliant with the 6U CubeSat Design Specification Revision 1.0 (CP-6UCDS-1.0 California Polytechnic State University). It is intended as a technology demonstrator for advanced communications and supercomputer architectures. Node I is an advanced SDR communications card designed to be used on CubeSats missions. The main purpose for using the Node I, is to test the main operational aspects of the card (radios, sensors) and the transmission performance in the broad range of operation for the radio, in order to test as much as possible the radio performance, both for Space-to-Earth and Earth-to-Space operations. The satellite will carry a Space AI MCC supercomputer and COTS AGP and supercomputer cards in order to test different high-computing intensive algorithms under space environment:

1. Node I Space AI card
2. MCC Space AI Supercomputer card
3. Space AI Node IO subsystem
4. AGP COTS card
5. COTS high computing card
6. Atmospheric Gas Spectrometer
7. SSD Memory array
8. iMTQ Magnetorquer Board ADCS
9. 4 reaction wheel 3-axis stabilization subsystem
10. Wideband antennas
11. Deployable Gallium Arsenide Solar Cells panels
12. EPS system with a 350WHr Lithium-Ion Battery packs
13. TT&C UHF antenna

3.2 Transmitting Devices Description

The SAI-2 satellite carry-on 3 Analog Devices AD9375 chips, integrated on the Node I Space AI card, also will carry 4 Lime Microsistems LMS6002D SDR inside the Node IO.

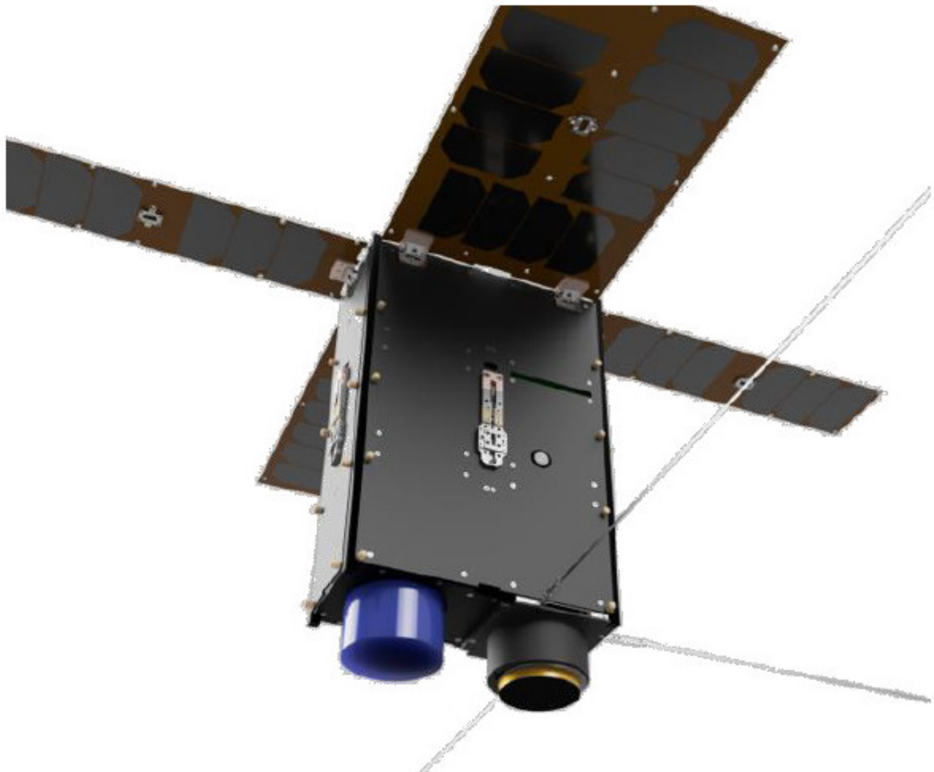

Figure 1. SAI-2

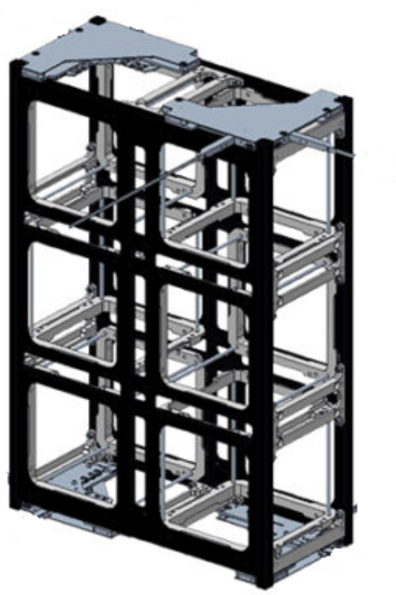

Figure 2. SAI-2 internal structure

3.3 Total satellite mass at launch, including all propellants and fluids.

The table 1 shown the detailed mass description, the satellite does not have any propellant or fluid.

