HORIS-3 AND HORIS-4 Orbital Debris Assessment Report

(ODAR) per NASA-STD 8719.14C

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Revision A

Date July 17, 2026

ACME AtronOmatic LLC d/b/a MyRadar

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Required Signatures

Title: CEO Andrew Green	
Date 7/17/26	

Title: Chief Scientist & CTO Sarvesh Garimella	
Date 7/17/26	

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Date 7/17/26	

Version History

Version	Date	Change Author	Change Summary
Rev 0	7/17/26	Douglas Franz	Initial Release
Rev 1			

1. Self Assessment

A self-assessment against NASA-STD-8719.14C spacecraft mission requirements is shown below for HORIS-3 and HORIS-4.

Requirement	Resolution	Remarks
4.3-1.a MRD 25-year limit	N/A	No separating objects
4.3-1.b MRD<100 object x year limit	N/A	No separating objects
4.3-2 GEO MRD	N/A	No separating objects
4.4-1 <0.001 Explosion Risk	COMPLIANT	
4.4-2 Passivate Energy Sources	COMPLIANT	
4.4-3 Limit Intentional BU, Long Term	N/A	No planned breakups
4.4-4 Limit Intentional BU, Short Term	N/A	No planned breakups
4.5-1 <0.001 10cm Impact Risk	COMPLIANT	

Requirement	Resolution	Remarks

4.5-2 <0.01 Small MMOD Impacts	N/A	No planned controlled reentry, no critical subsystems to protect
4.6-1a-c LEO Disposal	COMPLIANT	
4.6-2 Storage or Earth-escape	N/A	
4.6-3 Long-term Reentry	N/A	
4.6-4 Disposal Reliability	COMPLIANT	
4.7-1 Reentry Risk	COMPLIANT	
4.8-1 Special Classes	N/A	Not considered a special class of space mission

Table 1-1: Self-Assessment of HORIS-3 AND HORIS-4

ESA DRAMA version 4.1.4 was used for evaluation.

2. ODAR Section 1: Program Management and Mission Overview

a. Identification of the Company and Program Executive

Company: ACME AtronOmatic LLC d/b/a MyRadar (“MyRadar”)

Program Executive: Andy Green CEO

Address: 111 W Jefferson Street #200, Orlando, FL 32801

Complete as many of the following as needed, or substitute other

titles **Program Manager**: Douglas Franz

Chief Scientist: Sarvesh Garimella

b. Identification of any foreign government or space agency participation in the mission and a summary of NASA's responsibility under the governing agreement(s)

None

c. Brief description of the mission

HORIS-3 and HORIS-4 are two identical spacecraft, launched together and deployed into the same orbit at the same time. This ODAR addresses the behavior of either of these spacecraft.

The mission is an orbital demonstration of multi-sensor data capture and downlink, AI assisted low-bandwidth alerting, and training dataset generation. The primary mission objective is collection of orbital training data for AI model development.

d. Identification of the anticipated launch vehicle and launch site

The spacecraft will be launched aboard SpaceX Transporter 22, from Vandenberg Air Force Base.

e. Identification of the proposed launch date and mission duration

The spacecraft will be launched NET October 1, 2027, and the mission duration will be 2 years.

f. Description of the launch and deployment profile, including all parking, transfer, and operational orbits with apogee, perigee, and inclination

The spacecraft will be inserted into a circular Sun synch orbit at 510 km altitude, on an inclination from the equator of 97.4 degrees.

g. Description of the spacecraft's maneuver capability, including both attitude and orbit control. Give the time period during which the capabilities will be exercised.

Attitude control is accomplished using the Guidance, Navigation and Control (GNC) Subsystem. This includes an L1 GPS receiver, triaxial magnetometer, triaxial Control Moment Gyroscope, and a star tracker for attitude determination and guidance/navigation; a real-time clock (RTC) to maintain relative time knowledge; and three orthogonal magnetorquers.

The spacecraft has no maneuver capability. There is no propulsion.

h. Reason for selection of operational orbit(s) (such as ground track, SSO, GEO sync, instrument resolution, con-loncate with other spacecraft, …)

The orbit was selected for launch availability, and it is SSO.

