EcAMSat Formal Orbital Debris Assessment Report (ODAR) and End of Mission Plan (EOMP)

In accordance with NPR 8715.6A, this report is presented as compliance with the required reporting format per NASA-STD-8719.14.

Note: This analysis only covers the NASA portion of the EcAMSat Spacecraft mission.

No analysis is implied for the launch vehicle or other systems.

Report Version: 3/18/15

Document Data is Not Restricted.

This document contains no proprietary, ITAR, or export controlled information.

DAS Software Version Used In Analysis: v2.0.2

This document is a part of the EcAMSat Satellite Project Documentation, which is controlled by the EcAMSat Project Configuration Manager under the direction of the EcAMSat Satellite Project at NASA, Ames Research Center, Moffett Field, California.

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Record of Revisions

Rev	Date	Affected Pages	Description of Change	Author (s)
A	3/18/15	All	Initial Release	Dave Squires

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Self Assessment and OSMA Review of ODARs (per Appendix A.2 Of NASA-STD-8719.14):

Per NASA-STD-8719.14 and NPR-8716.5, paragraph 2.2:

> Each delivered ODAR will be reviewed by the OSMA and by the Space Operations Mission Directorate with technical assistance from the NASA ODPO. After the OSMA review, the check sheet … will be returned to the Headquarters Sponsoring Mission Directorate Program Executive for distribution back to the program. OSMA will also provide a copy to the orbital debris lead at the Center supporting the program for assisting with corrective actions.
>
> Each EOMP is reviewed by OSMA with technical assistance from the NASA Orbital Debris Program Office (ODPO) with final approval and all associated risks accepted by the Associate Administrator of the Mission Directorate sponsoring the mission.

A self assessment of ODAR and EOMP compliance is provided below (next page) in accordance with the assessment formats provided in Appendix sections A.2, and B.2 of NASA-STD-8719.14 The matrices in the NPR are identical and therefore only a single matrix is provided in this combined ODAR-EOMP report. A copy of the assessment is included in Appendix B for use in OSMA review and updates.

The EcAMSat project notes that the ODAR is initially due prior to PDR, and the EOMP is initially due, much later, at the Safety and Mission Success review (SMSR). Accordingly, content in the initial release of this document should be viewed as ODAR-driven content, and any modified version of this document released for SMSR will reflect changes to EOM planning. The final ODAR and EOMP document will reflect any inputs or change requirements received from OSMA.

Please refer to Appendix B of this document for a copy of the independent assessment from OSMA.

ODAR and EOMP Self-Assessment Report Evaluation: EcAMSat Mission

Requirement #	Launch Vehicle (NA, see note 1)				Spacecraft			Comments
	Compliant	Not Compliant	Incomplete	Standard Non-Compliant	Compliant	Not Compliant	Incomplete	
4.3-1.a	☐	☐	☒	☐	☒	☐	☐	No intentional release of debris in LEO. See note 1.
4.3-1.b	☐	☐	☒	☐	☒	☐	☐	No intentional release of debris in LEO. See note 1.
4.3-2	☐	☐	☒	☐	☒	☐	☐	N/A - LEO. See note 1.
4.4-1	☐	☐	☒	☐	☒	☐	☐	Battery explosion probability is estimated at 0.000064. (Requirement is <0.001).
4.4-2	☐	☐	☒	☐	☒	☐	☐	
4.4-3	☐	☐	☒	☐	☒	☐	☐	No planned breakups.
4.4-4	☐	☐	☒	☐	☒	☐	☐	No planned breakups.
4.5-1	☐	☐	☒	☐	☒	☐	☐	Prob of large object collision using DAS 2.0.2 = 0.00000
4.5-2	☐	☐	☒	☐	☒	☐	☐	Prob of small object collision using DAS 2.0.2 = 0.000000
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					☒	☐	☐	NA. No tethers used.

Notes:

1. EcAMSat is a secondary payload, and the launch vehicle is not managed by NASA. The EcAMSat Project will therefore not analyze LV debris or other payload debris issues.

Assessment Report Format:

ODAR and EOMP Section Format Requirements:

The ODAR and EOMP follow the formats prescribed in NASA-STD-8719.14, Appendix A.1 and B.1 respectively. Required content is provided for each "ODAR section…" 2 through 8 below for the E. coli anti-microbial satellite (EcAMSat). ODAR sections 9 through 14 of the NASA Standard are not covered here as they apply to the launch vehicle. The EOMP section uses the ODAR as a primary basis of compliance information.

Sections provided below are labeled according to ODAR and EOMP Section Numbering.

Mission Description:

The EcAMSat satellite launches as a secondary payload on a Falcon 9 launch from Vandenberg. The primary payload is not a NASA payload.

At the launch site EcAMSat is inserted into QuadPack-2XL™ payload dispenser attached to a SHERPA™ payload adapter. The QuadPack-2XL™ dispenser provides full enclosure of the satellite until deployment in orbit. The satellite will operate in its elliptical orbit using passive magnetic stabilization until natural orbit decay results in reentry. There are no propellants.

EcAMSat's primary experiment functions are to be completed within 1 month after launch. Data from the experiment, spacecraft, and payload systems are retrieved over the course of subsequent weeks, up to 6 months. After the primary mission data is collected, the satellite operations will be handed over to Santa Clara University to meet ongoing secondary objectives.

The spacecraft and payload commanding, and data telemetry delivery functions utilize a 2.4 GHz ISM-band transceiver. A small subset of satellite telemetry will also be transmitted over an amateur beacon for Education and Public Outreach purposes. There are no propellants.

Launch vehicle and launch site: Falcon-9, Vandenberg AFB, California.

Proposed launch date: December 2015, to February 2016

Mission duration: Planned primary operations are to last roughly six (6) months after launch. This period will be followed by many years in LEO secondary operations. From launch, the spacecraft is expected to remain in LEO for ~17.1 years prior to reentry after natural decay of the orbit.

Launch and deployment profile, including all parking, transfer, and operational orbits with apogee, perigee, and inclination: As detailed in Figure 1, the Falcon 9 will launch its primary payload into a circular sun-synchronous orbit at 720 Km altitude and an inclination of 98.27 degrees, 10:30 AM local time descending node. Following separation of the primary payload, a perigee lowering burn (SES-3) will be used to establish the final elliptical orbit for EcAMSat with perigee at 450 Km. At 5093 seconds after launch, the SHERPA™ secondary payload adapter separates from the second stage motor. The second stage executes collision avoidance maneuvers, and the SHERPA™ subsequently initiates separation events for 72 secondary payloads including EcAMSat.