Table 1 . Detailed Component Mass Description

ID	Subsystem	Units	Nominal Mass per Card (grs)	Mass per Subsystem (grs)
M1	Space AI Node 1	3	132	396
M2	Space AI MCC	3	130	390
M3	Space AI Node IO	4	200	800
M4	Spectrometer	1	280	280
M5	AI Computer N	2	250	500
M6	AI Computer X	2	400	800
M7	2TB SSD Card	16	8.1	129.6
M8	SSD Card Support	16	15	240
M9	350WHr 56 cell Battery pack (8 cell per card, 7 cards)	1	1050	1050
M10	EPS IPDU Power Distribution Unit	4	58	232
M11	EPS IPCU Power Conditioning Unit	2	58	116
M12	EPS IPBU Power Battery Unit	2	49	98
M13	iMTQ Magnetorquer Board ADCS	1	196	196
M14	4 Reaction Wheel 3-axis ADCS subsystem	1	940	940
M15	RF PA LNA	4	15	60
M16	Antenna subsystem	1	115	115
M17	Deployable solar panels (2x6U + 2X3U + 1x2U)	1	1000	1000
M18	6U structure	1	1100	1100
M18	Cables, screws and other components	1	1000	1000
			Mass Total	**8,793**

3.4 Dry mass of satellites at launch
8.793 Kg.

3.5 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

The Satellite does not have any fluid (liquid or gas).

3.6 Description of all propulsion systems (cold gas, mono-propellant, bipropellant, electric, nuclear)

The Satellite does not have any propulsion system.

3.7 Description of all active and/or passive attitude control systems with an indication of the normal attitude of the spacecraft with respect to the velocity vector.
The satellite will have a magnetorquer board ADCS for basic detumbling and ideally keeping the satellite flying with the +Z face Earth oriented. Also will have a 4 reaction wheel subsistem for active precision pointing in order to keep the platform oriente to the Sun as much as possible.

The performance of the ADCS active system is shown in the Table 2:

Table 2. Performance parameters of the reaction wheel system

Parameter	Condition	Min	Typ.	Max	Unit
Torque	Continuous	-1.5		1.5	mNm
Speed	Maximum	-6000		6000	rpm
Speed	At max torque	-3500		3500	rpm
Momentum	At max speed		19		mNms
Supply voltage		4.9	5	5.1	V
Supply current	At maximum torque			500	mA
Supply current	At 4500 rpm (zero torque)		60		mA
Control accuracy	Speed control		0.5		rpm

And the figure 3 shown a picture of the system and the expected place inside the satellite:

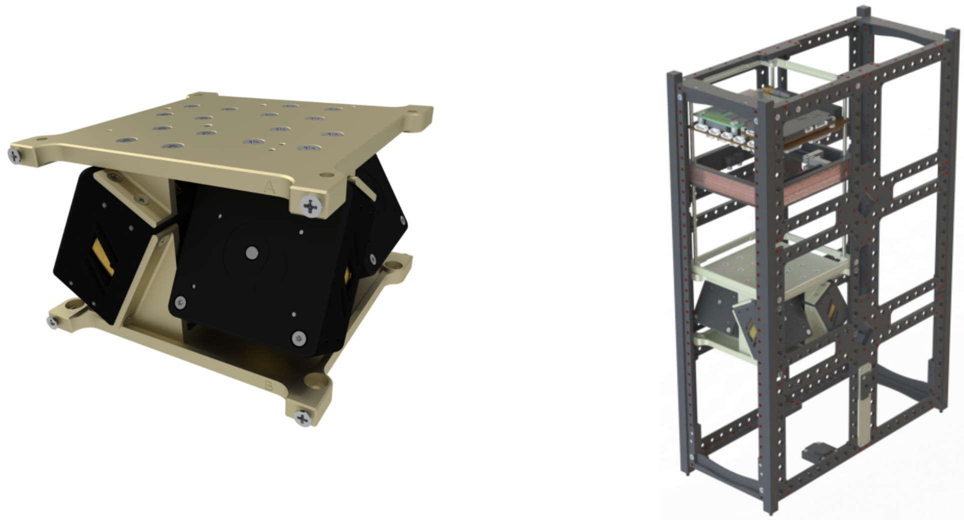

Figure 3. Reaction wheel system and its internal place on SAI-2.

