HiakaSat 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.

Note: This analysis only covers Hawaii Spaceflight Laboratory's spacecraft, HiakaSat.

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

Report Version: 3.2, 4/23/13

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DAS Software Version Used In Analysis: v2.0.2

This document is a part of the HiakaSat Satellite Project Documentation, which is controlled by the HiakaSat Project Configuration Manager under the direction of the HiakaSat Satellite Project at the Hawaii Space Flight Laboratory (HSFL), University of Hawaii, Manoa Campus.

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Approval signatures are not required from ORS. ODAR-defined risk is to be incorporated in the overall mission ODAR.

Record of Revisions				
REV	DATE	AFFECTED PAGES	DESCRIPTION OF CHANGE	AUTHOR (S)
3.1	4/22/13	All	ITAR-Free DRAFT Release, new orbit (Perigeee: 430 km; Apogee: 505 km; Inclination 91-93 degrees)	D. Squires
3.2	4/23/13	All	Completed references to ORS	J. Chan

Table of Contents

Self Assessment of ODAR Requirements (per Appendix A.2 Of NASA-STD-8719.14):

The orbital debris self-assessment matrix on the next page summarizes the result of this report. This template for this table is provided in Appendix A.2 of NASA-STD-8719.14. This is provided for convenience to the DoD ORS office. HSFL will not separately seek review or sign-off from the NASA Orbital Debris Program Office.

Orbital Debris Self-Assessment Report Evaluation: HiakaSat Mission

Requirement #	Launch Vehicle (Not Applicable (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	☐	☐	☒	☐	☒	☐	☐	See note 1.
4.4-2	☐	☐	☒	☐	☒	☐	☐	HiakaSat only has batteries that will be passivated at mission end.
4.4-3	☐	☐	☒	☐	☒	☐	☐	Compliant. No planned breakups. See note 1.
4.4-4	☐	☐	☒	☐	☒	☐	☐	Compliant. No planned breakups. See note 1.
4.5-1	☐	☐	☒	☐	☒	☐	☐	Compliant. See note 1.
4.5-2	☐	☐	☒	☐	☒	☐	☐	Compliant. See note 1.
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. The launch vehicle is the responsibility of Sandia National Laboratories and the DoD Operationally Responsive Space Office

Assessment Report Format:

ODAR Technical Sections Format Requirements:

This ODAR follows the format in NASA-STD-8719.14, Appendix A.1 and includes the required content in each section, 2 through 8, below for the University of Hawaii HiakaSat satellite. Sections 9 through 14 of the standard apply to the launch vehicle ODAR and are not covered here since the launch vehicle is the responsibility of a separate project.

ODAR Section 1: Program Management and Mission Overview

Mission Directorate: Operationally Responsive Space (ORS) Office

Mission Manager, ORS-4: Dr. Jeffry S. Welsh, ORS

HiakaSat Project manager: Jeremy Chan, HSFL

Senior Scientist: Paul Lucey, PhD., Hawaii Institute of Geophysics and Planetology

Senior Management: Dr. Luke Flynn, HSFL

Foreign government or space agency participation: None.

Summary of NASA's responsibility under the governing agreement(s):

N/A. The primary customers for this project include the ORS office, Kirtland AFB; as well as the University of Hawaii.

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

Mission Preliminary Design Review:	June 2010 (HawaiiSat; previous version)
Mission Critical Design Review (Delta):	March 2013
MRR:	June 2013
PSRR:	August 2013
Launch:	~October 30, 2013

Mission Description:

The HiakaSat satellite launches as a primary payload on the Spaceborne Payload Assist Rocket Kauai (SPARK), a Sandia National Laboratory developed launch vehicle with motors developed by Aerojet Corporation. HiakaSat will perform thermal hyper-spectral Earth imaging, and visible spectrum earth imaging in Low Earth Orbit (LEO). The primary experiment functions are completed within four months after launch. Data from the experiments are recovered daily from through ground stations. The satellite will operate in its elliptical orbit using active ADCS (star tracker, reaction wheels, and magnetorquers) for stabilization until natural orbit decay results in reentry. There are no propellants.

Launch vehicle and launch site:

SPARK LV; Pacific Missile Range Facility (PMRF), Barking Sands, Kauai.

Proposed launch date and Mission Duration:

Launch Date: October 30, 2013 (estimated).

Mission Duration (active mission and orbital life):

From the time of launch, the spacecraft is expected to remain in LEO for 5.46 years prior to reentry after natural decay of the orbit. However, the planned primary mission operations are to last only four months after launch. Soon after the completion of primary operations a decision will be made as to when to stop active operations. Upon this decision, a command will be sent to the spacecraft to disconnect the incoming solar power lines from the battery charge control system, allowing the batteries to drain over the course of roughly 24 hours. This leaves the spacecraft inactive and de-energized. It will re-enter through natural decay of its orbit.