There is no planned co-location with any other spacecraft.

i. Identification of any interaction nor potential physical interference with other operational spacecraft (Note: This does not include potential for RF interaction unless it affects the risk of generating orbital debris.)

There is no expected interference with other operational spacecraft.

3. ODAR Section 2: Spacecraft Description

a. Physical description of the spacecraft, including spacecraft bus, payload instrumentation, and all appendages, such as solar arrays, antennas, and instrument or attitude control binoculars

Each spacecraft is a single unit (1.5U) CubeSat with dimensions 15x10x10cm. The total mass is 2.949 kg. It contains the following systems:

Guidance, Navigation and Control (GNC) Subsystem:

GNC utilizes an L1 GPS receiver, triaxial magnetometer, triaxial Control Moment Gyroscope and a star tracker for attitude determination and guidance/navigation; and a real-time clock (RTC) to maintain relative time knowledge.

For attitude control, the spacecraft has three orthogonal magnetorquers.

Command and Data Handling (C&DH) Subsystem:

The C&DH consists of two systems, (1) a low-risk, radiation hardened system supervisor and (2) a high-level, risk-tolerant processor. The system supervisor manages spacecraft health, housekeeping, GNC, and critical data communications. The high level processor is used for S- and X-Band communications and more intensive tasks such as image processing.

Communications Subsystem (COMMS):

The spacecraft uses two radios, an Iridium transceiver and a LimeSDR transceiver.

The Iridium transceiver is used for regular spacecraft housekeeping, early notifications of potential event detections and basic spacecraft commanding.

The LimeSDR transceiver operates on S-Band for spacecraft receive (uplink), primarily used for software/firmware updates; and X-Band (downlink) for bulk/raw data transfer. Both bands have three operational data rates (safe, nominal and fast).

Commercial satellite data services for S- and X-Band are utilized, with ground stations located in Svalbard, Norway and Troll, Antarctica.

Electrical Power Subsystem (EPS)

The EPS is a direct energy transfer system using a solar array producing approximately 7.5W of orbital average power to charge the 6.6 A-hr battery system. The battery consists of two COTS Li-ion cells and the solar array utilizes COTS space-rated photovoltaic cells.

Thermal Control Subsystem (TCS):

The TCS controls hardware temperature through cold biasing of the thermal design, utilizing passive radiators for cooling and active heaters to stabilize temperatures. Sensors are wired to the system supervisor, which hosts thermal control algorithms to control the heaters. Further cooling capability is provided by active coolers, when needed.

Structure Subsystem:

The primary structure is COTS, space-rated, CubeSat chassis, measuring 100x100x150mm, and is fabricated from Aluminum 6061-T651 with a hard anodized finish. The structure includes two separation springs and kill switches required for dispenser integration.