The iMTQ Magnetorquer Board (Figure 4) have the following main characteristics:

Three-axis magnetometer
Three actuators; two torque rods and one air core torque.
Current sensors for each torque
Temperature telemetry of actuators
Nominal 0.2Am2 actuation per actuator
Suitable to detumble up to 12U (~24kg) CubeSats
Will be used for:
CubeSat detumbling & magnetic attitude control
Reaction wheel desaturation

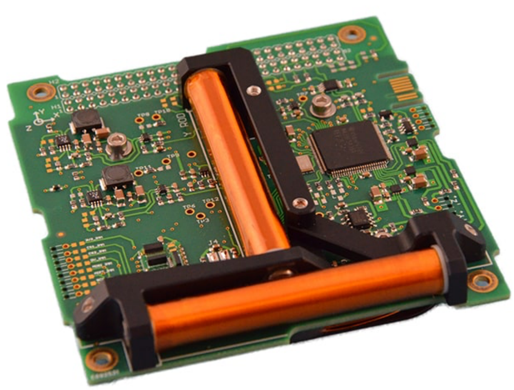

Figure 4. iMTQ Magnetorquer Board.

3.8 Description of any range safety or other pyrotechnic devices.

There is none devices of this type. The spacecraft deploy its antennas using a burn wire system. System power is locked off during launch with a safety mechanism to prevent premature deployment. The antenna spring constants are very low and can be held in place with minum force.

3.9 Description of the electrical generation and storage system.

The satellite will have a 350WHr battery pack made of 7 cards with 8 cells each card of Li-Ion slim battery, the system have the following characteristics:

Chemistry: Li-ion
Typical internal resistance:
    1 to 7 milliohms @ 25°C, Total impedance < 50 milliohms
High discharge rate:
    2C@30mins,4C@10mins,10C@2seconds
High speed charge rate:
    2 times the nominal capacity
Operating Temperature:
    -30 to +80°C w/o CN/TTB option
    -60 to +120°C w/ CN/TTB option
Radiation Tolerance:
    2 years minimum in LEO, 10 years if S/C has NEMEA shielding Outgassdata: TML<1%,CVCM<0.05%
UN38.3, MSDS, CoO certificates
96x90x99.8 mm

Total @BOL
    Voltage (max) 4.2/8.4V
    Current 84/42Ah
    Power 350Whr
    Weigh 1050 grs per 56 cells
Per Module
    Voltage (max) 4.2/8.4V
    Current 12/6Ah
    Power 50Whr
    Weigh 150gr per 8 cells

The system has flight heritage since 2013 in 6 missions in orbit and have been selected to fly in 10 more upcoming U.S. missions from 2020 to 2029

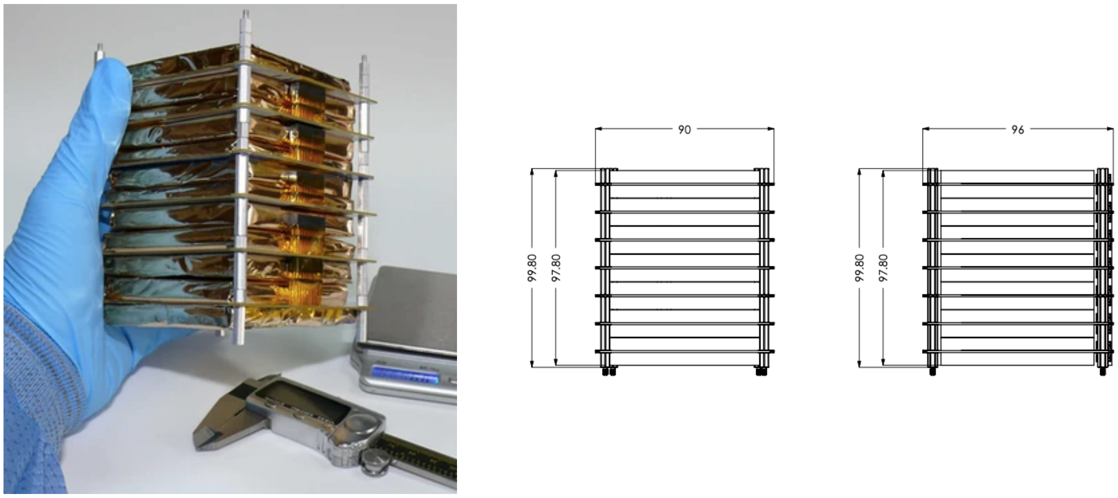

Figure 5. Battery Pack

3.10 Identification of any other sources of stored energy not noted above.

The satellite does not have any other stored energy source.

3.11 Identification of any hazard or radioactive materials on board.

The primary SAI-2 structure is made of aluminum 6061 anodized. All the boards as been made using all standard commercial off the shelf (COTS) materials, electrical components and PCBs. The solar cells are made of Triple Junction Gallium Arsenide.

There is no pressure vessels, hazardous or exotic materials, including radioactive materials.

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

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

None.

4.2 Rationale/necessity for release of each object.

Non applicable.

4.3 Time of release of each object, relative to launch time.

Non applicable.

4.4 Release velocity of each object with respect to spacecraft.

Non applicable.