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

The SPARK vehicle launches from a rail into the elliptical orbit needed for HiakaSat's imaging mission. Following third stage engine cutoff, the payload separation events begin. There may be some secondary payloads on this launch that are not under the control of HSFL. If so, the deployment sequence will be determined after the secondary payloads are identified. HiakaSat deployment timing is assumed to have priority over other payloads.

HiakaSat is deployed to, operates, and decays naturally from, the elliptical orbit defined as follows:

Orbital Lifetime: 5.46 years (Per DAS 2.0.2)
Perigee: 430km
Apogee: 505km
Inclination: 91 to 93 degrees (analyses in this report are for 93 degrees)
RAAN: 0 (TBR)
Argument of perigee: 0 (TBR)
Ecc: 0.005478
Period: 5636.89 sec (93.85 min)
Mass: 55kg

Reasons for Selection of the Operational Orbit:

The orbit was selected for its imaging benefits to the thermal hyperspectral imager and visible spectrum camera payloads operated by HiakaSat. It was also selected to allow passive disposal of the spacecraft by natural decay of orbit, excluding the complexity, cost, and mass of propulsion design.

Identification of interaction or potential physical interference with other operational spacecraft:

There might be some possibility of interference with multiple other spacecraft that might be deployed on this launch. It is our understanding at this time that four (or more) triple-unit CubeSats will also be deployed for other launch customers. The deployment sequence and collision probabilities with the CubeSats, or any other items deployed, are assumed to be included in a separate ODAR by the SPARK launch project.

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

ODAR Section 2: Spacecraft Description

Physical description of the spacecraft:
HiakaSat is an octagonal box with outer dimensions of 0.43 m x 0.62 m. The satellite structures are aluminum structural members and honeycomb panels. HiakaSat has three radios (UHF, VHF, and S-band), a battery pack, attitude determination and control system, flight computer system, and separation ring for attachment to, and release of the satellite from the launch vehicle once it is in orbit. The spacecraft is covered with fixed, body mounted solar arrays. There are two low resolution visible spectrum imaging camera payloads and one thermal hyperspectral imaging camera payload. One of the payloads contains sealed air or inert gas at a pressure of 1 atm. There is no propulsion system.

Total satellite mass at launch, including all propellants and fluids: 55 kg.

Dry mass of satellite at launch, excluding solid rocket motor propellants: 55 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.

The HiakaSat SUCHI payload contains less than 5,000 cc's of inert gas at 1 atmosphere relative to space.

Fluids in Pressurized Batteries: None. HiakaSat uses unpressurized COTS LiFePO4 battery cells.

Description of attitude control system and indication of the normal attitude of the spacecraft with respect to the velocity vector:

HiakaSat uses one reaction wheel and three magnetic torque coils for attitude control. An IMU provides rate sensing, an external attitude determination is performed through use of a star tracker. Normal attitude is to face the nadir side earthward, orthogonal to the flight direction.

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 LiFePO4 batteries are used.

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)

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

***Requirement 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 planned intentional breakup and 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:

Failure of a battery cell or 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 the seven (7) independent, mutually exclusive failure modes to lead to 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:

Only the LiFePO4 secondary battery pack requires passivation. Passivation is implemented by use of a commanded "EPS Disposal Mode", which is implemented in the EPS software. When in disposal mode the EPS system disconnects the solar panels from the batteries and then depletes the batteries by using the EPS, OBCS, and Telecom as loads. Full discharge should take no more than 24 hours.

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 (0.1%) excluding small particle impacts (Requirement 56449).

Compliance statement:

Required Probability: 1E-03, or lower.

Expected probability: 3.50E-05

Supporting Rationale and FMEA details:

Payload Pressure Vessel Failure:

The nominal SUCHI 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. Also, the spacecraft has many large leak paths allowing for venting of gas in the event that the SUCHI payload gas leaks out.

Battery explosion:

Effect: All failure modes below might result in battery explosion with the possibility of orbital debris generation.

Probability: Very Low. Overall probability of an event leading to battery explosion is estimated to be less than 0.00004 (0.004%) given that any of the independent (not common mode) faults listed in Table 1, can cause the ultimate effect (explosion).

Table 1, Battery Explosion Failure Modes and Probabilities

Independent Failure Mode #	P_f	
1	1.60E-05	
2	1.60E-05	
3	1.60E-11	
4	2.88E-10	
5	2.00E-06	
6	3.00E-9	
7	1.00E-6	
Total Probability (sum):	**3.50E-05**	**(1:28,570)**

Failure mode 1: Battery cell 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: 1.6E-5 based on millions of cells in circulation.

Calculation: We assume 1:10,000,000 as a baseline for manufacturer defects that could lead to internal short circuit in space use, but we derate by a factor of 10 to further account for launch and space environment effects when added to stresses caused during screening and environmental testing.

