TechEdSat-3P Formal Orbital Debris Assessment Report (ODAR)

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

Report Version: 5 (4/23/2013)

DAS Software Used in This Analysis: DAS v2.0

APPROVAL

Signature	Date	Signature	Date
	5-1-13		5-1-13
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	5/1/2013		5/1/13
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Prepared By:

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Suzanne Aleman **NASA Headquarters, Office of Safety and Mission Assurance, Orbital Debris Manager**	**Date**
Terrence W. Wilcutt **NASA Chief, Safety and Mission Assurance** **NASA Headquarters**	**Date**

Record of Revisions				
Rev	Date	Affected Pages	Description of Change	Author (s)
A	3/23/2013	All	Initial Release	Romalyn Mirador
B	3/25/2013	All	Range safety/ pyrotechnic device, DAS analysis	
C	4/2/2013	17-23	DAS Analysis	
D	4/2/2013	29	Wiring Diagrams	
E	4/24/13	All	Signature sheets (reference), Mission and Spacecraft description, Battery failure modes	

Table of Contents

Self-assessment and OSMA assessment of the ODAR using the format in Appendix A.2 of NASA-STD-8719.14:

A self-assessment is provided below in accordance with the assessment format provided in Appendix A.2 of NASA-STD-8719.14. In the final ODAR document, this assessment will reflect any inputs received from OSMA as well.

Orbital Debris Self-Assessment Report Evaluation: TechEdSat-3P Mission

Requirement #	Launch Vehicle				Spacecraft			Comments
	Compliant	Not Compliant	Incomplete	Standard Non Compliant	Compliant	Not Compliant	Incomplete	
4.3-1.a					x			No Debris Released in LEO. See note 1.
4.3-1.b					x			No Debris Released in LEO. See note 1.
4.3-2					x			No Debris Released in GEO. See note 1.
4.4-1					x			See note 1.
4.4-2					x			See note 1.
4.4-3					x			No planned breakups. See note 1.
4.4-4					x			No planned breakups. See note 1.
4.5-1					x			See note 1.
4.5-2					x			
4.6-1(a)					x			See note 1.
4.6-1(b)					x			See note 1.
4.6-1(c)					x			See note 1.
4.6-2					x			See note 1.
4.6-3					x			See note 1.
4.6-4					x			See note 1.
4.6-5					x			See note 1.
4.7-1					x			See note 1.
4.8-1					x			No tethers used.

Notes:

1. The primary payload belongs to Japan Aerospace Exploration Agency (JAXA). This is a NASA primary mission. All of the other portions of the launch stack are non-NASA and TechEdSat-3P is not the lead.

Assessment Report Format:

ODAR Technical Sections Format Requirements:

This ODAR follows the format in NASA-STD-8719.14, Appendix A.1 and includes the content indicated at a minimum in each section 2 through 8 below for the TechEdSat-3P satellite. Sections 9 through 14 apply to the launch vehicle ODAR and are not covered here.

ODAR Section 1: Program Management and Mission Overview

Mission Directorate: ARC Code R Office

Program Executive: Andres Martinez

Program Manager: Andres Martinez

Project Manager: Marcus Murbach

Senior Management: David Korsmeyer

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:	10-1-12
Mission Preliminary Design Review:	12-10-12
Mission Critical Design Review:	April 25th, 2013
PSRP 0/I/II:	May 8th, 2013
Launch:	August 3, 2013
Release from ISS, Begin Operation:	October 4, 2013

Mission Overview:

The Technical Education Satellite 3P (TechEdSat-3P) is a nanosatellite that will test and validate two different technologies in Low Earth Orbit (LEO): demonstration of the Exo-Brake to deorbit the satellite and demonstration of the viability of the Iridium 9603 communication module.

TechEdSat-3P will fly on the HTV-4 mission, soft stowed inside the JEM Small Satellite Orbital Deployer (J-SSOD) Satellite Install Case. The Satellite Install Case is stowed in a Common Transfer Bag (CTB) during launch. The satellite will then be transported onto the International Space Station (ISS) and integrated in the JEM Remote Manipulator System (JEMRMS).

The Japan Aerospace Exploration Agency (JAXA) will deploy the TechEdSat-3P from the JEMRMS using the J-SSOD on October 4, 2013. The J-SSOD uses a spring to "push" the TechEdSat-3P at a velocity of 5 cm/sec and at an angle of 45 degrees relative to the ISS. TechEdSat-3P will be inserted into orbit at an apogee of 413.2 km, perigee of 381.3 km, and an inclination of 51.6 degrees. There are no propellants on the satellite. The interface requirements between the J-SSOD and a satellite are developed based on the CubeSat Design Specification rev.12 published on August 1, 2009 by the California Polytechnic State University with JEM unique requirements.

