Preliminary Orbital Debris Assessment Report for Myriota Satellites IAW NASA-STD 8719.14A

REFERENCES:

A. Process for Limiting Orbital Debris, NASA-STD-8719.14A, 25 May 2012
B. HQ OSMA Policy Memo/Email to 8719.14: CubeSat Battery Non-Passivation, Suzanne Aleman to Justin Treptow, 10, March 2014

The following table summarizes the compliance status of the Myriota satellites with the NASA requirements for limiting orbital debris generation (ref A). The first 3 satellites with launches planned for late 2019 will utilize the 3U CubeSat form factor (34 cm x 10 cm x 10 cm). The remainder of the constellation may utilize a larger size up to 6U (24 cm x 36 cm x 10 cm), and all satellites will be designed to be fully compliant with all applicable requirements. A mass of 7 kg is the upper limit on the 3U design and hence will be assumed to be the worst-case ballistic coefficient. If the design size increases to 6U configuration this will be designed to be compliant with all regulations for limiting orbital debris.

Table 1: Orbital Debris Requirement Compliance Matrix

Requirement	**Compliance**	**Comments**
4.3-1a Debris Passing Through LEO: 25-Year Maximum Lifetime	Not applicable	No Planned Debris Release
4.3-1b Debris Passing Through LEO: Total Object-Time Product	Not applicable	No Planned Debris Release
4.3-2 Debris Passing Near Geosynchronous Altitude	Not applicable	No Planned Debris Release and apogee significantly lower than GEO.
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	Compliant	Onboard energy source (batteries) incapable of debris-producing failure
4.4-2 Design for passivation after completion of mission operations while in orbit about Earth or the Moon:	Compliant	Onboard energy source (batteries) incapable of debris-producing failure
4.4-3 Limiting the long-term risk to other space systems from planned breakups:	Not applicable	No Planned breakups
4.4-4 Limiting the short-term risk to other space systems from planned breakups	Not applicable	No Planned breakups
4.5-1 Limiting debris generated by collisions with large objects when operating in Earth orbit	Compliant	Probability of 0.00000

Requirement	Compliance	Comments
4.5-2 Limiting debris generated by collisions with small objects when operating in Earth or lunar orbit	Not applicable	No capability or plan for end-of-mission disposal
4.6-1(a) Disposal for space structures in or passing through LEO: Atmospheric reentry option	Compliant	The maximum perigee of 600 km and apogee of 600 km results in 24.7 years, which is within 25-year requirement.
4.6-1(b) Disposal for space structures in or passing through LEO: Storage orbit	Not applicable	
4.6-1(c) Disposal for space structures in or passing through LEO: Direct retrieval	Not applicable	
4.6-2 Disposal for space structures near GEO	Not applicable	
4.6-3 Disposal for space structures between LEO and GEO	Not applicable	
4.6-4 Reliability of post-mission disposal operations in Earth orbit	Not applicable	Passive atmospheric reentry disposal is planned, worst case dead on arrival analysis.
4.7-1 Limit the risk of human casualty	Compliant	Non-credible risk of human casualty, no components will survive reentry.
4.8-1 Mitigate the collision hazards of space tethers in Earth or Lunar orbits	Not applicable	No planned use of tethers on any satellites

Section 1: Program Management and Mission Overview

Containerised CubeSats will be deployed in groups at each orbital plane as required. The 2019 planned launches will use rideshare launches. Myriota may use dedicated launches for the activities to follow as the constellation is built out.

Tyvak Nano-satellites has been selected as the supplier of satellites for Myriota. Replenishment strategies will be implemented to sustain the constellation once it is fully deployed to account for decommissioning of satellites.