Hence, given that we will fly 16 cells, the failure probability estimate is:

$$P_f = 1E\text{-}7*10*16 = 1.6E\text{-}5$$

Failure Mode 2: Internal thermal rise due to high load discharge rate.

Mitigation 2: The LiFePO4 battery chemistry is inherently insensitive to internal thermal rise even under very high discharge rates of 20C (>40 Amps) through full discharge. Normal satellite discharge is expected to be less than 5Amps for the heaviest loaded mode, and current limit detection and limiting threshold will be set at 10 Amps. Charging current is limited by the solar panels themselves, which cannot produce more than approximately 3 Amps of charge current (well below the 10A chare rate limit of the battery cells). Charge control circuits balance cell voltages, prevent overvoltage and undervoltage of individual cells and of the overall battery pack, and limit discharge currents well below battery design limits.

Combined faults required for realized failure: The charge control functions and discharge current limit control functions must fail **AND** spacecraft thermal design must be incorrect **AND** battery manufacture must be flawed resulting in gross differences in thermal sensitivity relative to what is normal for LiFePO4 cells.

Expected Probability: 1.6E-5. based on millions of cells in circulation. (See failure mode 1 for calculation.)

Failure Mode 3: Over-voltage charging and excessive charge rate.

Mitigation 3: The charge control circuit design eliminates the possibility of the batteries being overcharged if circuits function nominally. This circuit will be 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 3.6V per cell (or less). 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. Charge current limits cannot be exceeded since solar panels can generate no more than about 3 Amps. The battery pack can handle 10Amps of charge current for fast charging, so here is no design failure mode related to rate of charge.

Combined faults required for realized failure:

1) For over-voltage charging: The charge control circuit must fail to function **AND** the overpressure relief device must be inadequate to vent generated gasses at acceptable rates to avoid explosion.

Estimated Probability: 1.6E-11 based on millions of cell vents in circulation and necessity of concurrent failure of the charge controller.

> Calculation: Again, we assume 1:10,000,000 as a baseline for manufacturer defects that could lead to internal short circuit in space use, but we derate by a factor of 10 to further account for launch and space environment effects when added to stresses caused during screening and environmental testing. The generic failure probability of the COTS charge control circuit is estimated as 1:100,000,000, but will be derated to 1:1,000,000 to match the environmental derating assumptions used for battery cells in circulation (assuming also that charge controllers must roughly match the number of cells and packs in circulation).
>
> Hence, given that we will fly 16 cells and one charge controller, where one cell vent must fail AND the charge controller must fail, the failure probability is:
>
> P_f = 1E-7*10*16*1E-6 = 1.6E-11.

2) For excessive charge rate: As discussed previously, there is no credible failure mode that would allow excessive charge rate. The solar panels cannot generate current beyond the normal allowable charge rate of the battery cells.

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 mitigated 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 or conductive panels 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: Battery thermal vents must fail **AND** an external load must fail/short-circuit **AND** over-current detection and disconnect function must fail enable this failure mode.

Estimated Probability: 2.88E-10 based on millions of cells in circulation, charge control circuit generic reliability, and proto-qualification verified reliability of payloads and subsystems.

> Calculation: Again, we assume 1:10,000,000 as a baseline for manufacturer defects that could lead to internal short circuit in space use, but we derate by a factor of 10 to further account for launch and space environment effects when added to stresses caused during screening and environmental testing. The generic failure probability of the COTS charge control circuit is estimated as 1:100,000,000, but will be derated to 1:1,000,000 to match the environmental derating assumptions used for battery cells in circulation (assuming also that charge controllers must roughly match the number of cells and packs in circulation). The probability of short circuit failure of COTS-based payloads and subsystems is a subset component of the total probability of COTS item failure. The magnitude of the component contribution to COTS electronics short circuit failure probability is not known, so total generic failure probability of the 18 custom-designed and COTS-based payloads and subsystems will be used for this calculation, each derated to 1:1,000,000.
>
> Hence, given that we will fly 16 cells, failure probability is:
>
> $$P_f = 1E\text{-}7*10*16*1E\text{-}6*18 = 2.88E\text{-}10.$$

Failure Mode 5: Crushing

Mitigation 5: This mode is nearly negated by spacecraft design and by use of LiFePO4 cell technology. However, there is a remote probability of secondary explosion due to crushing that might be caused by large orbital debris impacts. Therefore, DAS calculations for requirement 4.5-1 will be used to estimate probability of failure due to crushing.

Combined faults required for realized failure: A debris or meteorite impact must cause a catastrophic failure in an external system **AND** the failure must allow for 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.

Esimated Probability: 2E-6, based on a worst case large object collision probability calculated in requirement 4.5-1.

Failure Mode 6: Excess temperatures due to orbital environment and high discharge rate combined.