Transmission will begin 40 minutes after launch from the ISS. The use of the Exo-Brake will deorbit the satellite in approximately 18 days after deployment from the ISS, thus concluding the mission.

Launch vehicle and launch site: Japanese HTV-4, Tsukuba, Japan

Proposed launch date: August 3, 2013

Mission duration: <18 Days

Launch and deployment profile, including all parking, transfer, and operational orbits with apogee, perigee, and inclination:

TechEdSat-3P will be launched at a velocity of 5 cm/sec and at an angle of 45 degrees relative to the JEMRMS into a circular orbit initially approximately 300 or 400 km relative to Earth's surface.

The TechEdSat-3P orbit is defined as follows:

Apogee: 413.2 km

Perigee: 381.3 km

Inclination: 51.6 degrees.

TechEdSat-3P has no propulsion and therefore does not actively change orbits. TechEdSat-3P will deploy the Exo-Brake, slow down, lose altitude, and then disintegrate upon atmospheric reentry approximately 18 days after ISS deployment.

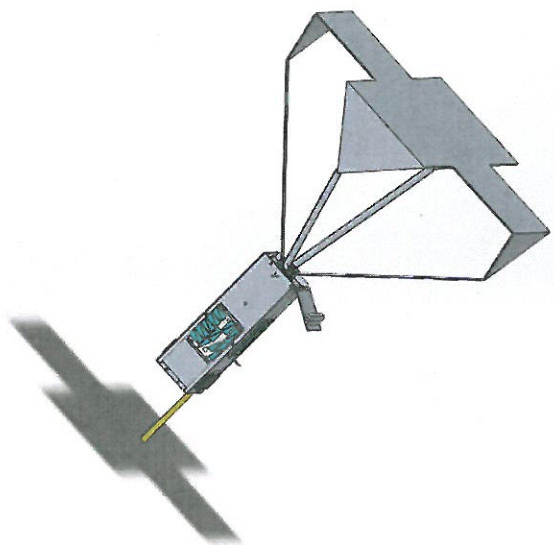

Figure 1: TechEdSat-3P Fully Deployed View

Interaction or potential physical interference with other operational Spacecraft:
One of the main risks that the TechEdSat-3P faces is the possibility of the satellite impacting the ISS. Since the TechEdSat-3P is a 3U CubeSat being launched from the system, and JAXA has shown that the likelihood of any CubeSat impacting the ISS is very minimal, we believe that this will be minimal. This is confirmed by the difference of the ballistic parameter between the satellite and the ISS.

ODAR Section 2: Spacecraft Description

Physical description of the spacecraft:
TechEdSat-3P is a 3U nanosatellite with dimensions of 10 cm X 10 cm X 34 cm and a total mass of about 2.6 kg. The TechEdSat-3P payload carries a deployable Exo-Brake as a technology demonstration. The deployed Exo-Brake has a cross-sectional area of 0.269 m^2. An illustration of the TechEdSat-3P is shown in Figure 2.