EcAMSat is deployed to, and decays naturally from, an operational circular orbit defined as follows:
Apogee: 720 Km

Perigee: 450 Km
Inclination: ~98.27 degrees.

EcAMSat has no propulsion and therefore does not actively change orbits. There is no parking or transfer orbit.

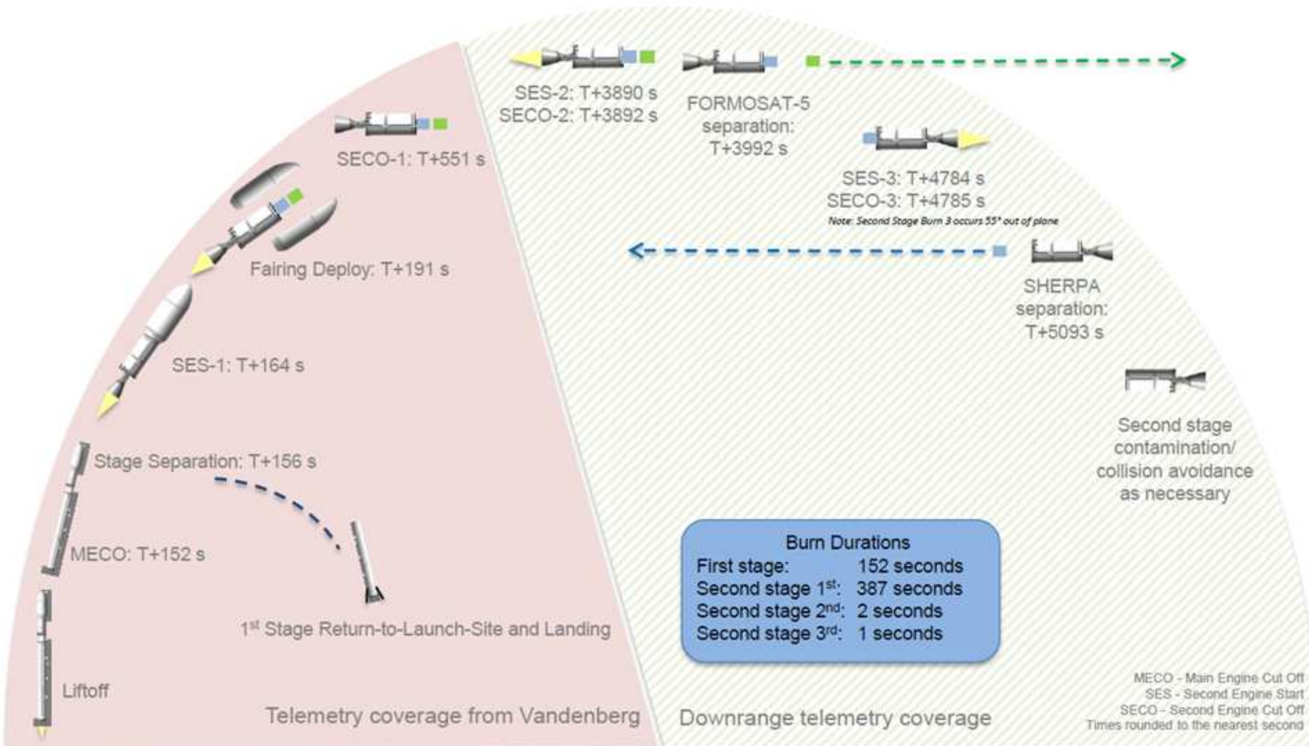


Figure 1, Launch and post-boost sequence figures for the SpaceX Falcon-9 launch.

Interaction or potential physical interference with other operational Spacecraft: No intentional interactions are planned. Interferences will be the subject of collision avoidance analysis provided by the launch provider and/or payload dispenser provider.

ODAR/EOMP Section 1: Program Management and Mission Overview

EcAMSat will be built at Ames Research Center.

Mission Directorate: ESMD, Advanced Concepts Division

Program Executive: David Tomko, PhD., NASA Headquarters

Program/project manager: Stevan Spremo / ARC

Senior scientist: A.C. Matin, PhD. Stanford University

Foreign government or space agency participation: None.

Summary of NASA's responsibility under the governing agreement(s): N/A.

Schedule of mission design and development milestones from NASA mission selection through proposed launch date, including spacecraft PDR and CDR (or equivalent) dates*:

Mission Selection:	10/2014
Mission Preliminary Design Review:	2/28/2013
Mission Critical Design Review:	9/4/2013
FRR:	7/1/2015
PSRR:	7/2015
ORR:	7/2015
Launch:	~December 2015 – February 2016
Primary Mission Complete	L+3 months
Secondary Mission Complete	L+17 years (at reentry due to orbit decay)

ODAR/EOMP Section 2: Spacecraft Description

Physical description of the spacecraft:
As shown in figure 2, EcAMSat is a rectangular box with a monopole antenna boom attached one corner. The satellite dimensions are 9.9 cm x 22.6 cm x 36.6 cm.

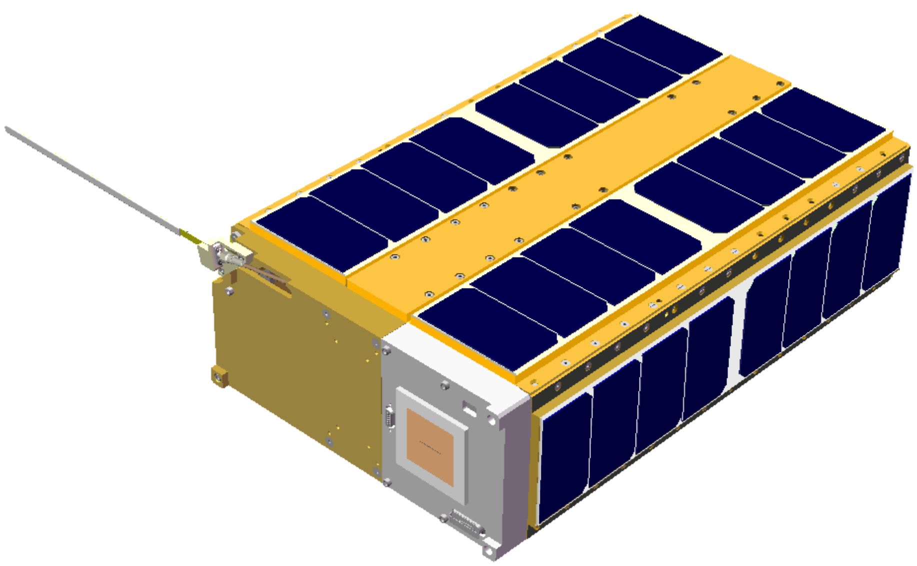


Figure 2, EcAMSat in free-flight operational configuration

Total satellite mass at launch, including all propellants and fluids: 13.6 kg.
Dry mass of satellite at launch, excluding solid rocket motor propellants: 13.6 kg