Section 2: Spacecraft Description

Overview

The mission utilizes Tyvak Nano-satellite's Mark 2 avionics suite architecture for power generation and management, telemetry and commanding (TC), Command and Data Handling (CDH), thermal management, and Guidance Navigation and control (GNC). The Tyvak system utilizes lithium-ion battery modules and deployable solar arrays for power generation and management. The TC consists of S-Band transceivers for bus telemetry downlinks and command uplinks; as well as an X-Band transmitter for payload data downlink. The CDH and GNC are packaged into the Inertial Reference Module (IRM) which consists of a CDH & GNC processor, an IMU, three magnetorquers, three reaction wheel assemblies, and two star trackers. The software-defined radio payload, developed by a third party and having heritage on a range of missions, will be used to enable the Myriota direct-to-orbit communication system.

CONOPS

Spacecraft commissioning will occur autonomously, and commercial service will be provided immediately after launch. The spacecraft has been designed to self-resolve from anomalies requiring minimal operator intervention.

The satellites will primarily operate in a reverse link mode. Customer data will be transmitted from terminals on the ground, received by satellite and the customer data will be transmitted to the ground in a store and forward approach. The forward link mode allows for data to be transmitted to terminals in the field.

An inhibit scheme actuated by deployer separations switches shall be employed to ensure that the satellite can only begin transmission after a delay following deployment. The delay will last 30-45 minutes depending on the launch vehicle requirements. Deployment of all solar panels and antennas will happen autonomously after an independent delay, approximately 90 minutes after deployment from the deployer.

Nominal operations will switch the attitude state of the satellite between ground station pointing and sun tracking. The maximum gain of the antenna payload antenna will be occasionally pointed at the ground.

Materials

The primary structure for the satellite is composed of Aluminum 7075-T7. The spacecraft is largely composed of components manufactured by Tyvak, which consist of electrical components, PCBs or FR4, and solar cells. Both the S-Band and X-Band transmit antenna and the GPS receive antenna use patches and are made from ceramic. 77g of Steel (A-286) fasteners with high melting temperature will be used on the satellite.

Hazards

There are no pressure vessels, hazardous or exotic materials.

Batteries

The electrical power storage system consists of common lithium-ion batteries with overcharge/current protection circuitry. The lithium batteries used are LG 18650 and have passed IEC/UN38.3. The theoretical peak energy storage is 83 Whrs, though the peak charge is deliberately limited to approximately 75 Whrs via circuit protection.

Section 3: Assessment of Spacecraft Debris Released during Normal Operations

The assessment of spacecraft debris requires the identification of an object (>1 mm) expected to be released from the spacecraft any time after launch, including object dimensions, mass, and material. The section 3 requires rationale/necessity for release of each object, time of release of each object, relative to launch time, release velocity of each object with respect to spacecraft expected orbital parameters (apogee, perigee, and inclination) of each object after release, the calculated orbital lifetime of each object, including time spent in Low Earth Orbit (LEO), and an assessment of spacecraft compliance with Requirements 4.3-1 and 4.3-2.

No releases are planned for the Myriota satellites; therefore, this section is not applicable.

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

There are NO plans for designed spacecraft breakups, explosions, or intentional collisions for the Myriota Satellites.

The probability of battery explosion is very low, and, due to the very small mass of the satellites and their short orbital lifetimes, the effect of an explosion on the far-term LEO environment is negligible (ref (B)).

The CubeSats batteries still meet Req. 56450 (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." (ref. (B)).

Limitations in space and mass prevent the inclusion of the necessary resources to disconnect the battery or the solar arrays at EOM. However, the low charges and small battery cells on the satellite’s power system prevent a catastrophic failure, so that passivation at EOM is not necessary to prevent an explosion or deflagration large enough to release orbital debris.

The satellites satisfy Requirements 4.4-1 and 4.4-2 by the batteries being equipped with protection circuitry.

Section 5: Limiting debris generated by collisions

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

Two deployment states shall be considered for the analysis of atmospheric reentry disposal. The equation for the mean cross-sectional area for complex shapes is most appropriate.