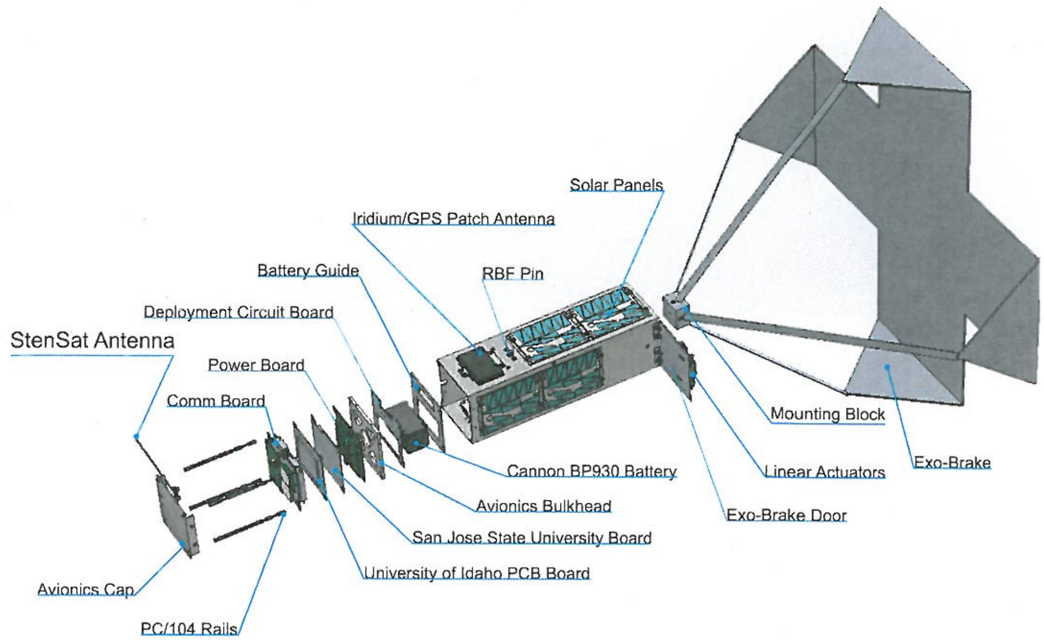

Figure 2: TechEdSat-3P Exploded View with Components

TechEdSat-3P will contain the following systems: one ATMega2560, one PIC32 microcontroller, one AAC Microtec power board, two Iridium 9603 modems, one StenSat beacon, two OEM 615 GPS, one Canon BP-930 battery, eight solar panels, one deployed antenna, four patch antennas, four TMP-36GT9 temperature sensors, 1 BMP-085 temperature and pressure sensor, and 1 MPU-9150 IMU.

- The ATMega2560 is the main board for one 9603 modem and one GPS.
- The PIC32 microcontroller is the main board for one 9603 model and one GPS.
- The AAC Microtec power board will control the deployment of the antenna and Exo-Brake.
- The Iridium 9603 modems will have two patch antennas.
- The StenSat beacon will be transmitting location coordinates to the Ground Station, and will include a 12-cm deployed antenna. The Ham Radio community will be invited to participate to collect the beacon call sign.
- The OEM615 GPS's will have two patch antennas.
- The Canon BP-930 battery will power the system and be recharged by the solar panels.
- There are four temperature sensors that will monitor the temperature of the battery, solar panels and boards in and on the CubeSat.
- There is one temperature and pressure sensor to collect scientific data of the upper atmosphere.
- There is one IMU to monitor the satellite's inertial state.

Total satellite mass at launch, including all propellants and fluids: 2.6 kg

Dry mass of satellite at launch, excluding solid rocket motor propellants: 2.6 kg

Description of all propulsion systems (cold gas, mono-propellant, bi-propellant, electric, nuclear): There will be no propulsion systems on TechEdSat-3P.

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.

Not applicable as there will be no fluids or gasses on board.

Fluids in Pressurized Batteries: None. TechEdSat-3P 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:

TechEdSat-3P does not have any attitude control system but does include a gyro and accelerometer to determine the orientation of the satellite.

Description of any range safety or other pyrotechnic devices: The TechEdSat-3P utilizes Firgelli L12-S linear actuators and does not use any NiChrome Wire heating element or pyrotechnic devices.

Description of the electrical generation and storage system: The power will be generated by Triangular Advanced Solar Cells (TASC) and a Lithium Ion battery. The TASC will be covered with Teflon tape to contain shattering. The battery that will be used is a Canon BP-930. See attached Material Safety Data Sheet (MSDS) (Appendix B). This battery is supplied and approved by the ISS for flight. The dimensions of the battery are 4 x 7 x 3.8 cm and the weight is 0.18 kg.

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

Identification of any radioactive materials on board: None.

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

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

4.3-1, Mission Related Debris Passing Through LEO: COMPLIANT

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

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

Potential causes of spacecraft breakup during deployment and mission operations:

There is no credible scenario that would result in spacecraft breakup during normal deployment and operations.

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

In-mission failure of a battery cell protection circuit could lead to a short circuit, or overcharging, resulting in overheating and a very remote possibility of battery cell explosion. Battery was tested by JSC ISS program and results can be found in document EP-WI-015 Rev C. 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 independent, failure modes that could lead to a battery explosion.

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

There are no planned breakups.

List of components which shall be passivated at End of Mission (EOM) including method of passivation and amount which cannot be passivated:

None.

Rationale for all items which are required to be passivated, but cannot be due to their design:
The TechEdSat-3P will be in deorbit in approximately18 days with successful deployment of the Exo-Brake based on the DAS analysis shown in this report. If the Exo-Brake does not deploy, the TechEdSat-3P will deorbit in 438 days based on the DAS analysis shown in this report. Therefore no postmission passivation will be performed, as the satellite will burn up on reentry at the end of the mission.

Assessment of spacecraft compliance with Requirements 4.4-1 through 4.4-4:

Requirement 4.4-1: Limiting the risk to other space systems from accidental explosions during deployment and mission operations while in orbit about Earth or the Moon:

For each spacecraft and launch vehicle orbital stage employed for a mission, the program or project shall demonstrate, via failure mode and effects analyses or equivalent analyses, that the integrated probability of explosion for all credible failure modes of each spacecraft and launch vehicle is less than 0.001 (excluding small particle impacts) (Requirement 56449).

Compliance statement:

Required Probability: 0.001.

Expected Probability: 0.000.

Supporting Rationale and FMEA details:

Payload Pressure Vessel Failure:

TechEdSat-3P is vented per ISS safety standards. It is not a sealed container.

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 the independent failure modes listed here could cause the ultimate effect (explosion).

Failure mode 1: Battery Internal short circuit.