Description of all propulsion systems (cold gas, mono-propellant, bi-propellant, electric, nuclear): None.
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:
Payload 1: The EcAMSat payload contains 6-8 cc of water with nutrient, dye, and salts in solution. The payload contains less than two liters of air within the experiment payload which is pressurized at 14.7 PSIa (+1.3 / -3.0 PSIa). In this air volume, aqueous fluids are contained in plastic tubing, storage bags, and a plastic biological growth containment. These fluids are pumped to pressures of 2 PSI above the payload air pressure for 4-5 days, with short pulses (seconds to minutes) of 6 PSI relative to the payload air pressure.
Payload 2 (or mass model): Identical to payload 1, or replaced with an inactive, unpressurized mass model.

Fluids in Pressurized Batteries: None. EcAMSat uses unpressurized standard COTS Lithium-Ion battery cells.

Description of attitude control system and indication of the normal attitude of the spacecraft with respect to the velocity vector:
EcAMSat uses passive magnetic attitude control. The magnets are aligned along the length of the spacecraft so the spacecraft aligns with the Earth's magnetic field. This means that the satellite cross section presented to the direction of flight varies with location within the Earth's magnetic field as well as varying due to nutation (coning rotation). The variable tilt toward earth caused by local magnetic inclination should have little effect on orbit decay. Variable magnetic declination effects (angles away from true north) should average out with minimal or no effect. The satellite will be rotating about its long axis at a rate of one rotation every 45-90 seconds. Coning rotation of this spin vector about the net angular momentum vector will occur at a rate of roughly 20-30 rotations per orbit and with a cone half-angle likely to be less than 12.5 degrees (as demonstrated by coarse measurements on the 2006 GeneSat mission), and possibly less than 2.5 degrees as suggested by a mathematical model.

Description of any range safety or other pyrotechnic devices: No pyrotechnic devices are used.

Description of the electrical generation and storage system: UTJ Solar Panels and Lithium Ion Batteries.

Identification of any other sources of stored energy not noted above: None.

Identification of any radioactive materials on board: None.

ODAR/EOMP 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: There are no intentional releases.

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

(Note that the EcAMSat project does not manage the release of staging components, deployment hardware, or other objects).

4.3-2, Mission Related Debris Passing Near GEO: COMPLIANT

ODAR/EOMP 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:

In-mission 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 nine (9) independent, mutually exclusive failure modes to lead to explosion.

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

Not applicable. 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:

Lithium Ion batteries shall be passivated at EOM. This will be done using accelerated cycles of battery charge-discharge. The accelerated charge-discharge cycles are implemented by commanding payload and bus system loads to remain ON, thereby increasing power consumption. A few percent of chargeable capacity (<7kJ) could remain in the batteries at the end of the passivation cycling.

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

Not applicable.

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: 0.000064

Supporting Rationale and FMEA details:
Payload Pressure Vessel Failure:
The nominal payload pressure is 14.7 PSIa. At this pressure, the payload is considered to be a “sealed container” and not a pressure vessel. This contained pressure is considered to be insufficient to cause catastrophic failure of the vessel.

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 vessel due to the lack of penetration energy.

Probability: Very Low. It is believed to be less than 0.1% 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: Complete proto-qualification and acceptance shock, vibration, thermal cycling, and vacuum tests followed by maximum system rate-limited charge and discharge to prove that no internal short circuit sensitivity exists.
Combined faults required for realized failure: Environmental testing **AND** functional charge/discharge tests must both be ineffective in discovery of the failure mode.

Expected Probability: ~0.000008 based on millions of cells in circulation (we will use 10 million as a baseline). We also have more than 40 cell-years of demonstrated reliability in space, increasing the probability of acceptable performance of this design. Cell-years are not considered in that calculation, but should add confidence in the reliability estimate. An overall derating factor of 10 is applied to account for space environment effects.
Hence, given that the spacecraft uses 8 cells: Pf = 0.0000001*10*8 = 0.000008

Failure Mode 2: Internal thermal rise due to high load discharge rate.
Mitigation 2: Each cell includes a positive temperature coefficient (PTC) variable resistance device that ensures high rate discharge is limited to acceptable levels if thermal rise occurs in the battery.
Combined faults required for realized failure: The PTC must fail **AND** spacecraft thermal design must be incorrect **AND** external over current detection and disconnect function must fail to enable this failure mode.

Expected Probability: ~0.000008 based on millions of cells in circulation (we will use 10 million as a baseline) and discharge rate limit protection. We also have more than 40 cell-years of demonstrated reliability in space, increasing the probability of acceptable performance of this design. Cell-years are not considered in that calculation, but should add confidence in the reliability estimate. An overall derating factor of 10 is applied to account for space environment effects.
Hence, given that the spacecraft uses 8 cells: Pf = 0.0000001*10*8 = 0.000008

Failure Mode 3: Overcharging and excessive charge rate.
Mitigation 3: The satellite bus battery charging circuit design eliminates the possibility of the batteries being overcharged if circuits function nominally. This circuit is proto-qualification tested for survival in shock, vibration, and thermal-vacuum environments. The charge circuit disconnects the incoming current when battery voltage indicates normal full charge at 4.2V. If this circuit fails to operate, continuing charge can cause gas generation. The batteries include overpressure release vents that allow gas to escape, virtually eliminating any explosion hazard.
Combined faults required for realized failure:
1) For overcharging: The charge control circuit must fail to function **AND** the PTC device must fail (or temperatures generated must be insufficient to cause the PTC device to modulate) **AND** the overpressure relief device must be inadequate to vent generated gasses at acceptable rates to avoid explosion.
2) For excessive charge rate: The solar arrays are capable of charging at a maximum rate of 1.0 Amp (equivalent to 0.113 CmA). This charge rate is well below the recommended 0.7 CmA charging rate limit for the Panasonic CGR18650 type batteries used. Due to solar panel current limits and their direction-facing arrangement on the satellite, there is no physical means of exceeding charging rate limits, even if only a single string of batteries were accepting charge. For this failure mode to become active three strings of batteries must fail to accept a charge **AND** the charge control circuit on the remaining string fails such that it allows charging below 5.8 volts (2-cell series voltage) **AND** the PTC device must fail **AND** the overpressure relief vent must be inadequate to relive generated gas.