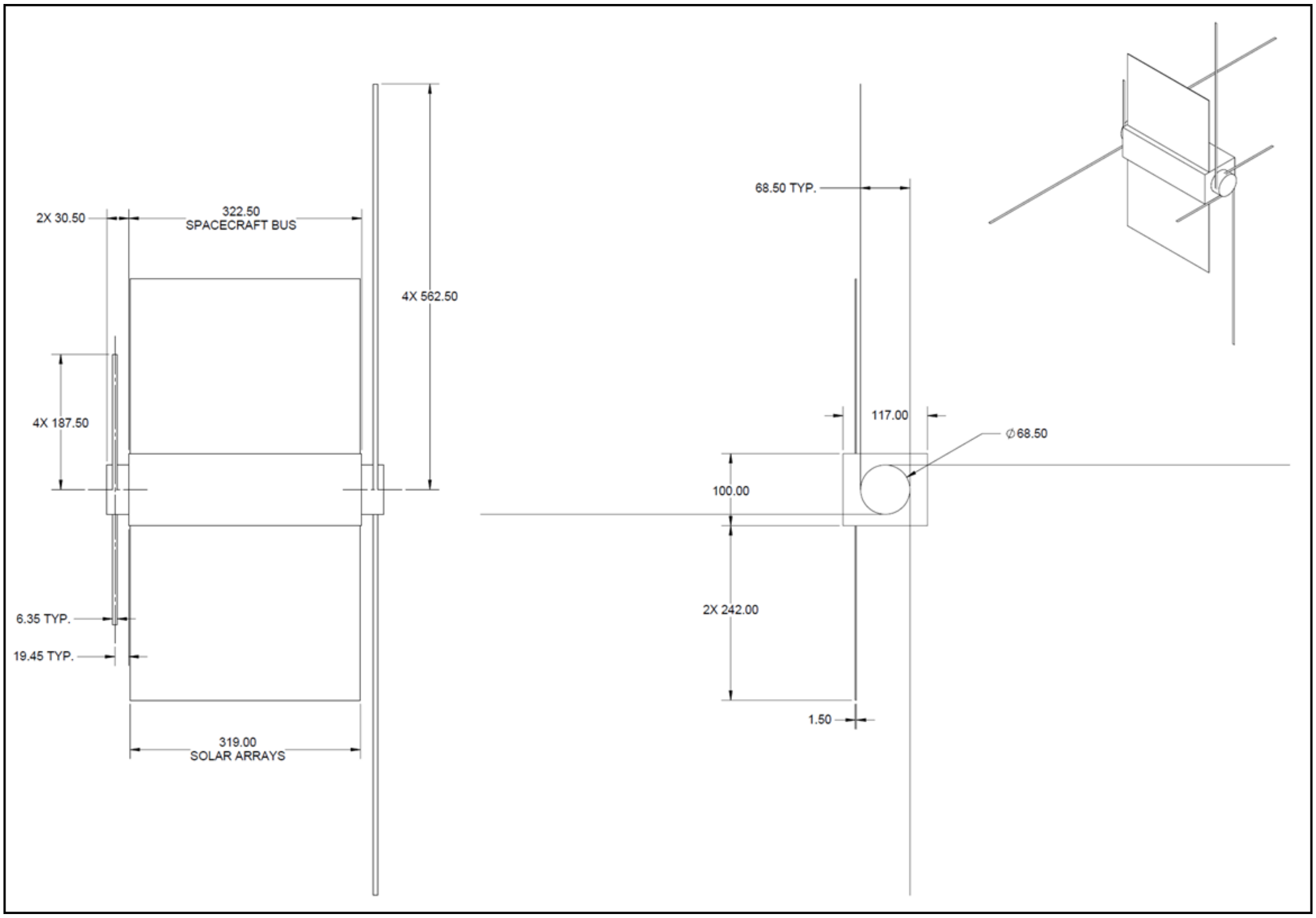

Figure 1: View of 3U satellites in deployed state

Equation 1: $$Mean\ CSA\ Stowed = \frac{\Sigma Surface\ Area}{4} = \frac{[2*(w*l)+2*(w*h)2*(w*h)]}{4} = 0.04084m^2$$