Mitigation 6: 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 models **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: 3.000006E-3. The generic failure probability of the overcurrent discharge control circuit is estimated as 1:100,000,000, but will be derated to 1:1,000,000 as an environmental derating assumption for space applications. Given that thermal modeling and margined environmental test are human operated procedures, there is some probability of modeling, design and test performance error which could leave a latent failure mode unscreened. This procedural failure probability will be assessed as 3E-3 assuming peer reviews of designs, models, testing, and validated checklists applied to procedures (NUREG/CR-1278).

Hence, failure probability is: P_f = 1E-6*3E-3 = 3.000006E-9.

Failure Mode 7: Polarity reversal due to over-discharge caused by continuous load during periods of negative power generation vs. consumption, combined with battery voltage differences. Under this condition, the battery cell that discharges completely first will have current forced through it by batteries that are still capable of discharging. This is, in effect, reverse charging and it will damage the weakest cell. The effect of this condition for LiFePO4 cells appears to be limited to loss of function and/or reduced capacity. This is due to the fact that the iron phosphate chemistry does not readily liberate oxygen, therefore limiting oxidation reactions. It is, therefore, not clear that any mitigation is required since the effect of this failure mode is also uncertain. Regardless, the failure mode will be assumed to exist, so existing mitigations are be described.

Mitigation 7: The charge control circuit balances battery voltage to ensure this condition will not occur.

Combined faults required for realized failure: The charge controller must stop fail to balance cell voltages **AND** significant loads must be commanded/stuck "on" **AND** power margin analysis must be wrong.

Expected Probability: 1E-6. The generic failure probability of the COTS charge control circuit is estimated as 1:100,000,000, but will be derated to 1:1,000,000 to match the environmental derating assumptions used for battery cells in circulation (assuming also that charge controllers must roughly match the number of cells and packs in circulation).

Hence, failure probability is: $P_f = 1E\text{-}6$

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 postmission 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 spacecraft Electrical Power Subsystem (EPS) provides passivation capability. The passivation occurs when "EPS Disposal Mode" is commanded. This mode disconnects the solar panels from the batteries and then depletes the batteries by using the EPS CPU and Telecom as loads.

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).

- **DAS 2.0.2 Calculation:**
  Large Object Impact and Debris Generation Probability: 0.000001; 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).

- **DAS 2.0.2 Calculation:**
  Probability of Damage from Small Debris: 0.000020; COMPLIANT
- **Identification of all systems or components required to accomplish any postmission disposal operation, including passivation and maneuvering:**

  The systems required to accomplish battery passivation are as follows:
  - Battery Passivation Box
  - UHF or VHF radio Boxes
  - On-board Computing System (OBCS Box)

  The box enclosures of these three systems are defined as their respective critical surfaces within DAS 2.0.2. The outer wall is defined as the solar panels because they have lower net areal density than the Zenith and Nadir decks. The probability of penetration of each critical surface was calculated by DAS 2.0.2 (see appendix A, requirement 4.5-2, for details).

  Functional failure scenario:

  The passiviation function begins with an “EPS Disposal Mode” command from the ground through either the UHF or VHF radio. The OBCS then commands the Battery Passivation Box to actuate latching relays that disconnect the solar panel power lines from the battery charging circuits. Within 24 hours the loads connected to the batteries will cause them to discharge completely.

ODAR Section 6: Assessment of Spacecraft Postmission 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 postmission disposal: NONE.

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

Spacecraft Mass: 55 kg

Cross-sectional Area: 0.28 m^2 (Calculated by DAS 2.0.2 for the configuration descried in section 1, using a gravity gradient assumption to emulate controlled nadir pointing behavior. Note that the mission plan should result in random tumbling after six months in orbit, so gravity gradient assumption is a worst case assumption (longest orbit duration)).

Area to mass ratio: 0.28/55 = 0.0051 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 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 HiakaSat satellite reentry is COMPLIANT using method "a.". HiakaSat will be left in an elliptical orbit (apogee: 525 km; perigee: 450 km), reentering in **~5.46** years after launch with orbit history as shown in Figure 1 (analysis assumes random tumbling behavior).