Estimated Probability: ~0.000008 based on millions of cells in circulation (we will use 10 million as a baseline), quadruple fault protection of proven devices for overcharge protection, and zero probability of exceeding charge rate limit due to absence of power generation. We also have more than 40 cell-years of demonstrated reliability in space, increasing the probability of acceptable performance of this design. Cell-years are not considered in that calculation, but should add confidence in the reliability estimate. An overall derating factor of 10 is applied to account for space environment effects. Hence, given that the spacecraft uses 8 cells: Pf = 0.0000001*10*8 = 0.000008

Failure Mode 4: 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 4: This failure mode is negated by a) proto-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) obviation of such other mechanical failures by proto-qualification and acceptance environmental tests (shock, vibration, thermal cycling, and thermal-vacuum tests).
Combined faults required for realized failure: The PTC must fail **AND** an external load must fail/short-circuit **AND** external over-current detection and disconnect function must all occur to enable this failure mode.
Estimated Probability: ~0.000008 based on millions of cells in circulation (we will use 10 million as a baseline to account for standard protection built into each cell), and triple fault of proven devices for excessive discharge protection. We also have more than 40

cell-years of demonstrated reliability in space, increasing the probability of acceptable performance of this design. Cell-years are not considered in that calculation, but should add confidence in the reliability estimate. An overall derating factor of 10 is applied to account for space environment effects.
Hence, given that the spacecraft uses 8 cells: Pf = 0.0000001*10*8 = 0.000008

Failure Mode 5: Inoperable vents.
Mitigation 5: Battery vents are not inhibited by the battery holder design or the spacecraft.
Combined effects required for realized failure: The manufacturer fails to install proper venting.
Expected Probability: ~0.000008 based on millions of cells in circulation (we will use 10 million as a baseline). We also have more than 40 cell-years of demonstrated reliability in space, increasing the probability of acceptable performance of this design. Cell-years are not considered in that calculation, but should add confidence in the reliability estimate. An overall derating factor of 10 is applied to account for space environment effects.
Hence, given that the spacecraft uses 8 cells: Pf = 0.0000001*10*8 = 0.000008

Failure Mode 6: Crushing.
Mitigation 6: 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.
Expected Probability: ~0.00000044 based on a worst case the collision probability calculated in requirement 4.5-1. This was below the round off or truncation value for reporting, and provided a result of 0.000000.

Failure Mode 7: Low level current leakage or short-circuit through battery pack case or due to moisture-based degradation of insulators.
Mitigation 7: 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.
Expected Probability: ~0.000008 based on millions of units in circulation (we will use 10 million as a baseline). We also have more than 40 cell-years of demonstrated reliability in space, increasing the probability of acceptable performance of this design. Cell-years are not considered in that calculation, but should add confidence in the reliability estimate. An overall derating factor of 10 is applied to account for space environment effects.
Hence, given that the spacecraft uses 8 cells: Pf = 0.0000001*10*8 = 0.000008

Failure Mode 8: Excess temperatures due to orbital environment and high discharge combined for the hottest full-sun orbit occurring more than 2 years after launch.
Mitigation 8: The spacecraft thermal design negates this possibility as demonstrated in the O/OREOS mission which used the same battery pack and experienced full sun orbits totaling roughly 13 weeks in 3.5 years of operations without failure. When full-sun orbits occur, battery temperatures are held in acceptable range by restricting power consumption. This is acceptable since the full-sun orbits phases occur at a time when power demands are low, long after the primary mission is completed.

Thermal rise has also been analyzed and tested in combination with space environment temperatures showing that batteries do not exceed normal allowable operating temperatures. Temperatures will be well below temperatures of concern for explosions with the estimated maximum operating temperature reaching just below 9 ^{O}C, allowing an operational temperature margin of 36 ^{O}C relative to the datasheet recommended maximum of 45 ^{O}C during charging. The margin during discharge is 51 ^{O}C relative to a datasheet recommended maximum of 60 ^{O}C.

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.
Expected Probability: ~0.000008 based on millions of units in circulation (we will use 10 million as a baseline) and discharge rate limit protection. We also have more than 40 cell-years of demonstrated reliability in space overall, increasing the probability of acceptable performance of this design. Cell-years are not considered in that calculation, but should add confidence in the reliability estimate. An overall derating factor of 10 is applied to account for vacuum thermal environment effects.
Hence, given that the spacecraft uses 8 cells: Pf = 0.0000001*10*8 = 0.000008

Failure Mode 9: Polarity reversal due to over-discharge caused by continuous load during periods of negative power generation vs. consumption.
Mitigation 9: In nominal operations, the spacecraft EPS design negates this mode because the processor will stop when voltage drops too low. This disables ALL connected loads, creating a guaranteed power-positive charging scenario. The spacecraft will not restart or connect any loads until battery voltage is above the acceptable threshold. At this point, only the main C&DH board, EPS board, radios, and low-level payload heater power are enabled, maintaining a power positive mode until ground commands are received for continuing mission functions.
Combined faults required for realized failure: The microcontroller must stop executing code **AND** significant loads must be commanded/stuck "on" **AND** power margin analysis must be wrong **AND** the charge control circuit must fail for this failure mode to occur.
Expected Probability: ~0.000008 based on millions of units in circulation (we will use 10 million as a baseline. We also have more than 40 cell-years of demonstrated reliability in space, increasing the probability of acceptable performance of this design. Cell-years are not considered in that calculation, but should add confidence in the reliability estimate. An overall derating factor of 10 is applied to account space environment effects. Hence, given that the spacecraft uses 8 cells: Pf = 0.0000001*10*8 = 0.000008

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:** As part of the education and public outreach aspect of this mission, the spacecraft is intended to continue operations until natural orbit decay occurs. The only significant stored energy is in the battery packs. If desired prior to reentry at EOM, energy storage capacity in the Lithium Ion batteries can be degraded more rapidly than normal through application of repeated 100% depth of discharge cycles (cycling between 60% and 100% depth of discharge). This function is enabled when a command is sent to increase power consumption in the bus and payload. This results in an accelerated number of charge-discharge cycles per day. A few percent of chargeable capacity (<7kJ) could remain in the batteries at the end of the passivation cycling. It is predicted that the chargeable capacity can be dropped to this level in less than 2 years after the command is issued, most likely faster since the batteries will have been in orbit for many years prior to initiating this command.