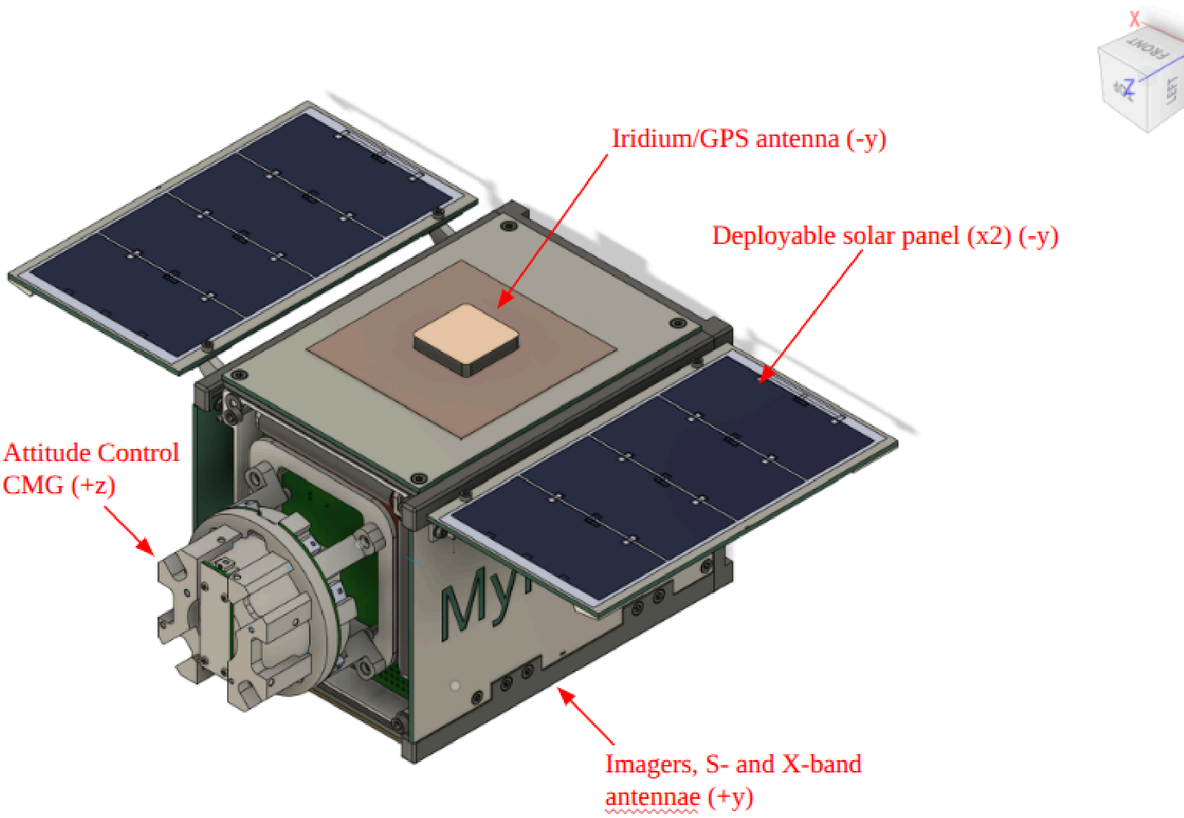
Propulsion Subsystem:

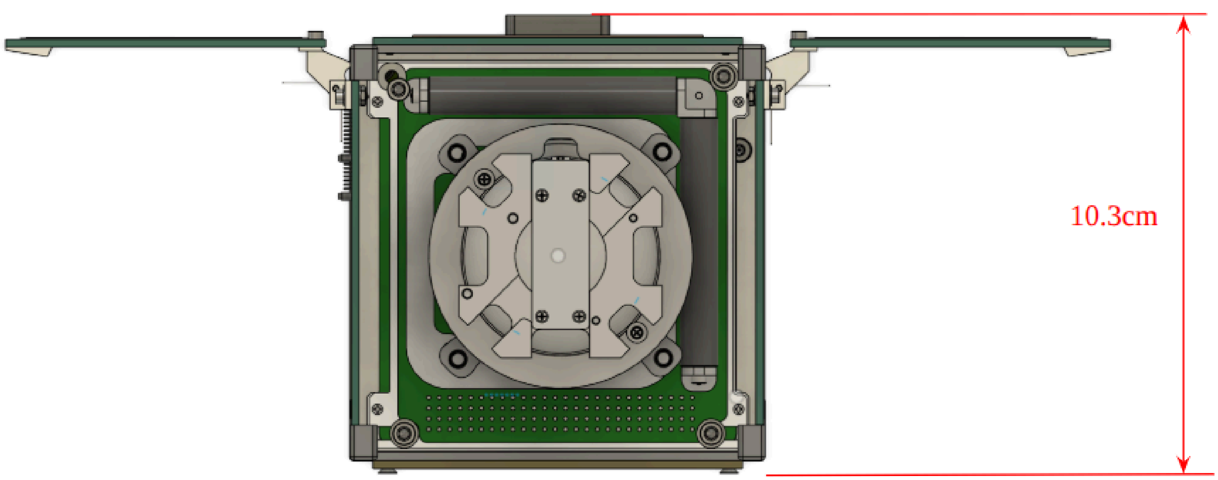
There are no propulsive systems.