4.5 Expected orbital parameters (apogee, perigee, and inclination) of each object after release.

Non applicable.

4.6 Calculated orbital lifetime of each object, including time spent in LEO.

Non applicable.

4.7 Assessment of spacecraft compliance with Requirements 4.3-1 and 4.3-2.

***Assessment of Compliance of 4.3-1a**, All debris released during the deployment, operation, and disposal phases shall be limited to a maximum orbital lifetime of 25 years from date of release:*

COMPLIANT – NO planned debris release in LEO for normal operations.

Assessment of Compliance of 4.3-1b*, The total object-time product shall be no larger than 100 object-years per mission. For the purpose of this standard, satellites smaller than a 1U standard CubeSat are treated as mission-related debris and thus are bound by this definition to collectively follow the same 100 object-years per mission deployment limit:*

COMPLIANT – NO planned debris release in LEO for normal operations

Assessment of Compliance of 4.3-2*, Debris passing near GEO: For missions leaving debris in orbits with the potential of traversing GEO (GEO altitude +/- 200 km and +/- 15 degrees inclination), released debris with diameters of 5 mm or greater shall be left in orbits which will ensure that within 25 years after release the apogee will no longer exceed GEO - 200 km or the perigee will not be lower than GEO + 200 km , and also ensures that the debris is incapable of being perturbed to lie within that GEO +/- 200 km and +/- 15 zone for at least 100 years thereafter. For the purpose of this standard, satellites smaller than a 1U standard CubeSat are treated as mission-related debris and thus are bound by this definition to follow this requirement:*

COMPLIANT – NO planned debris release in near GEO for normal operations

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

5.1 Identification of all potential causes of spacecraft breakup during deployment and mission operations.

There are NO plans for designed spacecraft breakups, explosions, or intentional collisions on the SAI-2 mission.

There is only one potential causes of spacecraft breakup during activation after deployment and mission operations caused by a Lithium-ion battery cell failure.

5.2 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. There are 7 possible scenarios of independent, mutually exclusive failure that can lead to an explosion of a Li-On battery cell.[1] :

Failure Scenario 1: Internal short circuit.
Failure Scenario 2: Internal thermal rise due to high load discharge rate.
Failure Scenario 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).
Failure Scenario 4: Inoperable vents.
Failure Scenario 5: Destruction by collapsing.
Failure Scenario 6: Low level current leakage or short-circuit through battery pack case or due to moisture-based degradation of insulators.
Failure Scenario 7: Excess temperatures due to orbital environment and high discharge combined.

All failure scenarios 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.

[1] Astro Digital Ignis Orbital Debris Assessment Report (ODAR) ASTRO-DIGITAL-IGNIS-ODAR-1.0

The probability is extremely low, given that multiple independent (not common mode) faults must occur for each failure scenario to cause the ultimate effect (explosion). In addition, due the short orbital lifetimes the effect of an explosion on the far-term LEO environment is negligible.

Energy management system and battery system has over-current switch protection, overcurrent bus protection, and battery under and over-voltage protection built into the system in order to mitigate any possibility of failure. In addition, and extensive environmental and functional testing for all the circuits, batteries and structural supports involved on these systems has produced a very reliable system.

5.3 Detailed plan for any designed spacecraft breakup, including explosions and intentional collisions.

There are NO plans for designed spacecraft breakups, explosions, or intentional collisions on the SAI-2 mission.

5.4 List of components which are passivated at EOM. List includes method of passivation and amount which cannot be passivated.

350WHr battery pack made of 7 cards with 8 cells each card at 12000 mAh of Lithium-Ion Battery cells.

5.5 Rationale for all items which are required to be passivated, but cannot be due to their design.

The SAI-2 satellite includes the ability to disconnect the batteries from the charging current of the solar arrays. At EOL, 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.

Additional even on the extreme case that the batteries cannot be disconnected from the solar panels still meet Req 4.4-2 by virtue of the HQ OSMA policy regarding CubeSat battery disconnect stating;

> *"CubeSats as a satellite class need not disconnect their batteries if flown in LEO with orbital lifetimes less than 25 years."*[2]

[2] Orbital Debris Assessment for The CubeSats on the ELaNa-XIX Mission per NASA-STD 8719.14ª (FCC) Rev C

5.6 Assessment of spacecraft compliance with Requirements 4.4-1 through 4.4-4.

***Assessment of Compliance of 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 (i.e., every individual free-flying structural object), the program or project shall demonstrate, via failure mode and effects analyses, probabilistic risk assessments, or other appropriate analyses, that the integrated probability of explosion for all credible failure modes of each spacecraft and launch vehicle does not exceed 0.001 (excluding small particle impacts.).*

COMPLIANT – On board energy source (batteries) incapable of debris producing failure