Mitigation 1: Complete proto-qualification and environmental acceptance tests of the Canon BP-930 battery was performed by JSC ISS program[1]. The acceptance tests are shock, vibration, thermal cycling, and vacuum tests followed by maximum system rate-limited charge and discharge to identify if any 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 a shorted battery.

[1] Battery qualification test program: Specification for Lot Testing and Flight Screening of Canon BP 927 and BP 930 Lithium-ion Batteries (EP-WI-015 Rev C)

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. Battery circuits include fuses and over discharge MOSFETs protection circuit.
Combined faults required for realized failure: The PTC must fail **AND** external over current detection and protection must fail for this failure mode to occur.

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 has been 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 8.4 V. If this circuit fails to operate, continuing charge can cause gas generation. The batteries include Current Interrupter Device (CID) overpressure release vents, which allow gas to escape, and permanently disconnects the battery. Refer to battery qualification test program document EP-WI-015 Rev C.
Combined faults required for realized failure:

1) **For overcharging:** The charge control circuit must fail to function **AND** the PTC device must fail in order to cause uncontrolled charging. The overpressure relief device must be inadequate to vent generated gasses at acceptable rates to avoid explosion.
2) **For excessive charge rate:** The maximum charging rate from a single solar panel when in AM 1.5G conditions (in space, perpendicular to the sun) is 124 mA. The maximum charge rate our battery can accept is 3 A. The battery is a proto-qualified Canon BP-930 battery pack supplied by the JSC ISS program, which is essentially four US18650S cells. The battery itself has two parallel strings of 2 cells connected in series, and thus having 4 cells. Due to solar panel current limits and their direction-facing arrangement on the satellite, there is no physical means of exceeding charging rate or temperature limits, even if only a single string from the battery was accepting charge. For this failure mode to become active one string must fail to accept a charge **AND** the charge control circuit on the remaining string must fail. The overpressure relief vent should keep the battery cells from rupturing, and is thus limited to worst-case effects of overcharging.

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), d) satellite power system is equipped with a 5A fuse.
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 fail to enable this failure mode.

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 and ISS environmental stress screening fails to detect failed vents.

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.

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.

Failure Mode 8: Excess temperatures due to orbital environment and high discharge combined.

Mitigation 8: The spacecraft thermal design will negate this possibility. Thermal rise has been analyzed in combination with space environment temperatures showing that batteries do not exceed normal allowable operating temperatures, which are well below temperatures of concern for explosions.
Combined faults required for realized failure: Thermal analysis **AND** thermal design **AND** mission simulations in thermal-vacuum chamber testing **AND** the PTC device must fail **AND** over-current monitoring and control must all fail for this failure mode to occur.

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, below 7 V. 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 safe mode processor and StenSat beacon are enabled and charging the battery. Once the battery reaches 90% of the peak voltage (around 7.5 V), it will switch to nominal mode and will be able to receive ground commands 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.

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:
The TechEdSat-3P will be in orbit for 18 days with successful deployment of the Exo-Brake. If the Exo-Brake did not deploy, the TechEdSat-3P will be in orbit for approximately 438 days based on the DAS analysis shown in this report. Therefore, no postmission passivation will be performed, as the satellite will burn up on reentry at the end of the mission.

Requirement 4.4-3. Limiting the long-term risk to other space systems from planned breakups:

Compliance statement:
This requirement is not applicable. There are no planned breakups.

Requirement 4.4-4: Limiting the short-term risk to other space systems from planned breakups:

Compliance statement:
This requirement is not applicable. There are no planned breakups.

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

Assessment of spacecraft compliance with Requirements 4.5-1 and 4.5-2 (per DAS v2.0, and calculation methods provided in NASA-STD-8719.14, section 4.5.4):

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

Large Object Impact and Debris Generation Probability: 0.000000; COMPLIANT.

Requirement 4.5-2. Limiting debris generated by collisions with small objects when operating in Earth or lunar orbit: For each spacecraft, the program or project shall demonstrate that, during the mission of the spacecraft, the probability of accidental collision with orbital debris and meteoroids sufficient to prevent compliance with the applicable postmission disposal requirements is less than 0.01 (Requirement 56507).

Small Object Impact and Debris Generation Probability: 0.000000; COMPLIANT

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

6.1 Description of spacecraft disposal option selected: Two cases will be considered for this section. The first case is called "Nominal Deployment" in which the Exo-Brake successfully deploys and deorbits the satellite. The second case is called "Failed Deployment" in which the Exo-Brake fails to deploy and the satellite deorbits naturally due to atmospheric friction.

Case 1: *Nominal Deployment* The satellite will deorbit due to the deployed Exo-Brake. There is no propulsion system.

Case 2: *Failed Deployment* The satellite will deorbit naturally by atmospheric reentry. There is no propulsion system.