Payload Subsystem:

The payload consists of four, fixed focal-length imagers, nominally pointed Nadir. One imager is used for initial alert detection while the others are used to capture the scene.

b. Detailed illustration of the entire spacecraft in the mission operation configuration with clear overall dimensional markings and marked internal component locations


Iridium/GPS antenna (-y)
Deployable solar panel (x2) (-y)
Attitude Control
CMG (+z)
Imagers, S- and X-band
antennae (+y)


10.3cm

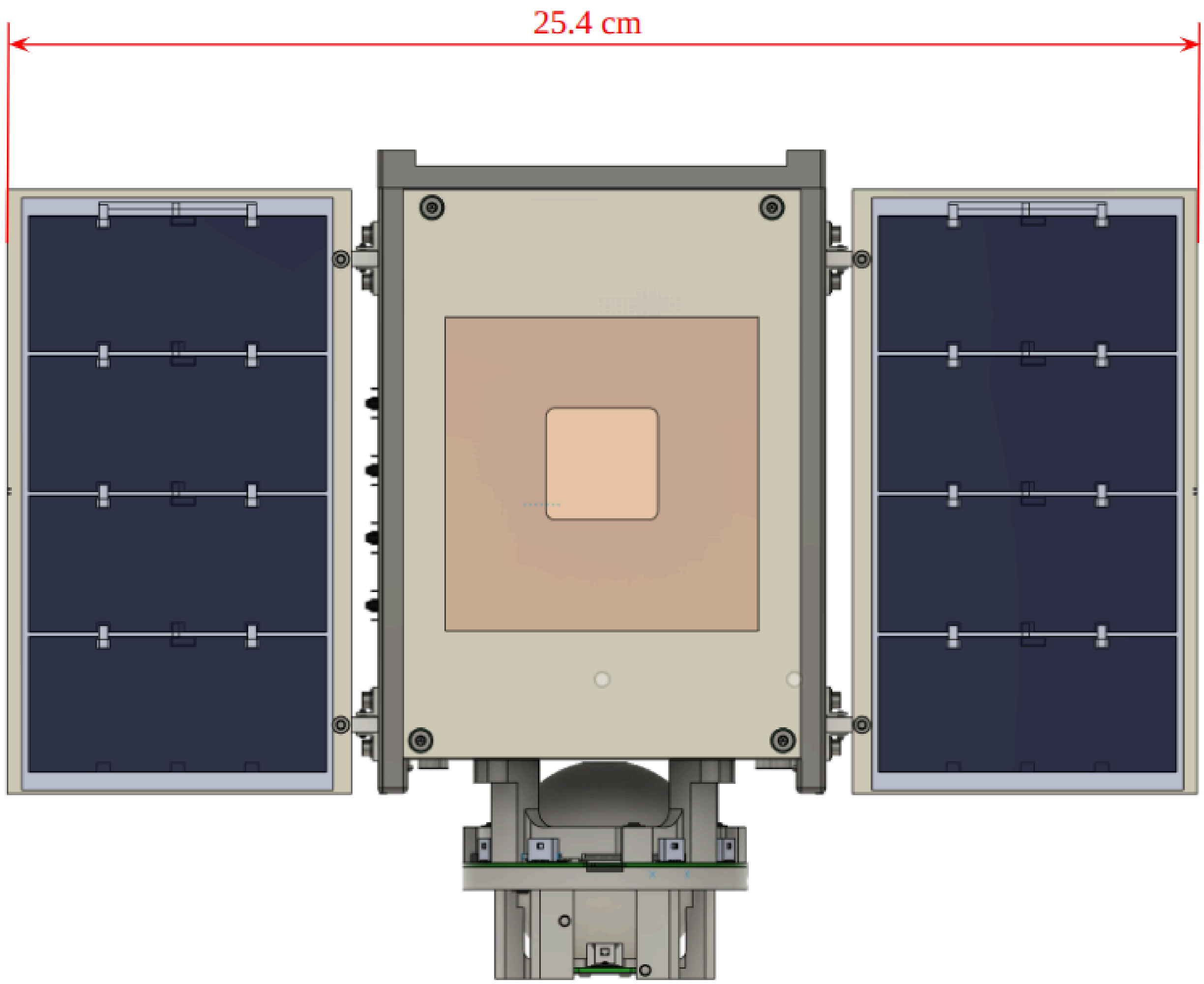


Figure 3-1: Spacecraft with Overall Dimensions

c. Total spacecraft mass at launch, including all propellants and fluids

The total mass is 2.949 kg.

d. Description of all propulsion systems (cold gas, monnon-pronpellant, bi propellant, electric, nuclear)

N/A There is no propulsion system.

e. 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. Description of all fluid systems, including size, type, and qualifications of fluid containers such as propellant and pressurization tanks, including pressurized batteries

N/A No fluids will be on board.

f. 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

Attitude control utilizes an L1 GPS receiver, triaxial magnetometer, triaxial Control Moment Gyroscope and a star tracker for attitude determination and guidance/navigation; and a real-time clock (RTC) to maintain relative time knowledge.