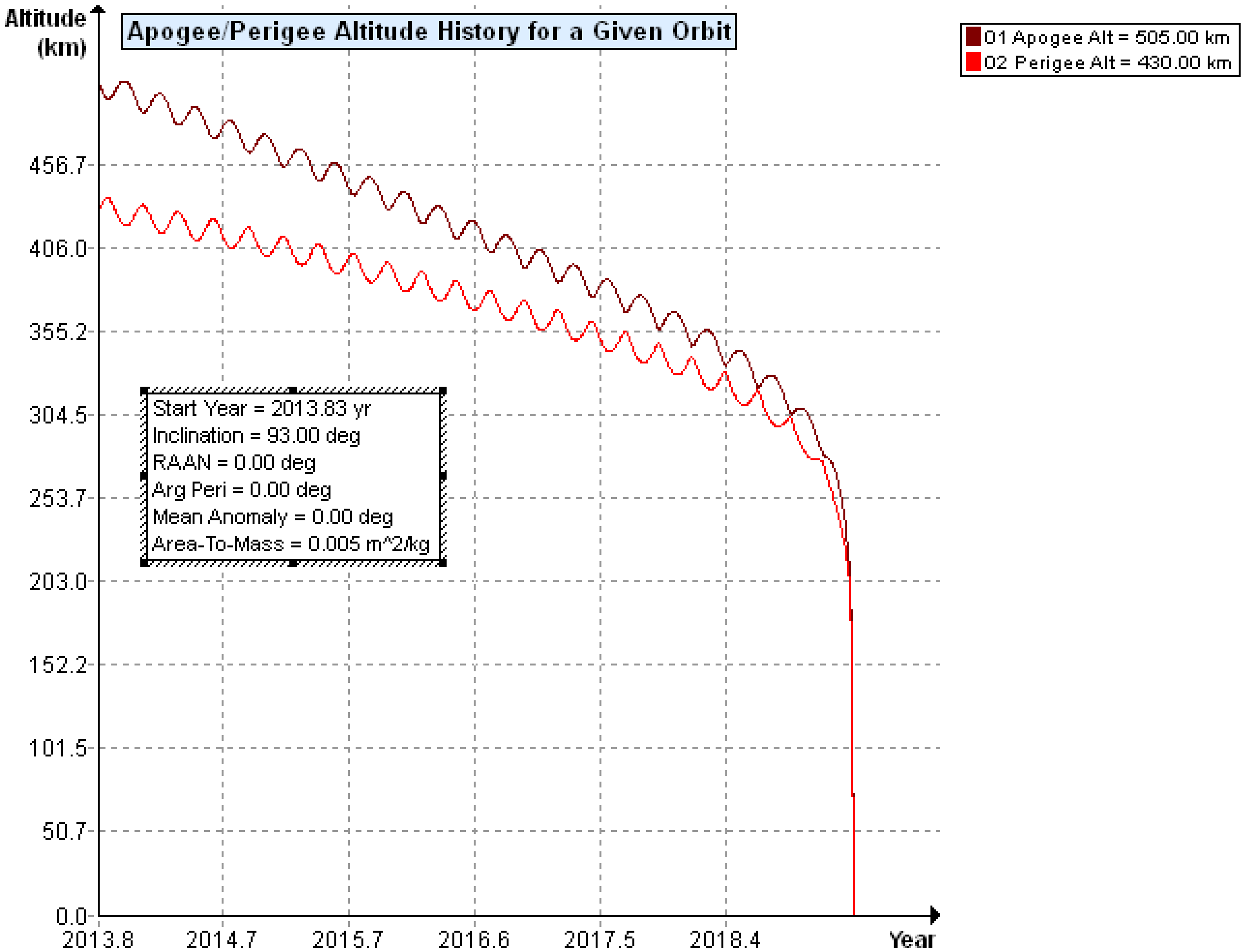

Figure 1, HiakaSat Projected Orbit History.

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

Analysis: Not applicable. HiakaSat orbit is LEO.

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

Analysis: Not applicable. HiakaSat orbit is LEO.

Requirement 4.6-4. Reliability of Postmission Disposal Operations

Analysis: For purposes of performing a margined analysis, we will assume that the operational mission becomes extended up to one (1) year post-launch. The HiakaSat battery passivation will occur at the end of the one year period when commanded by the ground. Reliability of this function is estimated at 99% or greater based on combined reliability of COTS electronics to functionally survive after 1 year of space exposure. Radiation design factor (RDF) is estimated to be very large (greater than 5) using SPENVIS and SHIELDOSE-2 trapped particle, *Brems-stralung*, and Solar Protons, plus estimated GCR. The SPENVIS data is the result of a worst case aluminum sphere shielding assumption around a silicon target. The RDF estimate is based on a generally accepted rule of thumb that COTS electronics can typically withstand 5 kRAD exposures ("Spacecraft Systems Engineering", 3ed; c. John Wiley and Sons, Ltd.; P. Fortescue, J. Stark, and G. Swinerd editors.). To exceed 5 kRAD dose, multiple large Solar Particle Events (SPE's) resulting in >4 kRAD additional dose would have to occur within the 1 year period. This represents more than eight (8) times the predicted solar proton dose for the shielding model analyzed and is therefore a negligible possibility. Accordingly, the reliability of the electronics needed to passivate the batteries is best derived from the generic reliability of COTS electronics after burn-in testing and environmental stress screening tests, rather than from the much lower random probability of a disabling GCR hit. In this regard, any one of three (3) independent critical electronics boards would need to fail to prevent the passivation function from occurring. These circuit boards have been assessed to have generic life failure probability of ~1E-8/hr each. So, for a one year mission, the total failure probability due to component life failure is estimated as:

Pf = (3*1E-8/hr)*(24hr/day)*(365 days/yr)*(1 yr)*100 = 0.026%

That is, there is roughly one (1) chance of functional failure of the battery passivation "disposal" design in 3,805 design-identical missions based on generic design reliability expectations.