6.2 Plan for any spacecraft maneuvers required to accomplish postmission disposal: None.

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

Case 1: *Nominal Deployment*

Spacecraft Mass: 2.6 kg

Cross-sectional Area: 0.269 m^2

Area to mass ratio: 0.269/2.6 = 0.10347 m^2/kg

Case 2: *Failed Deployment*

Spacecraft Mass: 2.6 kg

Cross-sectional Area: 0.01 m^2

Area to mass ratio: 0.01/2.6 = 0.00385 m^2/kg

6.4 Assessment of spacecraft compliance with Requirements 4.6-1 through 4.6-5 (per DAS v 2.0 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 deorbit 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:

Case 1: *Nominal Deployment*

The TechEdSat-3P satellite reentry is COMPLIANT using Method “a.” TechEdSat-3P will re-enter approximately 18 days after launch with orbit history shown in Figure 3.

Case 2: *Failed Deployment*

The TechEdSat-3P satellite reentry is COMPLIANT using Method “a.” TechEdSat-3P will re-enter approximately 438 days after launch with orbit history as shown in Figure 4 (analysis assumes an approximate random tumbling behavior).

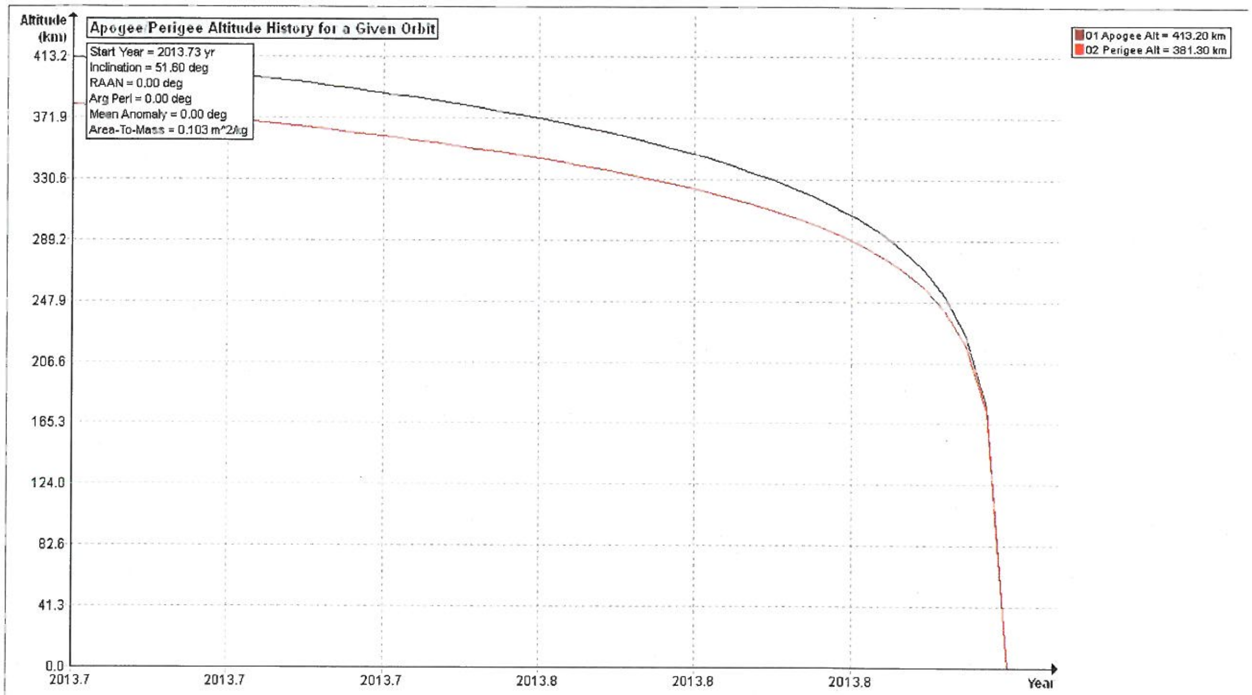

Figure 3: TechEdSat-3P Orbit History for Nominal Deployment

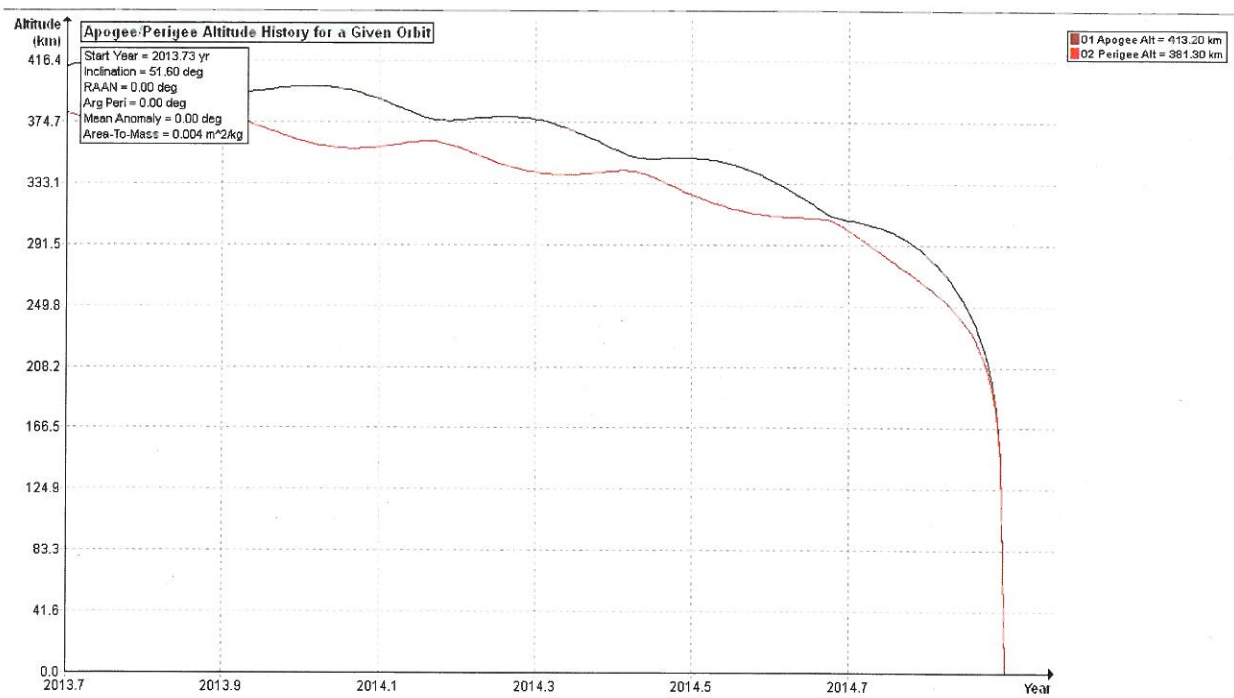

Figure 4: TechEdSat-3P Orbit History for Failed Deployment

Requirement 4.6-2. Disposal for space structures near GEO.

Analysis: Not applicable. TechEdSat-3P orbit is in LEO.

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

Analysis: Not applicable. TechEdSat-3P orbit is in LEO.

Requirement 4.6-4. Reliability of Postmission Disposal Operations

Analysis:

Case 1: *Nominal Deployment*

TechEdSat-3P deorbiting relies on the Exo-Brake deorbiting device. Release of the Exo-Brake will result in deorbiting in approximately <18 days with no disposal or deorbiting actions required.

Case 2: *Failed Deployment*

TechEdSat-3P deorbiting does not rely on deorbiting devices. Release from the ISS with a downward, retrograde vector will result in deorbiting in approximately 438 days with no disposal or deorbiting actions required.

ODAR Section 7: Assessment of Spacecraft Reentry Hazards

Assessment of spacecraft compliance with Requirement 4.7-1:

Requirement 4.7-1. Limit the risk of human casualty: The potential for human casualty is assumed for any object with an impacting kinetic energy in excess of 15 joules:

a. For uncontrolled reentry, the risk of human casualty from surviving debris shall not exceed 0.0001 (1:10,000) (Requirement 56626).