The spacecraft has three orthogonal magnetorquers. As shown in Figure 3-1, the normal attitude with respect to the velocity vector is with the solar panels facing zenith, and the radio antennae and imagers facing nadir.

g. Description of any range safety or other pyrotechnic devices

There are no range safety or pyrotechnic devices on the spacecraft.

h. Description of the electrical generation and storage system

The Electric Power Subsystem is a direct energy transfer system using a solar array producing approximately 7.5W of orbital average power to charge the 6.6 A-hr battery system. The battery consists of two COTS Li-ion cells and the solar array utilizes COTS space-rated photovoltaic cells. .

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

There are no other sources of energy storage on the spacecraft

j. Identification of any radioactive materials on board or make a positive statement that there are non radioactive materials onboard

There are no radioactive materials on spacecraft.

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

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

There are no separating objects from the spacecraft during any phase of the HORIS-3 and HORIS-4 mission.

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

4.3-1a/b, Mission Related Debris Passing Through LEO: **N/A**

There are no plans to separate objects from the spacecraft.

4.3-2, Mission Related Debris Passing Near GEO: **N/A**

There are no plans to separate objects from the spacecraft.

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

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

During nominal deployment and mission operations, there are no credible scenarios that result in spacecraft breakup.

b. Summary of failure modes and effects analyses of all credible failure modes which may lead to an accidental explosion

The spacecraft contains Lithium-ion batteries as the only source of stored energy. Protections are in place to prevent on-orbit explosions.

Lithium-Ion Batteries

Unpressurized UL-Listed 18650 lithium-ion cells arranged into battery packs are used. Battery packs also feature mitigations against elevated temperatures, overvoltage protection and current interrupt devices. These mitigations prevent any catastrophic failure mode and potential for on-orbit explosion to occur.

The cells undergo UN38.3 safety testing to further ensure battery integrity, including charge and discharge testing, internal resistance characterization, and cycle performance. Low-performing or non-conformant cells are rejected, and cells with similar performance are grouped together into battery packs.

Cell Safety Devices

18650 cells include a set of standard safety features described below. These features are included in all 18650-format cells and provide mitigation against elevated temperatures and current.

- Positive Temperature Coefficient Device (PTC): The resistance of the PTC is low during normal operation and increases when the temperature rises above a critical level to reduce current flow.
- Current Interrupt Device (CID): a fuse-type device that cuts off the electrical circuit permanently when triggered by excessive cell pressure. This is enacted by releasing the connection of the positive terminal.

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

The spacecraft is not designed for breakup and the mission does not include any planned explosions or collisions.

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

Batteries will not be passivated.

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

Per NASA's Process for Limiting Orbital Debris Objective 2-2, complete disconnection (hard passivation) is not required as energy generation and storage will be controlled (soft passivation) to a level which will not cause an explosion or deflagration large enough to release orbital debris or break up the spacecraft.

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

Requirement 4.4-1: Limiting probability of accidental explosions: **COMPLIANT**

"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)."

The only stored energy is in the two battery cells, and there is no credible scenario in which energy release in the batteries causes an explosion.

Therefore, the expected probability of accidental explosions is under 0.001 and is Compliant to this requirement.

Requirement 4.4-2: Passivate to limit probability of accidental explosions: **COMPLIANT**
Requirement 4.4-3: Limit the long-term risk to other space systems from planned breakups: **N/A**

There are no planned breakups.

Requirement 4.4-4: Limit the long-term risk to other space systems from planned breakups: **N/A**

There are no planned breakups.

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

a. 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, including orbital debris and meteoroids, of sufficient size to prevent post mission disposal

The probability of accidental collision with space objects larger than 10 cm in diameter was calculated for two parts of the orbit life: for 2 years during the mission; and the following 2.93 years, during post mission disposal.