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

Assessment of spacecraft compliance with Requirements 4.5-1 and 4.5-2 (per DAS v2.0.2, and calculation methods provided in NASA-STD-8719.14, section 4.5.4):

Requirement 4.5-1. Limiting debris generated by collisions with large objects when operating in Earth orbit: For each spacecraft and launch vehicle orbital stage in or passing through LEO, the program or project shall demonstrate that, during the orbital lifetime of each spacecraft and orbital stage, the probability of accidental collision with space objects larger than 10 cm in diameter is less than 0.001 (Requirement 56506).

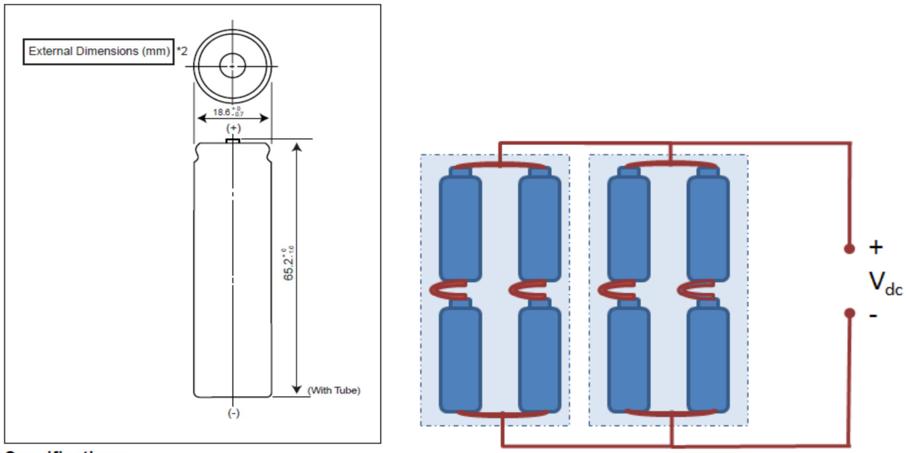
> **Large Object Impact and Debris Generation Probability:** COMPLIANT. Below required probability; less than the round off value of the DAS 2.0.2 software.

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 post mission disposal requirements is less than 0.01 (Requirement 56507).

- **Small Object Impact and Debris Generation Probability:** 0.000000; COMPLIANT
- **Identification of all systems or components required to accomplish any post mission disposal operation, including passivation and maneuvering:**

Critical surface1: Battery Passivation Circuits
EcAMSat can passivate its battery pack at end of mission through use of a command. The spacecraft bus and payload contain circuits that must execute the battery passivation functions. The integrated circuits that control the passivation function are on printed circuit cards within the spacecraft bus frame and sealed payload vessel. These integrated circuits have negligible areal density associated mainly with the plastic encapsulant, circuit card material, and conformal coating surrounding the semiconductor chips. To be highly conservative, this analysis considers the protective benefit of only the exposed areal density of the plastic encapsulant. This is estimated using polycarbonate plastic at 1250 kg/m^3. Assuming 0.5 mm thickness and a total of 2 cm^2 surface area for the devices of concern, mass of 0.125g, and areal density of 0.0625 g/cm^2 are estimated. The closest distance of this surface to the spacecraft outer wall panels is approximately 3cm.

Critical Surface 2: Battery Cells/Battery Pack outer layers
If one of the cells in a battery pack became disabled due to meteoroid impact, then passivating one of the series-connected cells would be prevented. Each battery cell has attributes as provided in figure 3. There are 8 cells in all. The cells are contained behind the external panels of the spacecraft (described above). Surface area per cell is 43.5 cm^2. Mass per cell is 44.5 grams. Hence the per-cell areal density may be seen as 1.02 g/ cm^2. But, estimating that failure might be induced at meteoroid penetration depth of roughly one tenth the cell diameter, the effective areal density used will be (1/100)*1.02 g/cm^2, or 0.0102 g/cm^2. The closest distance of this surface to the spacecraft outer wall panels is approximately 1cm.



Specifications

Nominal Voltage		3.6V
Standard Capacity*1		2150mAh
Dimensions*2	Diameter	18.6 +0/-0.7mm
	Height	65.2 +0/-1.0mm
	Weight	Approx. 44.5g
Operating Temperature		-20 to +60 C

Figure 3, Left: EcAMSat Battery Cell (1 of 8); Right: Battery Pack wiring.

Outer walls:
The critical surfaces are surrounded on all sides by aluminum-backed solar panels made of 6061-T6 Aluminum. The thinnest aluminum areas are 1.5 mm thick. Therefore, the effective areal density of these panels is at least 0.406 g/cm^2 (ignoring solar cell contributions) as seen from the location of critical surfaces. In some cases an effective density of twice or three times this value may be seen for surfaces that are intermediated by payload walls and/or other structures using predominantly 7075-T6 and 6061-T6 aluminum. Values selected for this analysis appear in the DAS 2.0.2 log file provided in “ODAR Section 7” content in this document.

ODAR/EOMP 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. There is no propulsion system.

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: ~13.6 kg
Cross-sectional Area: 0.07 m^2 (Calculated by DAS 2.0.2 for the configuration of figure 1).
Area to mass ratio: 0.07/13.6 = 0.00515 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 EcAMSat satellite reentry is COMPLIANT using method “a.” above. EcAMSat will be left in a 450 Km by 720 Km elliptical orbit, reentering in **~17.1** years after launch with orbit history as shown in Figure 4 (analysis assumes an approximate random tumbling behavior).