Summary Analysis Results: DAS v2.0 reports that TechEdSat-3P is compliant with the requirement. It predicts that no components reach the ground. The TechEdSat-3P structure, its components and the Exo-Brake will burn up on reentry. As seen in the analysis outputs below, the impact kinetic energies are 0.000000 Joules and impact casualty areas are all 0.000000 square meters. Also, there are no titanium components that will be used on TechEdSat-3P, which also is a reason why no components reach the ground.

```
04 01 2013; 14:24:10PM Processing Requirement 4.3-1:     Return Status :
Not Run

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No Project Data Available
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=============== End of Requirement 4.3-1 ===============
04 01 2013; 14:24:12PM Processing Requirement 4.3-2: Return Status :
Passed

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No Project Data Available
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=============== End of Requirement 4.3-2 ===============
04 01 2013; 14:24:14PM Requirement 4.4-3:  Compliant

=============== End of Requirement 4.4-3 ===============
04 01 2013; 14:24:17PM Processing Requirement 4.5-1:     Return Status :
Passed

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Run Data
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**INPUT**

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      Space Structure Type = Payload
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      Apogee Altitude = 413.200000 (km)
      Inclination = 51.600000 (deg)
      RAAN = 0.000000 (deg)
      Argument of Perigee = 0.000000 (deg)
      Mean Anomaly = 0.000000 (deg)
      Final Area-To-Mass Ratio = 0.103500 (m^2/kg)
      Start Year = 2013.729000 (yr)
      Initial Mass = 2.600000 (kg)
      Final Mass = 2.600000 (kg)
      Duration = 0.027400 (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 = 0.000000
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     Returned Error Message: Normal Processing
     Date Range Error Message: Normal Date Range
     Status = Pass

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=============== End of Requirement 4.5-1 ===============
04 01 2013; 14:24:23PM Requirement 4.5-2:  Compliant
04 01 2013; 14:24:23PM Processing Requirement 4.6   Return Status :
Passed

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

**INPUT**

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     Space Structure Type = Payload

     Perigee Altitude = 381.300000 (km)
     Apogee Altitude = 413.200000 (km)
     Inclination = 51.600000 (deg)
     RAAN = 0.000000 (deg)
     Argument of Perigee = 0.000000 (deg)
     Mean Anomaly = 0.000000 (deg)
     Area-To-Mass Ratio = 0.103500 (m^2/kg)
     Start Year = 2013.729000 (yr)
     Initial Mass = 2.600000 (kg)
     Final Mass = 2.600000 (kg)
     Duration = 0.027400 (yr)
     Station Kept = False
     Abandoned = True
     PMD Perigee Altitude = 335.983219 (km)
     PMD Apogee Altitude = 361.587212 (km)
     PMD Inclination = 51.593764 (deg)
     PMD RAAN = 309.379651 (deg)
     PMD Argument of Perigee = 33.453617 (deg)
     PMD Mean Anomaly = 0.000000 (deg)

**OUTPUT**

     Suggested Perigee Altitude = 335.983219 (km)
     Suggested Apogee Altitude = 361.587212 (km)
     Returned Error Message = Passes LEO reentry orbit criteria.

     Released Year = 2013 (yr)
     Requirement = 61
     Compliance Status = Pass
```

```
==============

=============== End of Requirement 4.6 ===============
04 01 2013; 14:24:45PM *********Processing Requirement 4.7-1
      Return Status :  Passed

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

name = TES3
quantity = 1
parent = 0
materialID = 8
type = Box
Aero Mass = 2.600000
Thermal Mass = 2.600000
Diameter/Width = 0.100000
Length = 0.340000
Height = 0.100000

name = Exo-brake
quantity = 1
parent = 1
materialID = 44
type = Flat Plate
Aero Mass = 0.300000
Thermal Mass = 0.300000
Diameter/Width = 1.000000
Length = 1.000000

name = Battery
quantity = 1
parent = 1
materialID = 39
type = Box
Aero Mass = 0.181000
Thermal Mass = 0.181000
Diameter/Width = 0.039000
Length = 0.076000
Height = 0.036400

name = Patch Antenna
quantity = 4
parent = 1
materialID = 8
type = Box
Aero Mass = 0.060500
Thermal Mass = 0.060500
```

```
Diameter/Width = 0.043400
Length = 0.073700
Height = 0.011700

name = Iridium 9603
quantity = 2
parent = 1
materialID = 5
type = Box
Aero Mass = 0.011000
Thermal Mass = 0.011000
Diameter/Width = 0.029600
Length = 0.031500
Height = 0.008100

name = GPS
quantity = 2
parent = 1
materialID = 77
type = Box
Aero Mass = 0.024000
Thermal Mass = 0.024000
Diameter/Width = 0.046000
Length = 0.071000
Height = 0.011000

name = Stensat
quantity = 1
parent = 1
materialID = 77
type = Flat Plate
Aero Mass = 0.020000
Thermal Mass = 0.020000
Diameter/Width = 0.045000
Length = 0.080000

name = Powerboard
quantity = 1
parent = 1
materialID = 8
type = Box
Aero Mass = 0.043000
Thermal Mass = 0.043000
Diameter/Width = 0.095000
Length = 0.095000
Height = 0.002000

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

```
name = TES3
Demise Altitude = 77.992824
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

*************************************
name = Exo-brake
Demise Altitude = 77.938426
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

*************************************
name = Battery
Demise Altitude = 77.644847
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

*************************************
name = Patch Antenna
Demise Altitude = 75.048816
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

*************************************
name = Iridium 9603
Demise Altitude = 76.246511
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

*************************************
name = GPS
Demise Altitude = 77.720379
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

*************************************
name = Stensat
Demise Altitude = 77.728566
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

*************************************
name = Powerboard
Demise Altitude = 76.612519
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 TechEdSat-3P 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 Section 8: Assessment for Tether Missions

Not applicable. There are no tethers in the TechEdSat-3P mission.

END of ODAR for TechEdSat-3P.