During the mission, the nominal orientation of the solar panels is facing zenith, and the imagers and antennae are facing nadir.

Effective cross sectional area is calculated based on maintaining one face of the 10 x 10 x 15 cm satellite oriented in the RAM direction. This yields the minimum cross sectional area of 0.01 m^2.

Post mission, random tumbling is assumed. See Section 7 for calculation of the effective cross sectional area.

The mass of 2.949 kg remains constant throughout the orbit lifetime.

Orbit Life Phase	**Area/Mass Ratio, m2/kg**	**Initial Altitude, km**	**Final altitude, km**	**Duration, years**	**Large Object Collision Risk**
Mission	0.0116	510	405	2	1.9276E-07
Post Mission	0.0116	405	Demise	2.93	1.9201E-07
				Total	**3.8477E-07**

Table 6-1 Lifetime, and Large Object Collision Risk

Table 6-1 shows the total probability of Large Object collision to be 3.8477E-07, which is less than the required 0.001 probability.

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

Requirement 4.5-1: Limit debris generated by collisions with large objects when operating in Earth orbit: **Compliant**

Requirement 4.5-2: Limit debris generated by collisions with small objects when operating in Earth orbit: **N/A post mission disposal by passive demise does not require any systems to be functional.**

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

a. Description of spacecraft disposal option selected

Post mission disposal will be achieved by passive demise. Per ESA DRAMA, this will occur 2.93 years after the end of the 2 year mission.

No systems or components are required to accomplish post mission disposal by passive demise.

No maneuvering of the vehicle is needed to accomplish a post mission disposal by passive demise.

b. Calculation of area-ton-mass ratio after post mission disposal.

Mass: 2.949 kg

Effective Cross Section Area: During the post mission disposal period with random tumbling, the effective cross sectional area A_{eff} can be estimated as follows:

A_{max} = Maximum cross sectional area face, 0.0395 m^2 (solar panel deployed face)

A_1 = Second Face cross sectional area, 0.0187 m^2

A_2 = Third Face cross sectional area, 0.01 m^2

$A_{eff} = (A_{max} + A_1 + A_2) /2, = 0.0341\ m^2$

So, the spacecraft has 0.0341 m^2 effective cross sectional area.

Area to mass ratio: 0.0116 m^2/kg.

At the beginning of the post mission disposal phase, which is the end of the 2 year mission, the spacecraft will have lowered orbit due to drag, to 405 km. See Figure 7-1.

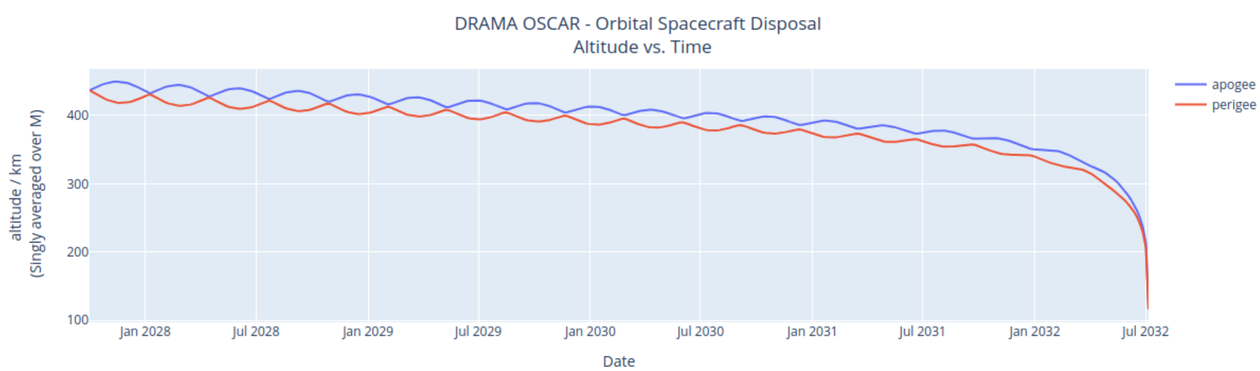


Figure 7-1 Altitude vs Time During Mission

So the post mission disposal duration is calculated starting at 405 km. Based on this, ESA DRAMA shows that demise is estimated to occur 2.93 years after the end of the mission.