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 v2.0.2 reports that HiakaSat is COMPLIANT with demise altitudes above 59 km for all materials. **Total human casualty probability is reported by the DAS software as 1:0, which is interpreted as zero probability (no probability could be calculated given that no impacts are expected).** As seen in the analysis outputs provided in Appendix A below, the impact kinetic energy is 0.0 Joules and Debris Casualty Area is 0.0 m^2.

Requirements 4.7-1b, and 4.7-1c below are non-applicable requirements because HiakaSat 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 HiakaSat mission.

END of ODAR for HiakaSat.

Appendix A: Analysis per DAS v2.0.2

```
04 22 2013; 23:01:59PM  DAS Application Started
04 22 2013; 23:01:59PM  Opened Project C:\Program Files (x86)\NASA\DAS 
2.0\project\HiakaSat\
04 22 2013; 23:02:09PM  Processing Requirement 4.3-1: Return Status :  Not Run

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

=============== End of Requirement 4.3-1 ===============
04 22 2013; 23:02:11PM  Processing Requirement 4.3-2: Return Status : Passed

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

=============== End of Requirement 4.3-2 ===============
04 22 2013; 23:02:15PM  Requirement 4.4-3:  Compliant

=============== End of Requirement 4.4-3 ===============
04 22 2013; 23:02:22PM  Processing Requirement 4.5-1: Return Status :  Passed

==============
Run Data
==============

**INPUT**

      Space Structure Name = HiakaSat
      Space Structure Type = Payload
      Perigee Altitude = 430.000000 (km)
      Apogee Altitude = 505.000000 (km)
      Inclination = 93.000000 (deg)
      RAAN = 0.000000 (deg)
      Argument of Perigee = 0.000000 (deg)
      Mean Anomaly = 0.000000 (deg)
      Final Area-To-Mass Ratio = 0.005100 (m^2/kg)
      Start Year = 2013.830000 (yr)
      Initial Mass = 55.000000 (kg)
      Final Mass = 55.000000 (kg)
      Duration = 7.000000 (yr)
      Station-Kept = False
      Abandoned = True
      PMD Perigee Altitude = -1.000000 (km)
      PMD Apogee Altitude = -1.000000 (km)
      PMD Inclination = 0.000000 (deg)
      PMD RAAN = 0.000000 (deg)
      PMD Argument of Perigee = 0.000000 (deg)
      PMD Mean Anomaly = 0.000000 (deg)
```

```
**OUTPUT**

      Collision Probability = 0.000001
      Returned Error Message: Normal Processing
      Date Range Error Message: Normal Date Range
      Status = Pass

==============

=============== End of Requirement 4.5-1 ===============
04 22 2013; 23:08:43PM  Requirement 4.5-2:  Compliant

==================================================
Spacecraft = HiakaSat
Critical Surface = OBCS_Box
==================================================

**INPUT**

      Apogee Altitude = 505.000000 (km)
      Perigee Altitude = 430.000000 (km)
      Orbital Inclination = 93.000000 (deg)
      RAAN = 0.000000 (deg)
      Argument of Perigee = 0.000000 (deg)
      Mean Anomaly = 0.000000 (deg)
      Final Area-To-Mass = 0.005100 (m^2/kg)
      Initial Mass = 55.000000 (kg)
      Final Mass = 55.000000 (kg)
      Station Kept = No
      Start Year = 2013.830000 (yr)
      Duration = 7.000000 (yr)
      Orientation = Gavity Gradient
      CS Areal Density = 1.353500 (g/cm^2)
      CS Surface Area = 0.018940 (m^2)
      Vector = (0.000000 (u), 1.000000 (v), 1.000000 (w))
      CS Pressurized = No
      Outer Wall 1   Density: 0.541400 (g/cm^2)  Separation: 6.000000 (cm)

**OUTPUT**

      Probabilty of Penitration = 0.000008
      Returned Error Message: Normal Processing
      Date Range Error Message: Normal Date Range

==================================================
Spacecraft = HiakaSat
Critical Surface = Batt_Passivation_box
==================================================

**INPUT**

      Apogee Altitude = 505.000000 (km)
      Perigee Altitude = 430.000000 (km)
```

```
      Orbital Inclination = 93.000000 (deg)
      RAAN = 0.000000 (deg)
      Argument of Perigee = 0.000000 (deg)
      Mean Anomaly = 0.000000 (deg)
      Final Area-To-Mass = 0.005100 (m^2/kg)
      Initial Mass = 55.000000 (kg)
      Final Mass = 55.000000 (kg)
      Station Kept = No
      Start Year = 2013.830000 (yr)
      Duration = 7.000000 (yr)
      Orientation = Gavity Gradient
      CS Areal Density = 1.082800 (g/cm^2)
      CS Surface Area = 0.010000 (m^2)
      Vector = (0.000000 (u), -1.000000 (v), -1.000000 (w))
      CS Pressurized = No
      Outer Wall 1   Density: 0.541400 (g/cm^2)  Separation: 5.000000 (cm)

**OUTPUT**

      Probabilty of Penitration = 0.000008
      Returned Error Message: Normal Processing
      Date Range Error Message: Normal Date Range

==================================================
Spacecraft = HiakaSat
Critical Surface = UHF_or_VHF_Radios
==================================================

**INPUT**

      Apogee Altitude = 505.000000 (km)
      Perigee Altitude = 430.000000 (km)
      Orbital Inclination = 93.000000 (deg)
      RAAN = 0.000000 (deg)
      Argument of Perigee = 0.000000 (deg)
      Mean Anomaly = 0.000000 (deg)
      Final Area-To-Mass = 0.005100 (m^2/kg)
      Initial Mass = 55.000000 (kg)
      Final Mass = 55.000000 (kg)
      Station Kept = No
      Start Year = 2013.830000 (yr)
      Duration = 7.000000 (yr)
      Orientation = Gavity Gradient
      CS Areal Density = 1.101500 (g/cm^2)
      CS Surface Area = 0.004860 (m^2)
      Vector = (0.000000 (u), 1.000000 (v), 1.000000 (w))
      CS Pressurized = No
      Outer Wall 1   Density: 0.541400 (g/cm^2)  Separation: 6.000000 (cm)

**OUTPUT**

      Probabilty of Penitration = 0.000004
      Returned Error Message: Normal Processing
```

```
      Date Range Error Message: Normal Date Range


04 22 2013; 23:09:10PM  Processing Requirement 4.6    Return Status :  Passed

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

**INPUT**

      Space Structure Name = HiakaSat
      Space Structure Type = Payload

      Perigee Altitude = 430.000000 (km)
      Apogee Altitude = 505.000000 (km)
      Inclination = 93.000000 (deg)
      RAAN = 0.000000 (deg)
      Argument of Perigee = 0.000000 (deg)
      Mean Anomaly = 0.000000 (deg)
      Area-To-Mass Ratio = 0.005100 (m^2/kg)
      Start Year = 2013.830000 (yr)
      Initial Mass = 55.000000 (kg)
      Final Mass = 55.000000 (kg)
      Duration = 7.000000 (yr)
      Station Kept = False
      Abandoned = True
      PMD Perigee Altitude = -1.000000 (km)
      PMD Apogee Altitude = -1.000000 (km)
      PMD Inclination = 0.000000 (deg)
      PMD RAAN = 0.000000 (deg)
      PMD Argument of Perigee = 0.000000 (deg)
      PMD Mean Anomaly = 0.000000 (deg)

**OUTPUT**

      Suggested Perigee Altitude = 430.000000 (km)
      Suggested Apogee Altitude = 505.000000 (km)
      Returned Error Message = Reentry during mission (no PMD req.).

      Released Year = 2019 (yr)
      Requirement = 61
      Compliance Status = Pass

==============

=============== End of Requirement 4.6 ===============
04 22 2013; 23:09:43PM  *********Processing Requirement 4.7-1
      Return Status :  Passed

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