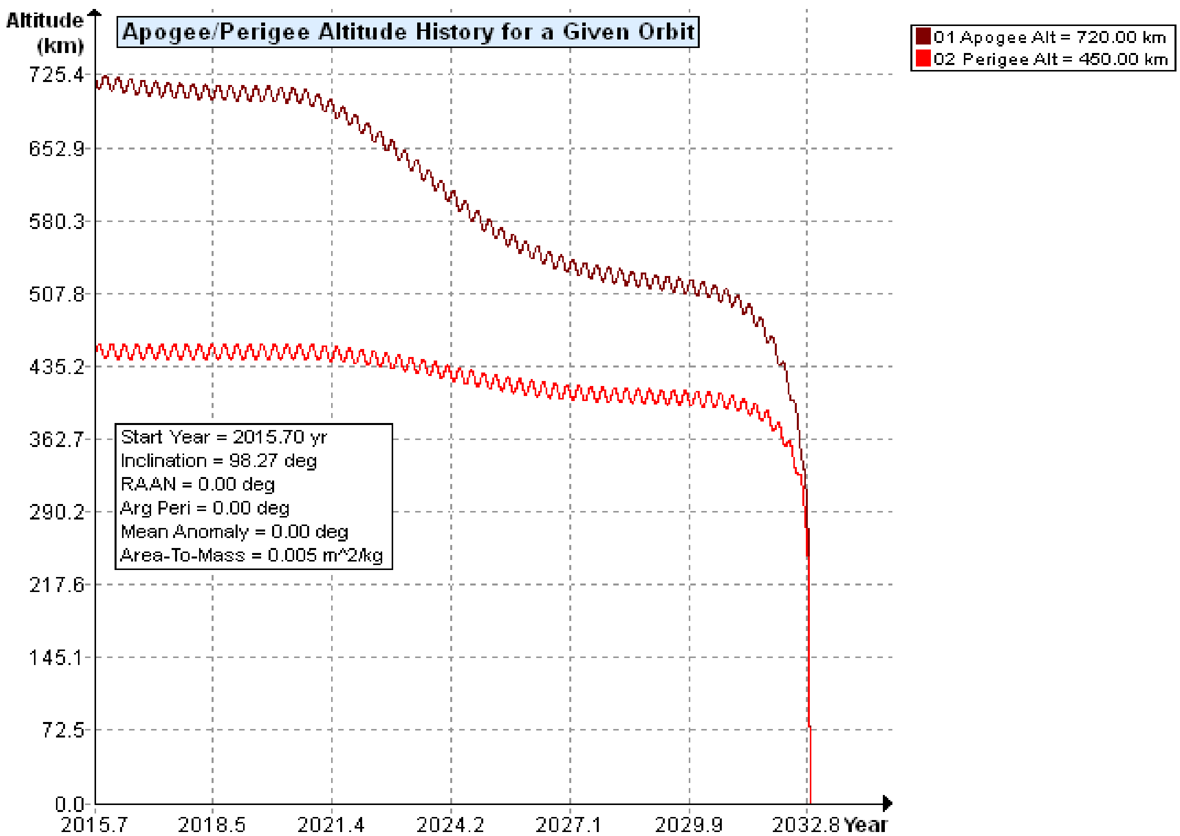


Figure 4, EcAMSat Projected Orbit History.

Requirement 4.6-2. Disposal for space structures near GEO.
Analysis: Not applicable. EcAMSat uses LEO.

***Requirement 4.6-3. Disposal for space structures between LEO and GEO*.**
Analysis: Not applicable. EcAMSat orbit is LEO.

Requirement 4.6-4. Reliability of Post mission Disposal Operations
Analysis: There are no EcAMSat post mission disposal operations. The spacecraft is allowed to reenter by natural decay of orbit (see Requirement 4.6.1, above).

ODAR/EOMP 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 EcAMSat is compliant with the requirement. Only the titanium screws reach the ground. Total human casualty probability is reported by the DAS software as 1:0. This is an erroneous output, presumably meaning a probability of zero since the impact energy is below 15 J. As seen in the analysis outputs below (see Requirement 4.7-1), the impact kinetic energies are 0.000000 Joules and impact casualty areas are all 0.000000 square meters for all objects except the titanium screws which have impact energy of 0.016915 joules, and Debris Casualty Area of 29.282001 m^2. This is an extraordinarily low energy and should be considered of negligible concern as the energy is far below the 15 J level of concern for human casualties.