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

Requirement 4.6-1, Disposal for space structures passing through LEO: **Compliant**

Per ESA DRAMA, the passive demise occurs 2.93 years after the end of the mission, which is less than the limit of 5 years after the end of the mission set by FCC.

Requirement 4.6-2, Disposal for space structures near GEO: **N/A**

Requirement 4.6-3, Disposal for space structures between LEO and GEO:

N/A Requirement 4.6-4, Reliability of post-mission disposal operations: **N/A**

8. ODAR Section 7: Assessment of Spacecraft Reentry Hazards a. Detailed description of spacecraft components by size, mass, material, shape, and original location on the space vehicle, if the atmospheric reentry option is selected

Table 8-1 lists spacecraft components used in the reentry analysis.

Table 8-1: Components List

Num	Name	Parent	Qty	Material	Shape	Mass (kg)	Width (m)	Length (m)	Height (m)
1	HORIS-3 AND HORIS-4 1.5U CubeSat	0	1	Aluminum 7075-T6	Box	3	0.1	0.15	0.1
2	Chassis	1	1	Aluminum 6061-T6	Box	0.12	0.1	0.15	0.1
3	Outer Panels	1	4	Aluminum 7075-T6	Box	0.045	0.09	0.14	0.002
4	Hinge Base	1	4	Aluminum 7075-T6	Box	0.012	0.03	0.0713	0.005

5	Hinge Panel	1	8	Aluminum 7075-T6	Box	0.0015	0.009	0.01	0.0088
6	Solar Panel Assembly	1	2	Fiberglass	Flat Plate	0.06	0.077	0.135	
7	Power Board	1	1	Fiberglass	Flat Plate	0.015	0.0785	0.079	
8	Battery Cell	1	2	Stainless Steel (generic)	Cylinder	0.045	0.0183	0.065	
9	Processor Board	1	1	Fiberglass	Box	0.023	0.092	0.092	0.0016
10	Compute Module	1	1	Fiberglass	Flat Plate	0.009	0.03	0.055	
11	M4 Processor Board	1	1	Fiberglass	Flat Plate	0.024	0.092	0.092	
12	Radio Board	1	1	Fiberglass	Flat Plate	0.011	0.042	0.072	
13	XS-Band Antenna PCB	1	1	Fiberglass	Flat Plate	0.036	0.1	0.1	
14	Software Defined Radio	1	1	Fiberglass	Flat Plate	0.0092	0.028	0.065	
15	X-Band Antenna	1	1	Fiberglass	Flat Plate	0.006	0.043	0.043	
16	S-Band Antenna	1	1	Macor Ceramic	Box	0.0018	0.01	0.02	0.004
17	Iridium/GPS PCB	1	1	Fiberglass	Flat Plate	0.036	0.098	0.098	
18	Iridium/GPS Antenna	1	1	Fiberglass	Flat Plate	0.01	0.04	0.04	
19	StarTracker Camera Assembly	1	1	Aluminum (generic)	Box	0.04	0.02	0.04	0.02
20	Base	1	1	Aluminum 7075-T6	Box	0.384	0.096	0.096	0.076
21	35mm Imager	1	1	Aluminum (generic)	Box	0.01	0.017	0.02	0.013
22	35mm Sensor Housing	1	1	Aluminum 7075-T6	Box	0.005	0.029	0.029	0.0218
23	35mm Lens	1	1	Aluminum (generic)	Box	0.053	0.037	0.0695	0.037
24	35 mm Sensor Mount	1	1	Aluminum 7075-T6	Box	0.006	0.0123	0.037	0.005
25	35 mm Upper Mount	1	1	Aluminum 7075-T6	Box	0.006	0.0165	0.045	0.0045