```
name = HiakaSat
quantity = 1
parent = 0
materialID = 8
type = Cylinder
Aero Mass = 55.000000
Thermal Mass = 55.000000
Diameter/Width = 0.621000

name = SUCHI_Dry_Air_Vessel
quantity = 1
parent = 1
materialID = 54
type = Cylinder
Aero Mass = 4.157000
Thermal Mass = 0.407000
Diameter/Width = 0.120650
Length = 0.038913

name = SUCHI_Dry_Air_Seal_Flanges
quantity = 2
parent = 2
materialID = 54
type = Cylinder
Aero Mass = 1.875000
Thermal Mass = 1.875000
Diameter/Width = 0.151638
Length = 0.050000

name = SolarPanels_Main
quantity = 8
parent = 1
materialID = 8
type = Flat Plate
Aero Mass = 0.647600
Thermal Mass = 0.647600
Diameter/Width = 0.240000
Length = 0.400000

name = SolarPanels_Zenith
quantity = 4
parent = 1
materialID = 8
type = Flat Plate
Aero Mass = 0.307200
Thermal Mass = 0.307200
Diameter/Width = 0.143000
Length = 0.224200

name = N_and_Z_Decks
quantity = 2
parent = 1
materialID = 8
```

```
type = Flat Plate
Aero Mass = 6.371000
Thermal Mass = 6.371000
Diameter/Width = 0.620000
Length = 0.620000