Analysis (per DAS v2.0.2):

```
12 17 2014; 15:06:36PM  DAS Application Started
12 17 2014; 15:09:34PM  Project Data Saved To File

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

=============== End of Requirement 4.3-1 ===============

12 17 2014; 15:11:53PM  Mission Editor Changes Applied
12 17 2014; 15:12:01PM  Processing Requirement 4.3-1: Return Status:  Not Run
=====================
No Project Data Available
=====================
=============== End of Requirement 4.3-1 ===============

12 17 2014; 15:12:06PM  Processing Requirement 4.3-2: Return Status: Passed

=====================
No Project Data Available
=====================
=============== End of Requirement 4.3-2 ===============

12 17 2014; 15:12:12PM  Requirement 4.4-3:  Compliant
=============== End of Requirement 4.4-3 ===============

12 17 2014; 15:12:23PM  Processing Requirement 4.5-1: Return Status:  Passed

==============
Run Data
==============
**INPUT**

      Space Structure Name = EcAMSat
      Space Structure Type = Payload
      Perigee Altitude = 450.000000 (Km)
      Apogee Altitude = 720.000000 (Km)
      Inclination = 98.270000 (deg)
      RAAN = 0.000000 (deg)
      Argument of Perigee = 0.000000 (deg)
      Mean Anomaly = 0.000000 (deg)
```

```
      Final Area-To-Mass Ratio = 0.005147 (m^2/kg)
      Start Year = 2015.600000 (yr)
      Initial Mass = 13.600000 (kg)
      Final Mass = 13.600000 (kg)
      Duration = 17.000000 (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 = 0.000002
      Returned Error Message: Normal Processing
      Date Range Error Message: Normal Date Range
      Status = Pass
=============== End of Requirement 4.5-1 ===============

12 17 2014; 15:23:58PM  Requirement 4.5-2:  Compliant

==================================================
Spacecraft = EcAMSat
Critical Surface = Passivation_Circuit
==================================================
**INPUT**
      Apogee Altitude = 720.000000 (Km)
      Perigee Altitude = 450.000000 (Km)
      Orbital Inclination = 98.270000 (deg)
      RAAN = 0.000000 (deg)
      Argument of Perigee = 0.000000 (deg)
      Mean Anomaly = 0.000000 (deg)
      Final Area-To-Mass = 0.005147 (m^2/kg)
      Initial Mass = 13.600000 (kg)
      Final Mass = 13.600000 (kg)
      Station Kept = No
      Start Year = 2015.600000 (yr)
      Duration = 17.000000 (yr)
      Orientation = Random Tumbling
      CS Areal Density = 0.062500 (g/cm^2)
      CS Surface Area = 0.000200 (m^2)
      Vector = (0.000000 (u), 0.000000 (v), 0.000000 (w))
      CS Pressurized = No
      Outer Wall 1   Density: 0.406000 (g/cm^2)  Separation: 3.000000 (cm)
      Outer Wall 2   Density: 1.218000 (g/cm^2)  Separation: 33.000000 (cm)
      Outer Wall 3   Density: 0.406000 (g/cm^2)  Separation: 3.000000 (cm)
      Outer Wall 4   Density: 0.812000 (g/cm^2)  Separation: 13.000000 (cm)
      Outer Wall 5   Density: 0.406000 (g/cm^2)  Separation: 5.000000 (cm)
      Outer Wall 6   Density: 0.406000 (g/cm^2)  Separation: 5.000000 (cm)
```

```
**OUTPUT**
      Probability of Penetration = 0.000000
      Returned Error Message: Normal Processing
      Date Range Error Message: Normal Date Range

==================================================
Spacecraft = EcAMSat
Critical Surface = Battery_wall
==================================================
**INPUT**
      Apogee Altitude = 720.000000 (Km)
      Perigee Altitude = 450.000000 (Km)
      Orbital Inclination = 98.270000 (deg)
      RAAN = 0.000000 (deg)
      Argument of Perigee = 0.000000 (deg)
      Mean Anomaly = 0.000000 (deg)
      Final Area-To-Mass = 0.005147 (m^2/kg)
      Initial Mass = 13.600000 (kg)
      Final Mass = 13.600000 (kg)
      Station Kept = No
      Start Year = 2015.600000 (yr)
      Duration = 17.000000 (yr)
      Orientation = Random Tumbling
      CS Areal Density = 0.010200 (g/cm^2)
      CS Surface Area = 0.004350 (m^2)
      Vector = (0.000000 (u), 0.000000 (v), 0.000000 (w))
      CS Pressurized = No
      Outer Wall 1   Density: 0.406000 (g/cm^2)  Separation: 1.000000 (cm)
      Outer Wall 2   Density: 1.218000 (g/cm^2)  Separation: 35.000000 (cm)
      Outer Wall 3   Density: 0.406000 (g/cm^2)  Separation: 1.000000 (cm)
      Outer Wall 4   Density: 0.812000 (g/cm^2)  Separation: 11.000000 (cm)
      Outer Wall 5   Density: 0.406000 (g/cm^2)  Separation: 1.000000 (cm)
      Outer Wall 6   Density: 0.406000 (g/cm^2)  Separation: 1.000000 (cm)

**OUTPUT**
      Probability of Penetration = 0.000000
      Returned Error Message: Normal Processing
      Date Range Error Message: Normal Date Range


12 17 2014; 15:50:23PM  Processing Requirement 4.6    Return Status :  Passed

==============
Project Data
==============
**INPUT**
      Space Structure Name = EcAMSat
      Space Structure Type = Payload

      Perigee Altitude = 450.000000 (Km)
      Apogee Altitude = 720.000000 (Km)
      Inclination = 98.270000 (deg)
      RAAN = 0.000000 (deg)
```

```
      Argument of Perigee = 0.000000 (deg)
      Mean Anomaly = 0.000000 (deg)
      Area-To-Mass Ratio = 0.005147 (m^2/kg)
      Start Year = 2015.600000 (yr)
      Initial Mass = 13.600000 (kg)
      Final Mass = 13.600000 (kg)
      Duration = 17.000000 (yr)
      Station Kept = False
      Abandoned = True
      PMD Perigee Altitude = 295.295431 (Km)
      PMD Apogee Altitude = 338.826941 (Km)
      PMD Inclination = 98.235978 (deg)
      PMD RAAN = 278.651482 (deg)
      PMD Argument of Perigee = 123.913818 (deg)
      PMD Mean Anomaly = 0.000000 (deg)

**OUTPUT**
      Suggested Perigee Altitude = 295.295431 (Km)
      Suggested Apogee Altitude = 338.826941 (Km)
      Returned Error Message = Passes LEO reentry orbit criteria.

      Released Year = 2032 (yr)
      Requirement = 61
      Compliance Status = Pass


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

12 17 2014; 15:53:54PM  *********Processing Requirement 4.7-1
      Return Status :  Passed

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

name = EcAMSat
quantity = 1
parent = 0
materialID = 8
type = Box
Aero Mass = 13.600000
Thermal Mass = 13.600000
Diameter/Width = 0.200000
Length = 0.300000
Height = 0.100000

name = Sat_End_Plates
quantity = 2
parent = 1
materialID = 9
type = Flat Plate
Aero Mass = 0.500000
Thermal Mass = 0.500000
Diameter/Width = 0.100000
```

```
Length = 0.200000

name = Sat_Misc_Struct
quantity = 1
parent = 1
materialID = 9
type = Box
Aero Mass = 1.000000
Thermal Mass = 1.000000
Diameter/Width = 0.100000
Length = 0.100000
Height = 0.100000

name = PL-Ti_screws
quantity = 80
parent = 1
materialID = 66
type = Cylinder