26	35 mm Center Mount	1	1	Aluminum 7075-T6	Box	0.006	0.023	0.0412	0.0045
27	35 mm Bottom Mount	1	1	Aluminum 7075-T6	Box	0.008	0.0231	0.049	0.0056
28	25mm Imager	1	1	Aluminum (generic)	Box	0.026	0.026	0.026	0.026
29	25mm Lens	1	1	Aluminum (generic)	Box	0.028	0.0338	0.0482	0.0338
30	25 mm Sensor Mount	1	1	Aluminum 7075-T6	Box	0.004	0.0095	0.0378	0.005
31	25 mm Upper Mount	1	1	Aluminum 7075-T6	Box	0.008	0.024	0.046	0.005
32	25 mm Lower Mount	1	1	Aluminum 7075-T6	Box	0.007	0.0194	0.0396	0.005
33	Thermal Imager Mount	1	1	Aluminum 7075-T6	Box	0.007	0.0226	0.0387	0.0055
34	LWIR 18mm Imager	1	1	Aluminum (generic)	Box	0.071	0.0345	0.0448	0.0289
35	Torque PCB	1	4	Fiberglass	Flat Plate	0.0215	0.07	0.08	
36	Torque Coil	1	3	Copper Alloy	Box	0.06	0.049	0.059	0.006
37	Torque CMG	1	1	Aluminum 7075-T6	Box	0.083	0.014	0.058	0.058

b. Summary of objects expected to survive an uncontrolled reentry, using NASA Debris Assessment Software (DAS), NASA Object Reentry Survival Analysis Tonal (ORSAT), or comparable software

Based on ESA DRAMA, no components are expected to survive reentry.

c. Calculation of probability of human casualty for the expected year of uncontrolled reentry and the spacecraft orbital inclination

The risk of human casualty is 0.

d. Assessment of spacecraft compliance with Requirement 4.7-1

4.7-1, Limit the risk of human casualty: **Compliant**

9. ODAR Section 8: Assessment for Special Classes on Space Missions

Specify the special mission class(es) and detail how the ODAR addresses additional measures applied to the mission.

The requirements for special classes of space missions do not apply to the HORIS-3 AND HORIS-4 mission.

Assessment of Compliance with Requirement 4.8-1

Requirement 4.8-1, Assessment of compliance for special classes of space missions: **N/A**

This is not considered a special class of space mission, as described by the provided list.

10. ODAR Section 9-14: Launch Vehicle

Sections 9 through 14 pertain to the Launch Vehicle and are not covered in this document.

Appendix A – ESA DRAMA Activity Logs

Debris Risk Assessment and Mitigation Analysis Tool Suite
MASTER(-based) Assessment of Risk Events Statistics Software

This file was generated by the program OSCAR (v2.5.2) on 2026-07-21 13:54:10

Input data:
Begin date (YYYY-MM-DD hh:mm:ss): 2027-10-01 12:00:00.00

Target orbit:
6878.137 - semi-major axis / km
1e-07 - eccentricity / -
97.4 - inclination / °
0 - right ascension of the ascending node / °
0 - argument of perigee / °
Spacecraft parameters:
0.01447589 - cross-sectional area / m²
2.949 - dry mass / kg
2.2 - drag coefficient / -
1.3 - reflectivity coefficient / -
Disposal option: Circular Delayed De-Orbit

Disposal system: Chemical Propulsion

Lifetime limit: 5 year(s)

Lifetime margin: 5

Solar and geometric activity:
Scenario: Latest Prediction
Solar & geomagnetic activity data used: ESA Data
General Output:
Final date and Kepler elements:
2032-07-05 01:12:20 - Date (YYYY-MM-DD hh:mm:ss)
6496.36 - Semi-major axis / km
0.0002747 - Eccentricity / -
97.44 - Inclination / °
29.16 - Right ascension of ascending node / °
218.06 - Argument of perigee / °
66.01 - Mean anomaly / °
Force model used with FOCUS v2.6.2
Geopotential: GEM-T1 (6x6)
Drag: NRLMSISE-00
4.761 Remaining lifetime of disposal orbit / year(s)
(incl. margin of 5.00 % according to ISO 27852:2011.)
422.3 Perigee altitude / km
required to match the user-defined lifetime limit
(compliant using the chosen disposal system)
43.36 Required delta-v / m/s
0.23 Required propellant mass / kg
disposalDuration not found in PyOscar.oop Disposal duration / day(s)
1 Number of Hohmann maneuvers / -
Propulsion System:
Chemical Propulsion system parameters:
Specific impulse: 60 s

ESA Space Debris Mitigation Requirements compliance check:
500.0 Perigee altitude / km
required to ensure compliance with the
ESA Space Debris Mitigation Requirements
(compliant using the chosen disposal system)