***Assessment of Compliance of 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 and a plan to either 1) deplete all onboard sources of stored energy and disconnect all energy generation sources when they are no longer required for mission operations or postmission disposal or 2) control to a level which cannot cause an explosion or deflagration large enough to release orbital debris or break up the spacecraft. The design of depletion burns and ventings should minimize the probability of accidental collision with tracked objects in space:*

COMPLIANT – On board energy source (batteries) incapable of debris producing failure

***Assessment of Compliance of 4.4-3**, Limiting the long-term risk to other space systems from planned breakups for Earth*

NOT APPLICABLE – No planned breakups

***Assessment of Compliance of 4.4-4**, Limiting the short-term risk to other space systems from planned breakups for Earth orbital missions*

NOT APPLICABLE – No planned breakups

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

6.1 Calculation of spacecraft probability of collision with space objects larger than 10 cm in diameter during the orbital lifetime of the spacecraft.

Calculation of spacecraft probability of collision with space objects larger than 10 cm in diameter during the orbital lifetime of the SAI-2 takes into account both the mean cross-sectional area and orbital lifetime.

The largest mean cross-sectional area (CSA) is that of the satellite with the solar panels deployed, this can create an apparent area in the satellite is rotating of 5,600 cm^2, and a 2,000 cm^2 taking into account the effective solid areas (Figure 6). For this calculation we take the maximum apparent dimensions of the satellite as the largest maximum possible CSA. The minimum value will be taking in account the nominal solid dimensions, according to the following formula values:

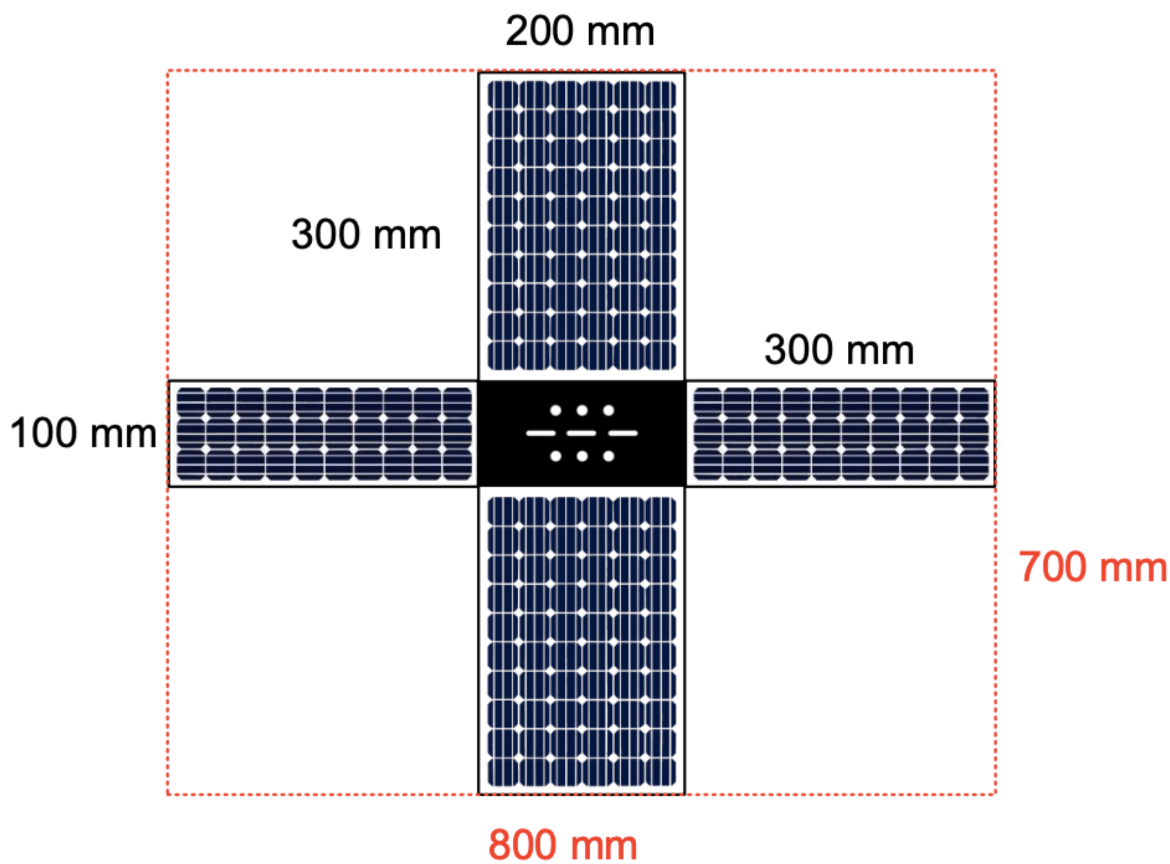

Figure 6. Maximum CSA in red dot lines.