Equation 2: $$Mean\ CSA\ Deployed\ = \frac{(A_1\ +\ A_2\ +\ A_3 +\ A_{Solar\ Array})}{2} = 0.13174m^2$$

All satellites will utilize a containerized CubeSat form factor. They are stowed in a convex configuration prior to deployment into space, indicating there are no elements of the satellite obscuring another element of the same satellite from view. Thus, mean CSA for all stowed satellites was calculated using Equation 1 as this configuration renders the longest orbital lifetimes for all satellites and is hence the worst case. The upper design mass for each satellite is 7kg, resulting in an Area to mass ratio of 0.0058 m^2/kg

An apogee of 600 km and perigee at an upper limit of 600 km were used, as this is the maximum altitude of any launch option presently being considered for these satellites.

DAS yields 24.7 year orbit lifetime for the stowed state, which in turn is used to obtain the collision probability. The probability of each satellite configuration has a probability of collision of 0.0. Calculation of spacecraft probability of collision with space objects larger than 10 cm in diameter during the orbital lifetime of the spacecraft takes into account both the mean cross-sectional area and orbital lifetime. Solar Flux table dated January 2nd, 2019 was used for this analysis.

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.

Table 2: Satellite orbital lifetime and collision probability

Property \ Configuration	Stowed	Deployed
Mass (kg)	7	7
Mean CSA (m2)	0.040841	0.131742
Area-to-Mass (m2/kg)	0.005834	0.018820
Orbital Lifetime* (yrs)	24.7	5.4
Probability of Collision*	0	0

* Solar Flux Table Dated January 2nd, 2019

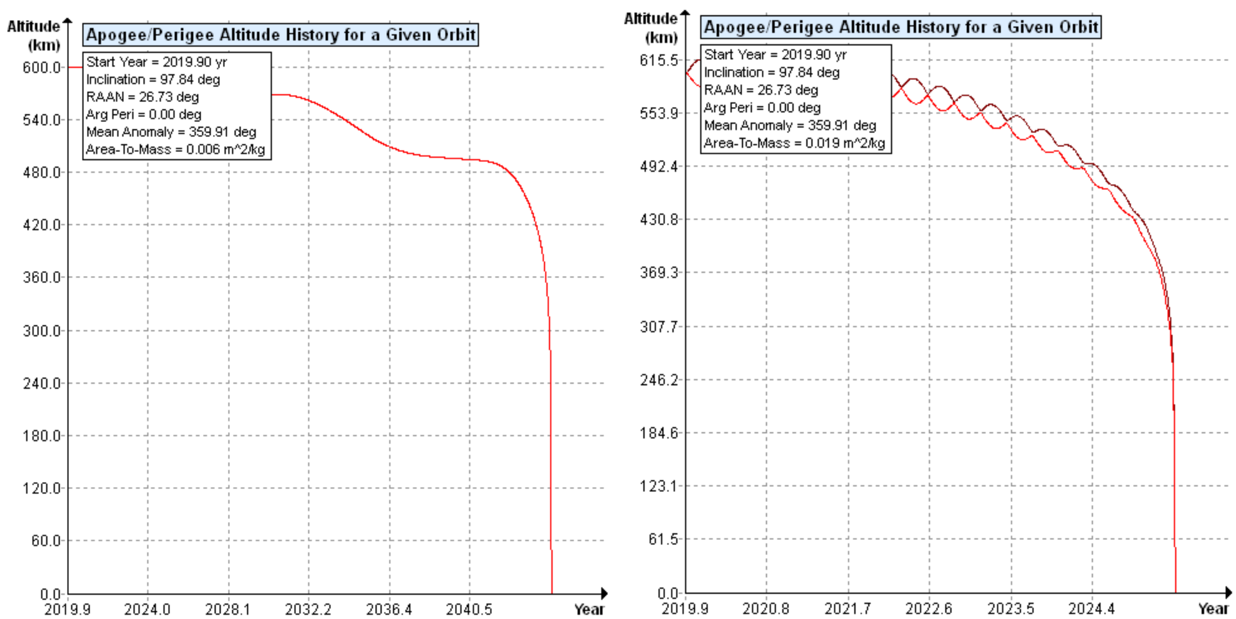

Figure 2: Atmospheric Demise from 600km for Stowed (left) and Deployed (right) satellites

The probability of any Myriota satellite collision with debris and meteoroids greater than 10cm in diameter and capable of preventing post-mission disposal is less than 0.00000 for any configuration. This satisfies the 0.001 maximum probability requirement 4.5-1.