name = Battery_Box
quantity = 1
parent = 1
materialID = 8
type = Box
Aero Mass = 2.892300
Thermal Mass = 2.892300
Diameter/Width = 0.098400
Length = 0.260000
Height = 0.086000

name = OBCS_Stack_Box
quantity = 1
parent = 1
materialID = 8
type = Box
Aero Mass = 2.098000
Thermal Mass = 2.098000
Diameter/Width = 0.138000
Length = 0.138000
Height = 0.138000

name = SUCHI_and_EPS_Boxes
quantity = 2
parent = 1
materialID = 8
type = Box
Aero Mass = 2.400000
Thermal Mass = 2.400000
Diameter/Width = 0.100000
Length = 0.150000
Height = 0.100000

name = Small_Boxes
quantity = 13
parent = 1
materialID = 8
type = Box
Aero Mass = 0.313600
Thermal Mass = 0.313600
Diameter/Width = 0.055900
Length = 0.092700
Height = 0.025400

name = UCHIS_Assy_Stack
quantity = 1
parent = 1
```

```
materialID = 8
type = Box
Aero Mass = 5.898000
Thermal Mass = 5.898000
Diameter/Width = 0.136000
Length = 0.240000
Height = 0.070000

name = MISC_Struts_and_Brackets
quantity = 40
parent = 1
materialID = 8
type = Flat Plate
Aero Mass = 0.250000
Thermal Mass = 0.250000
Diameter/Width = 0.100000
Length = 0.300000

name = Lightband_Sep_Ring
quantity = 1
parent = 1
materialID = 9
type = Cylinder
Aero Mass = 0.667000
Thermal Mass = 0.667000
Diameter/Width = 0.432000
Length = 0.130000

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

name = HiakaSat
Demise Altitude = 77.999519
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

*************************************
name = SUCHI_Dry_Air_Vessel
Demise Altitude = 71.980410
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

*************************************
name = SUCHI_Dry_Air_Seal_Flanges
Demise Altitude = 61.050796
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

*************************************
name = SolarPanels_Main
Demise Altitude = 76.053668
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
```

```
************************************
name = SolarPanels_Zenith
Demise Altitude = 76.000066
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

************************************
name = N_and_Z_Decks
Demise Altitude = 65.181640
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

************************************
name = Battery_Box
Demise Altitude = 68.753394
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

************************************
name = OBCS_Stack_Box
Demise Altitude = 69.314292
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

************************************
name = SUCHI_and_EPS_Boxes
Demise Altitude = 67.538542
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

************************************
name = Small_Boxes
Demise Altitude = 73.059496
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

************************************
name = UCHIS_Assy_Stack
Demise Altitude = 59.663878
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

************************************
name = MISC_Struts_and_Brackets
Demise Altitude = 76.358558
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

************************************
name = Lightband_Sep_Ring
Demise Altitude = 77.077652
Debris Casualty Area = 0.000000
```

```
Impact Kinetic Energy = 0.000000

************************************

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

Appendix B: Acronyms

ADCS	Attitude Determination and Control Subsystem
Arg peri	Argument of Perigee
CDR	Critical Design Review
cm, cm^2	Centimeter, centimeters squared
COTS	Commercial Off-The-Shelf (items)
DAS	Debris Assessment Software
EOM	End Of Mission
EPS	Electrical Power Subsystem
GEO	Geosynchronous Earth Orbit
HiakaSat	HiakaSat Spacecraft Bus and Payloads
HIP	HSFL Imager Payload experiment
HSFL	Hawaii Space Flight Laboraotry
ITAR	International Traffic In Arms Regulations
kg	kilogram
km	kilometer
LEO	Low Earth Orbit
LiFePO4	Lithium Iron Phospate
LS	HSFL Launch Services
m^2	Meters squared
mm	millimeter
MRR	Mission Readiness Review
N/A	Not Applicable.
NASA	National Aeronautics and Space Administration
OBCS	On Board Computer System
ODAR	Orbital Debris Assessment Report
ODPO	Orbital Debris Program Office
ORR	Operations Readiness Review
ORS	Operationally Responsive Space
OSMA	(NASA) Office of Safety and Mission Assurance
PDR	Preliminary Design Review
PMRF	Pacific Missile Range Facility
PSIa	Pounds Per Square Inch, absolute
PSRR	Pre-Ship Readiness Review
RAAN	Right Ascension of the Ascending Node
SC	Spacecraft Bus and Payloads
SIP	Separation Imager Payload
SMA	Safety and Mission Assurance
SPARK	Spaceborne Payload Assist Rocket Kauai

Telecom	Telecommunications Subsystem
SUCHI	Space Ultra-Compact Hyperspectral Imager
UH	University of Hawaii
UH Manoa	University of Hawaii, Manoa Campus
USAF	United States Air Force
UTJ	Ultra Triple Junction (solar cell)
yr	year