The values used and estimated are:

Mean CSA_{max} = [2*(30x20) + 4*(80x70)]/4 = 5,900 cm^2

Mean CSA_{solid} = 600+600+200+300+300 = 2,000 cm^2

Mean CSA_{min} = [2*(10x20) + 4*(30x20)]/4 = 700.0 cm^2

Mass = 8.793 Kg

CSA_{max} = 0.59 m^2 Ratio-area-to-$mass_{max}$= 0.0670

CSA_{solid} = 0.20 m^2 Ratio-area-to-$mass_{solid}$= 0.02274

CAS_{min} = 0.0700 m^2 Ratio-area-to-$mass_{min}$= 0.0079

Probability of $Collision_{max}$ = 0.00000

Probability of $Collision_{min}$ = 0.00000

6.2 Calculation of spacecraft probability of collision with space objects, including orbital debris and meteoroids, of sufficient size to prevent postmission disposal.

There will be no post-mission disposal operation. As such the identification of all systems and components required to accomplish post-mission disposal operation, including passivation and maneuvering, is not applicable. The only action will be to disconnect de batteries from the solar panels, however this will not have an impact on the postmission disposal that must occur for a natural orbit decay.

6.3 Assessment of spacecraft compliance with Requirements 4.5-1 and 4.5-2.

Assessment of Compliance of 4.5-1*, Limiting debris generated by collisions with large objects when 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 does not exceed 0.001. For spacecraft and orbital stages passing through the protected region +/- 200 km and +/- 15 degrees of geostationary orbit, the probability of accidental collision with space objects larger than 10 cm in diameter shall not exceed 0.001 when integrated over 100 years from time of launch:*

COMPLIANT – Probability well below 0.001

***Assessment of Compliance of 4.5-2**, Limiting debris generated by collisions with small objects when operating in Earth 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 maneuver requirements does not exceed 0.01.:*

NOT APPLICABLE – There is not postmission disposal actions

7 ODAR Section 6: Assessment of Spacecraft Postmission Disposal Plans and Procedures.

7.1 Description of spacecraft disposal option selected.

The satellite will de-orbit naturally by atmospheric re-entry.

7.2 Identification of all systems or components required to accomplish any postmission disposal maneuvers. Plan for any spacecraft maneuvers required to accomplish postmission disposal.

No actions are required to accomplish postmission disposal.

7.3 Calculation of area-to-mass ratio after postmission disposal, if the controlled reentry option is not selected.

Mass = 8.793 Kg
Worst case scenario is with minimal CSA of 0.0079

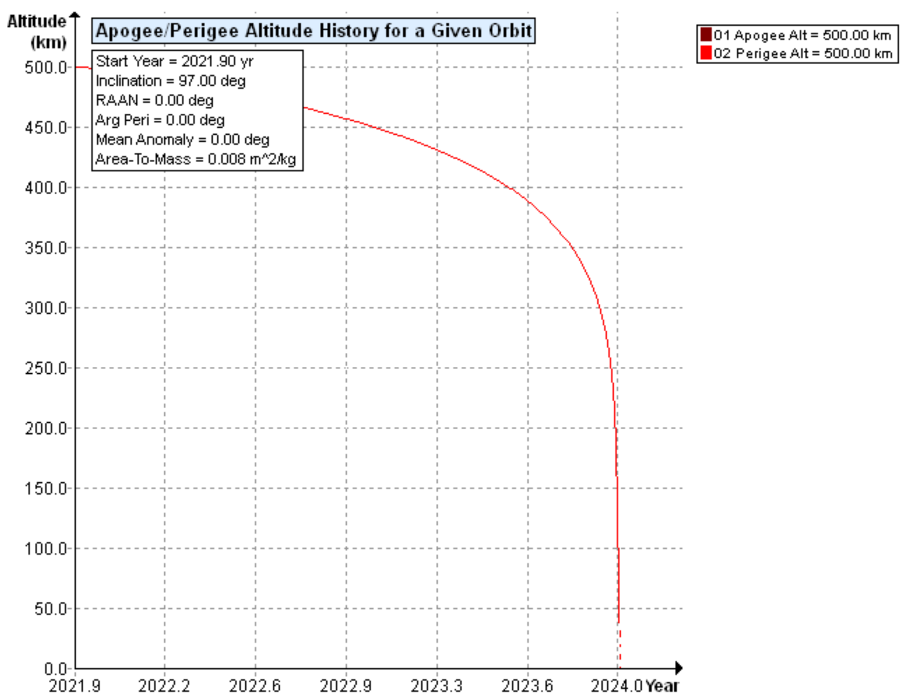

Figure 6. Satellite de-orbit naturally by atmospheric re-entry calculations.

The simulations consider a launch happening in December, 2021. The estimated orbital life time will be near 2.07 years.