Appendix A: Acronyms

AFRL	Air Force Research Lab
ARC	Ames Research Center
Arg peri	Argument of Perigee
CDR	Critical Design Review
CID	Current Interrupter Device
cm	centimeter
COTS	Commercial Off-The-Shelf (items)
DAS	Debris Assessment Software
EOM	End Of Mission
ESMD	Exploration Systems Mission Directorate
FRR	Flight Readiness Review
GEO	Geosynchronous Earth Orbit
ITAR	International Traffic In Arms Regulations
kg	kilogram
km	kilometer
LEO	Low Earth Orbit

Li-Ion	Lithium Ion
m^2	Meters squared
ml	milliliter
mm	millimeter
N/A	Not Applicable.
ODAR	Orbital Debris Assessment Report
TechEdSat-3P	Technical Education Satellite-3P
ORR	Operations Readiness Review
OSMA	Office of Safety and Mission Assurance
PDR	Preliminary Design Review
PL	Payload
P-POD	Poly Picosatellite Orbital Deployer
PSIa	Pounds Per Square Inch, absolute
PSRR	Pre-Ship Readiness Review
PTC	Positive Temperature Coefficient polyswitch
RAAN	Right Ascension of the Ascending Node
SESLO	Space environment survivability of live organisms (payload)
SMA	Safety and Mission Assurance
Ti	Titanium
USAF	United States Air Force
UTJ	Ultra Triple Junction
yr	year

Appendix B: Battery Material Safety Data Sheet

MATERIAL SAFETY DATA SHEET

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SECTION 1 IDENTIFICATION OF THE SUBSTANCE/MIXTURE AND OF THE COMPANY/UNDERTAKING

Product Name:	Lithium Ion Battery
Product Code:	BP-930
Company Name:	Canon Inc.
Address:	30-2, Shimomaruko 3-Chome, Ohta-ku, Tokyo 146-8501, Japan
Use of the Product:	Battery for Video camera
Supplier:	
Address:	
Phone number:	

With regard to air transport, the International Civil Aviation Organization (ICAO) Packing Instruction 965 Part 1 complies with the Recommendation as is; further, the International Air Transport Association (IATA) adopts ICAO Packing Instruction 965 Part 1. In addition, the regulations of the US Department of Transportation for land, sea and air transportation are based on the UN Recommendations.

SECTION 2 MATERIALS AND INGREDIENTS INFORMATION

IMPORTANT NOTE:	The battery pack uses four US 18650S lithium-ion rechargeable cells and control circuit on the PWB. The cells are connected in 2 parallel strings of 2 cells in series. The battery pack should not be opened or burned since the following ingredients contained within the cells could be harmful under some circumstance if exposed or misused. The cells contain neither metallic lithium nor lithium alloy.	
Cathode:	Lithium-Cobalt Dioxides	(active material)
	Polyvinyldiene Fluoride	(binder)
	Graphite	(conductive material)
Anode:	Graphite	(active material)
	Polyvinyldiene Fluoride	(binder)
Electrolyte:	Organic Solvent	(non-aqueous liquid)
	Lithium Salt	
Others:	Heavy metals such as Mercury, Cadmium, Lead, and Chromium are not used in the cells.	
Enclosure:	Plastic (PC)	

SECTION 3 FIRE HAZARD DATA

In case of fire, use CO_2 or dry chemical extinguishers.

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SECTION 4 HEALTH HAZARD DATA

Under normal condition of use, these chemicals are contained in sealed can. Risk of exposure occurs only if the cells are mechanically abused.

Inhalation:	Contents of an opened cell can cause respiratory irritation.
	Remove to fresh air immediately and call a doctor.
Skin Contact:	Contents of an opened cell can cause skin irritation.
	Wash skin with soap and water.
Eye Contact:	Contents of an opened cell can cause eye irritation.
	Immediately flush eyes thoroughly with water for at least 15 minutes. Seek medical attention.

SECTION 5 PRECAUTIONS FOR SAFE HANDLING AND USE

Storage:	Store within the recommended limit of -20 degrees C to 45 degrees C (-4 degrees F to 113 degrees F), well-ventilated area.
	Do not expose to high temperature (60 degrees C/140 degrees F). Since short circuit can cause burn hazard or safety vent to open, do not store with metal jewelry, metal covered tables, or metal belt.
Handling:	Do not disassemble, remodel, or solder. Do not short + and - terminals with a metal. Do not open the battery pack.
Charging:	Charge within the limits of 0 degrees C to 40 degrees C (32 degrees F to 104 degrees F) temperature.
	Charge with specified charger designed for this battery pack.
Discharging:	Discharge within the limits of -10 degrees C to 50 degrees C (14 degrees F to 122 degrees F) temperature.
Disposal:	Dispose in accordance with applicable federal, state and local regulation.
Caution:	Attach the cover to the battery pack to prevent short circuits.
	Do not disassemble. Do not incinerate. Do not expose to temperature above 140 degrees F.

SECTION 6 SPECIAL PROTECTION INFORMATION

Respiratory Protection:	Not necessary under normal use.
Ventilation:	Not necessary under normal use.
Eye Protection:	Not necessary under normal use.
Protective Gloves:	Not necessary under normal use.

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Appendix C: Wiring Schematics