Since the satellites have no capability or plan for end-of-mission disposal, requirement 4.5-2 is not applicable.

Section 6: Assessment of Spacecraft Post Mission Disposal Plans and Procedures

It is planned that the Myriota satellites will naturally decay from orbit within 25 years after the end of the mission. This applied to a worst case assumption that a satellite was not capable of deploying its solar arrays at the start of the mission, satisfying requirement 4.6-1(a) detailing the spacecraft disposal option. Planning for spacecraft maneuvers to accomplish post-mission disposal is not applicable. Disposal is achieved via passive atmospheric reentry.

Calculation of the worst-case (smallest Area-to-Mass) post-mission disposal finds the stowed configuration as the worst case.

The assessment of the spacecraft illustrates they are compliant with Requirements 4.6-1 through 4.6-4.

DAS 2.1.1 Orbital Lifetime Calculations:
DAS inputs are: 600 km maximum apogee 600km maximum perigee altitudes with an inclination of 97.77° at deployment no earlier than 2019.9. An area to mass ratio of 0.005834 m^2/kg for satellites was used. DAS 2.1.1 yields a 24.7 years orbit lifetime for the stowed case.

Assessment results show compliance.

Section 7: Assessment of Spacecraft Reentry Hazards

Material selection of components for the satellites will be influenced to ensure all requirements are satisfied. The assessment used DAS 2.1.1, a conservative tool used by the NASA Orbital Debris Office to verify Requirement 4.7-1. The analysis is intended to provide a bounding analysis for characterizing the survivability of a satellite component during re-entry. For example, when DAS shows a component surviving reentry it is not considering the material ablating away or charring due to oxidative heating. Both physical effects are experienced upon reentry and will decrease the mass and size of the real-life components as they reenter the atmosphere, reducing the risk they pose still further.

The following steps are used to identify and evaluate a component's 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 component survives reentry.

1. 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 re-entry for any size component up to the dimensions of a 1U CubeSat.
2. 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.

Table 3: Myriota Satellite High Melting Temperature Materials

Component Name	Material	Mass (kg)	Demise Alt (km)	Kinetic Energy (J)
Fasteners	Steel (A-286)	0.077	67.8	0

The DAS analysis predicts that a majority of stainless-steel components demise upon reentry. If a component survives to the ground but has less than 15 Joules of kinetic energy it is not included in the Debris Casualty Area that inputs into the Probability of Human Casualty calculation. This is why the Myriota Satellites have a 1:0 probability of human casualty, as none of the few components that may survive reentry have more than 15J of energy.

All Myriota Satellites will be in compliance with Requirement 4.7-1 of NASA-STD-8719.14A.

Section 8: Assessment for Tether Missions

No Myriota Satellites will be deploying any tethers, so Requirement 4.8-1 is inapplicable.

If you have any questions, please contact Myriota.

Attachment - CSA calculations

Stowed	
w (m)	0.1
l (m)	0.117
h (m)	0.3225
Surface Area4=[2*(w*l)+2*(l*h)+2*(w*h)]/4	0.040841
mass (kg)	7
Area-to-Mass (m2/kg)	0.005834

Deployed	
A1	0.0364285
A2	0.041911
A3	0.0117
Asolar	0.154396
Aant	0.01905
(A1 + A2 + A3+ ASol + Aant)/2	0.13174275
mass (kg)	7
Area-to-Mass (m2/kg)	0.01882039286