7.4 If appropriate, preliminary plan for spacecraft controlled reentry.

Non applicable.

7.5 Assessment of spacecraft compliance with Requirements 4.6-1 through 4.6-4.

Assessment of Compliance of 4.6-1 Option (a)*, Leave the space structure in an orbit in which natural forces will lead to atmospheric reentry within 25 years after the completion of mission:*

COMPLIANT – The satellite will re-entry inside the 25 years period.

Assessment of Compliance of 4.6-1 Option (b)*, Storage orbit option: Maneuver the space structure into an orbit with perigee altitude above 2000 km and ensure its apogee altitude will be below 19,700 km, both for a minimum of 100 years:*

NOT APPLICABLE – Option (a) selected

Assessment of Compliance of 4.6-1 Option (c)*, Direct retrieval: Retrieve the space structure and remove it from orbit within 10 years after completion of mission:*

NOT APPLICABLE – Option (a) selected

Assessment of Compliance of 4.6-2*, Disposal for space structures near GEO:*

NOT APPLICABLE – NO GEO ORBIT

Assessment of Compliance of 4.6-3*, Disposal for space structures between LEO and GEO:*

NOT APPLICABLE – NO MEO ORBIT

Assessment of Compliance of 4.6-4*, Reliability of postmission disposal maneuver operations in Earth orbit: NASA space programs and projects shall ensure that all postmission disposal operations to meet Requirements 4.6-1, 4.6-2, and/or 4.6-3 are designed for a probability of success as follows:*

a. Be no less than 0.90 at EOM, and

b. For controlled reentry, the probability of success at the time of reentry burn must be sufficiently high so as not to cause a violation of Requirement 4.7-1 pertaining to limiting the risk of human casualty:

COMPLIANT – The probability will be over the 0.90 at EOM.

8 ODAR Section 7: Assessment of Spacecraft Reentry Hazards.

8.1 Detailed description of spacecraft components by size, mass, material, shape, and original location on the space vehicle, if the atmospheric reentry option is selected.

The following steps are suggested to be used to identify and evaluate a components potential reentry risk relative to the 4.7-1 requirement of having less than 15 J of kinetic energy and a 1:10,000 probability of a human casualty in the event the survive reentry.

a) Low melting temperature (less than 1000 °C) components are identified as materials that would never survive reentry and pose no risk to human casualty. This is confirmed through DAS analysis that showed materials with melting temperatures equal to or below that of copper (1080 °C) will always demise upon reentry for any size component up to the dimensions of a 1U CubeSat.
b) The remaining high temperature materials are shown to pose negligible risk to human casualty through a bounding DAS analysis of the highest temperature components, stainless steel (1500°C). If a component is of similar dimensions and has a melting temperature between 1000 °C and 1500°C, it can be expected to possess the same negligible risk as stainless-steel components.

Table1 . Highest Melting Point Materials Description

Component	Material	Melt Temperature (° C)[3]	Mass(g)
Antennas	Stainless Steel	1426.85	115
Screws	Stainless Steel	1426.85	2
Solar Cells	Gallium Arsenide (GaAs)	1236.85	1000

Based on these guidelines, there are no materials capable to survive the re-entry or will not have the dimensions to poses a risk under the parameters of the analysis conducted.

[3] NASA/TP-2016-218600-REV1. Debris Assessment Software User's Guide Version 2.1.

8.2 Assessment of spacecraft compliance with Requirement 4.7-1.

Assessment of Compliance of 4.7-1*, 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).

COMPLIANT – Well below the 1:10,000 risk

9 ODAR Section 7A: Assessment of Spacecraft Hazardous Materials.

9.1 Summary of the hazardous materials contained on the spacecraft.

There are no hazardous materials on the satellite.

10 ODAR Section 8: Assessment for Tether Missions.

10.1 Type of tether; e.g., momentum or electrodynamics.

None . There are no tethers used on the SAI-2 mission.

10.2 Description of tether system, including at a minimum (1) tether length, diameter, materials, and design (single strand, ribbon, multi-strand mesh), and (2) end-mass size and mass.

Non applicable.

10.3 Determination of minimum size of object that could sever the tether.

Non applicable.

10.4 Tether mission plan, including duration and postmission disposal.

Non applicable.

10.5 Probability of tether colliding with large space objects.

Non applicable.

10.6 Probability of tether being severed during mission or after postmission disposal.

Non applicable.

10.7 Maximum orbital lifetime of a severed tether fragment.

Non applicable.