Aero Mass = 0.000200
Thermal Mass = 0.000200
Diameter/Width = 0.002500
Length = 0.010000

name = PL_Vessel_1
quantity = 1
parent = 1
materialID = 9
type = Box
Aero Mass = 3.000000
Thermal Mass = 3.000000
Diameter/Width = 0.100000
Length = 0.200000
Height = 0.100000

name = PL_Vessel_2
quantity = 1
parent = 1
materialID = 9
type = Box
Aero Mass = 3.000000
Thermal Mass = 3.000000
Diameter/Width = 0.100000
Length = 0.200000
Height = 0.100000

name = Pump_Stepper
quantity = 1
parent = 1
materialID = 76
type = Cylinder
Aero Mass = 0.050000
Thermal Mass = 0.050000
Diameter/Width = 0.030000
```

```
Length = 0.086000

name = Bus
quantity = 2
parent = 1
materialID = 9
type = Box
Aero Mass = 1.388000
Thermal Mass = 1.388000
Diameter/Width = 0.100000
Length = 0.100000
Height = 0.100000

name = Solar_Panels
quantity = 6
parent = 1
materialID = 8
type = Flat Plate
Aero Mass = 0.127000
Thermal Mass = 0.127000
Diameter/Width = 0.100000
Length = 0.300000

name = Misc_hardware
quantity = 1
parent = 1
materialID = 9
type = Box
Aero Mass = 1.900000
Thermal Mass = 1.900000
Diameter/Width = 0.100000
Length = 0.100000
Height = 0.100000

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

name = EcAMSat
Demise Altitude = 77.995558
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

*************************************
name = Sat_End_Plates
Demise Altitude = 74.415363
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

*************************************
name = Sat_Misc_Struct
Demise Altitude = 72.403261
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
```

```
*************************************
name = PL-Ti_screws
Demise Altitude = 0.000000
Debris Casualty Area = 29.282001
Impact Kinetic Energy = 0.016915

*************************************
name = PL_Vessel_1
Demise Altitude = 69.285472
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

*************************************
name = PL_Vessel_2
Demise Altitude = 69.285472
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

*************************************
name = Pump_Stepper
Demise Altitude = 77.708105
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

*************************************
name = Bus
Demise Altitude = 70.453199
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

*************************************
name = Solar_Panels
Demise Altitude = 77.275332
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

*************************************
name = Misc_hardware
Demise Altitude = 68.084183
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

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

Requirements 4.7-1b, and 4.7-1c below are non-applicable requirements because EcAMSat does not use controlled reentry.

4.7-1, b) **NOT APPLICABLE.** 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).

4.7-1 c) **NOT APPLICABLE.** 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).

ODAR/EOMP Section 7A: Assessment of Spacecraft Hazardous Materials

There are no materials on the spacecraft that are designated as hazardous except the BSL-2 *E.coli* bacteria used in the antibiotic susceptibility experiment. The small amount of fluid containing this material will vaporize due o the heat of reentry, killing the bacteria such that none survives to reach Earth.

Material Description and Condition (State, quantity, and pressure of materials):

A: Chemical and Commercial Name of Material	B: Hazard Description	C: Condition At Launch	D: Condition In Operational Orbit	E: Condition At EOM	F: Condition At End of Passivation	G: Condition After Reentry
Uropathogenic *E. coli* (UPEC) bacteria typical of human urinary tract infections.	No hazard expected. Material will incinerate on reentry.	Dormant UPEC cells suspended in < 50mL at ~14.7 PSIa in potassium buffer solution.	Increased UPEC cell quantity suspended in < 50mL at ~14.7 PSIa in growth medium and antibiotic solution.	< 50mL at ~14.7 PSIa of non-viable UPEC cells suspended in depleted growth medium and antibiotic solution.	Same as column E.	No hazard expected. Material will incinerate on reentry.

ODAR/EOMP Section 8: Assessment for Tether Missions

Not applicable. There are no tethers in the EcAMSat mission.

Appendix A: Acronyms

ARC	Ames Research Center
CDR	Critical Design Review
cm	Centimeter
CmA	Discharge Rate as a Fraction of Rated Capacity in Milliamperes
cm^2	Centimeter Squared
COTS	Commercial Off-The-Shelf (items)
C&DH	Command and Data Handling
DAS	Debris Assessment Software
DCA	Debris Casualty Area
deg	Degree
EcAMSat	Escherichia coli Antimicrobial Small Satellite System
EPS	Electrical Power Subsystem
EOM	End Of Mission
EOMP	End Of Mission Plan
ESMD	Exploration Systems Mission Directorate
FRR	Flight Readiness Review
g	Gram
GEO	Geosynchronous Earth Orbit
ITAR	International Traffic In Arms Regulations
J	Joules
kg	kilogram
KE	Kinetic Energy
kJ	Kilo-Joules
Km	kilometer
LEO	Low Earth Orbit
m^2	Meters Squared
N/A	Not Applicable.
ODAR	Orbital Debris Assessment Report
ODPO	Orbital Debris Program Office
ORR	Operations Readiness Review
OSMA	Office of Safety and Mission Assurance
PDR	Preliminary Design Review
Pf	Probability of Failure
PL	Payload
PMD	Post Mission Disposal
PSIa	Pounds Per Square Inch, Absolute
PSRR	Pre-Ship Readiness Review
PTC	Positive Temperature Coefficient
RAAN	Right Ascension of the Ascending Node
S&MA	Safety and Mission Assurance
Ti	Titanium
u, v, w	Cartesian Coordinate System
UTJ	Ultra Triple Junction (Solar Cell)
yr	year

Appendix B: OSMA ODAR and EOMP Evaluation, EcAMSat Mission

(SUGGESTED Assessment Pending Review)

Reqm't #	Launch Vehicle				Spacecraft			Comments
	Compliant	Not Compliant	Incomplete	Standard Non Compliant	Compliant or N/A	Not Compliant	Incomplete	
4.3-1.a	☐	☐	☒	☐	☒	☐	☐	No Intentional release of debris
4.3-1.b	☐	☐	☒	☐	☒	☐	☐	No Intentional release of debris
4.3-2	☐	☐	☒	☐	☒	☐	☐	N/A - LEO
4.4-1	☐	☐	☒	☐	☒	☐	☐	Explosion probability is estimated at 0.000064 (Requirement: <0.001).
4.4-2	☐	☐	☒	☐	☒	☐	☐	
4.4-3	☐	☐	☒	☐	☒	☐	☐	No intentional break-up planned
4.4-4	☐	☐	☒	☐	☒	☐	☐	No intentional break-up planned
4.5-1	☐	☐	☒	☐	☒	☐	☐	Prob of large object collision using DAS 2.0.2 = 0.00000 < 0.001
4.5-2	☐	☐	☒	☐	☒	☐	☐	Prob of small object collision using DAS 2.0.2 = 0.000000 < 0.01
4.6-1(a)	☐	☐	☒	☐	☒	☐	☐	Anticipated natural reentry within 17 years of EOM
4.6-1(b)	☐	☐	☒	☐	☒	☐	☐	N/A
4.6-1(c)	☐	☐	☒	☐	☒	☐	☐	N/A
4.6-2	☐	☐	☒	☐	☒	☐	☐	N/A - LEO
4.6-3	☐	☐	☒	☐	☒	☐	☐	N/A - LEO
4.6-4	☐	☐	☒	☐	☒	☐	☐	
4.6-5	☐	☐	☒	☐	☒	☐	☐	
4.7-1	☐	☐	☒	☐	☒	☐	☐	DCA of 0.0 m^2 when excluding objects with KE < 15J
4.8-1					☒	☐	☐	N/A – No Tethers

Additional Comments:

1.