10.8 Assessment of compliance with Requirement 4.8-1.

***Assessment of Compliance of 4.8-1**, Tethers:*

NOT APPLICABLE – NO Tethers mission

11 DEBRIS ASSESSMENT SOFTWARE LOG

05 14 2021; 16:24:47PM Activity Log Started
05 14 2021; 16:28:29PM Processing Requirement 4.5-1: Return Status : Passed
==============
Run Data
==============

INPUT

Space Structure Name = SAI-2
Space Structure Type = Payload
Perigee Altitude = 500.000000 (km)
Apogee Altitude = 500.000000 (km)
Inclination = 97.000000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Mean Anomaly = 0.000000 (deg)
Final Area-To-Mass Ratio = 0.067000 (m^2/kg)
Start Year = 2021.000000 (yr)
Initial Mass = 8.793000 (kg)
Final Mass = 8.793000 (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 = 0.000001
Returned Error Message: Normal Processing
Date Range Error Message: Normal Date Range
Status = Pass

============== End of Requirement 4.5-1 ==============
05 14 2021; 16:28:45PM Requirement 4.5-2: Compliant

05 14 2021; 16:28:46PM Processing Requirement 4.6 Return Status : Passed

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

INPUT

Space Structure Name = SAI-2
Space Structure Type = Payload
Perigee Altitude = 500.000000 (km)
Apogee Altitude = 500.000000 (km)
Inclination = 97.000000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Mean Anomaly = 0.000000 (deg)
Area-To-Mass Ratio = 0.067000 (m^2/kg)
Start Year = 2021.000000 (yr)
Initial Mass = 8.793000 (kg)
Final Mass = 8.793000 (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

Suggested Perigee Altitude = 500.000000 (km)
Suggested Apogee Altitude = 500.000000 (km)
Returned Error Message = Reentry during mission (no PMD req.).

Released Year = 2021 (yr)
Requirement = 61
Compliance Status = Pass

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

05 14 2021; 16:28:56PM *********Processing Requirement 4.7-1
Return Status : Passed
***********INPUT****
Item Number = 1
name = SAI-2
quantity = 1
parent = 0
materialID = 8
type = Box
Aero Mass = 8.793000
Thermal Mass = 8.793000
Diameter/Width = 0.200000
Length = 0.300000
Height = 0.100000

name = S
quantity = 1
parent = 1
materialID = 8
type = Box
Aero Mass = 8.793000
Thermal Mass = 8.793000
Diameter/Width = 0.200000
Length = 0.300000
Height = 0.100000

*************OUTPUT****
Item Number = 1
name = SAI-2
Demise Altitude = 77.993660
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = S
Demise Altitude = 57.698299
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

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

05 14 2021; 16:20:18PM Science and Engineering - Apogee/Perigee History for a Given Orbit

INPUT

Perigee Altitude = 500.000000 (km)
Apogee Altitude = 500.000000 (km)
Inclination = 97.000000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Mean Anomaly = 0.000000 (deg)
Area-To-Mass Ratio = 0.007900 (m^2/kg)
Start Year = 2021.900000 (yr)
Integration Time = 5.000000 (yr)

OUTPUT

Plot
05 14 2021; 16:21:44PM Science and Engineering - Orbit Lifetime/Dwell Time

INPUT

Start Year = 2021.900000 (yr)
Perigee Altitude = 500.000000 (km)
Apogee Altitude = 500.000000 (km)
Inclination = 97.000000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Area-To-Mass Ratio = 0.007900 (m^2/kg)

OUTPUT

Orbital Lifetime from Startyr = 2.069815 (yr)
Time Spent in LEO during Lifetime = 2.069815 (yr)
Last year of Propagation = 2023 (yr)
Returned Error Message: Object reentered

12 REFERENCES.

[1] National Aeronautics and Space Administration. NASA. NASA TECHNICAL STANDARD NASA-STD-8719.14B Approved: 2019-04-25.

[2] National Aeronautics and Space Administration. NASA. NPR 8715.6B, NASA Procedural Requirements for Limiting Orbital Debris and Evaluating the Meteoroid and Orbital Debris Environments.

[3] Astro Digital US, Inc. Astro Digital Ignis Orbital Debris Assessment Report (ODAR) ASTRO-DIGITAL-IGNIS-ODAR-1.0. 4/19/2019.

[4] Justin Treptow, NASA/KSC/VA-G2. Orbital Debris Assessment for The CubeSats on the ELaNa-XIX Mission per NASA-STD 8719.14ª (FCC) Rev C. ELVL-2017-0044671. February 27, 2018.

[5] Whitmore, Stephen, A, et al. "Launch and Deployment of the High-Latitude Dynamic E-Field (HiDEF) Explorer Satellite Constellation". SSC08-IV-6. 22nd Annual AIAA/USU Conference on Small Satellites.

[6] The Australian Space Weather Agency. IPS Radio and Space Services. Satellite Orbital Decay Calculations.

[7] NASA/TP-2016-218600-REV1. Debris Assessment Software User's Guide Version 2.1. Orbital Debris Program Office. NASA Johnson Space Center.