CESIUM

Cesium Mission 1 FCC 442
Orbital Debris Assessment Report

FRN: 0029768140

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Contents

ODAR Signature Approval

Title	Name	Signature	Date
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ODAR Self-Assessment per NASA-STD-8719.14B, Process for Limiting Orbital Debris

Table 1 presents the Cesium Mission 1 FCC Orbital Debris Self-Assessment Report Evaluation in accordance with the assessment format provided in Appendix A.2 of NASA-STD-8719.14B. Requirement assessments for 4.3-1/2, 4.5-1/2, 4.6-1 to 4.6-3, and 4.7-1 were analyzed using **NASA's DAS v3.1.0** software.

Table 1: Cesium Mission 1 FCC ODAR self-assessment evaluation per NASA-STD-8719.14B

Reqm't #	Launch Vehicle Compliant / Not Compliant / Std. Non-Compliant / Incomplete	Spacecraft Compliant / Not Compliant / Incomplete	Comments For all incompletes, include risk assessment of non-compliance & Tracking #
4.3-1. a 25-year limit	N/A	Compliant	NASA DAS, no intentional release of debris
4.3-1. b <100 object x year limit	N/A	Compliant	NASA DAS, no intentional release of debris
4.3-2 GEO +/- 200km	N/A	N/A	NASA DAS, LEO orbit
4.4-1 <0.001 Explosive Risk	N/A	Compliant	
4.4-2 Passivate Energy Source	N/A	N/A	
4.4-3 Limit BU Long term Risk	N/A	N/A	
4.4-4 Limit BU Short term Risk	N/A	Compliant	
4.5-1 <0.001 10 cm Impact Risk	N/A	Compliant	NASA DAS v3.1.0
4.5-2 Post mission Disposal Risk	N/A	Compliant	NASA DAS v3.1.0
4.6-1a-c Disposal Method	N/A	Compliant	NASA DAS v3.1.0
4.6-2 GEO Disposal	N/A	N/A	
4.6-3 MEO Disposal	N/A	N/A	
4.6-4 Disposal Reliability	N/A	Compliant	
4.7-1 Ground Population Risk	N/A	Compliant	NASA DAS v3.1.0
4.8-1 Tether Risk	N/A	N/A	Neither CS1 nor CS2 utilize a tether

ODAR Section 1: Program Management and Mission Overview

Purpose: This report is intended to communicate to the FCC Cesium Mission 1's probability of generating orbital debris per NASA-STD-8719.14B guidelines to satisfy the requirements concerning the post-mission disposal of commission-licensed satellites.

Program Management: CesiumAstro, Inc.

Clear schedule of mission design and development milestones:

Mission Selection: 01/01/2019

Mission Preliminary Design Review: 10/30/2020

Mission Critical Design Review: 03/01/2021

Pre-Ship Readiness Review: 06/01/2021

Launch: 09/16/2021

Mission Overview: Cesium Mission 1 (CM1) is CesiumAstro's first mission to launch and operate our hardware in space. This is done through the use of two spacecraft, Cesium Satellite 1 (CS1) and Cesium Satellite 2 (CS2). CM1 provides an on-orbit platform for customer experiments that push the boundaries of small satellite communication. In addition, it gives Cesium the capability to demonstrate advanced features of our technology such as dynamic waveform switching and dynamic link optimization. With the launch of Mission 1, Cesium will have a complete commercial phased array communication system in LEO.

ODAR Summary: No debris released in normal operations; no credible scenario for breakups; the collision probability with other objects is compliant with NASA standards; and the estimated nominal decay lifetime due to atmospheric drag is well under 25 years following operations (< 11 years, as calculated by **DAS 3.1.0**).

Launch Vehicle: Atlas V 401

Launch Site: Vandenberg AFB, CA.

Proposed Launch Date: September 16, 2021

Mission Duration: 5 years

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

CS1 and CS2 (Cesium Spacecraft) will deploy from an Atlas V 401 upper stage into a sun-synchronous orbit (SSO) from which they will naturally decay due to atmospheric drag. The nominal deployment altitude is 550 km.

- Nominal Insertion Case: Apogee: 550 km, Perigee: 550 km
- Inclination: ~97.6º
- Local Time: 10:30 AM

Between payload experiment operations, the electric thrusters will be used to move Cesium Spacecraft to ~96.6º to allow Earth's J_2 effect to rotate the orbit's RAAN relative to SSO. Once the desired MLT is obtained, a second maneuver will be used to return the spacecraft to SSO (6:00 AM dawn-dusk), where Cesium Spacecraft will remain for the duration of the mission.

Reason for selection of orbit: The dawn-dusk SSO orbit was chosen for the ability to track the sun and ground simultaneously during operations.

Identification of any interaction or potential physical interference with other operational spacecraft: There are no planned physical interactions with other operational spacecraft.

ODAR Section 2: Spacecraft Description

Physical description of the spacecraft:

CS1 and CS2 are 6U CubeSats, each with a maximum launch mass of 14 kg. Basic physical dimensions not including the deployable solar panels are 222 mm × 106 mm × 365 mm. The structure is mainly comprised of aluminum, with internal rails along four corner edges. The solar arrays are deployable and hinge at the +Z face of the spacecraft. Physical dimensions of the solar arrays at their maximums are presented in Table 2. When deployed, the solar panels are parallel to the +Z face of the spacecraft. Power storage is provided by two 18650 battery modules which are recharged by the solar panels. A render of the spacecraft is depicted in Figure 1.

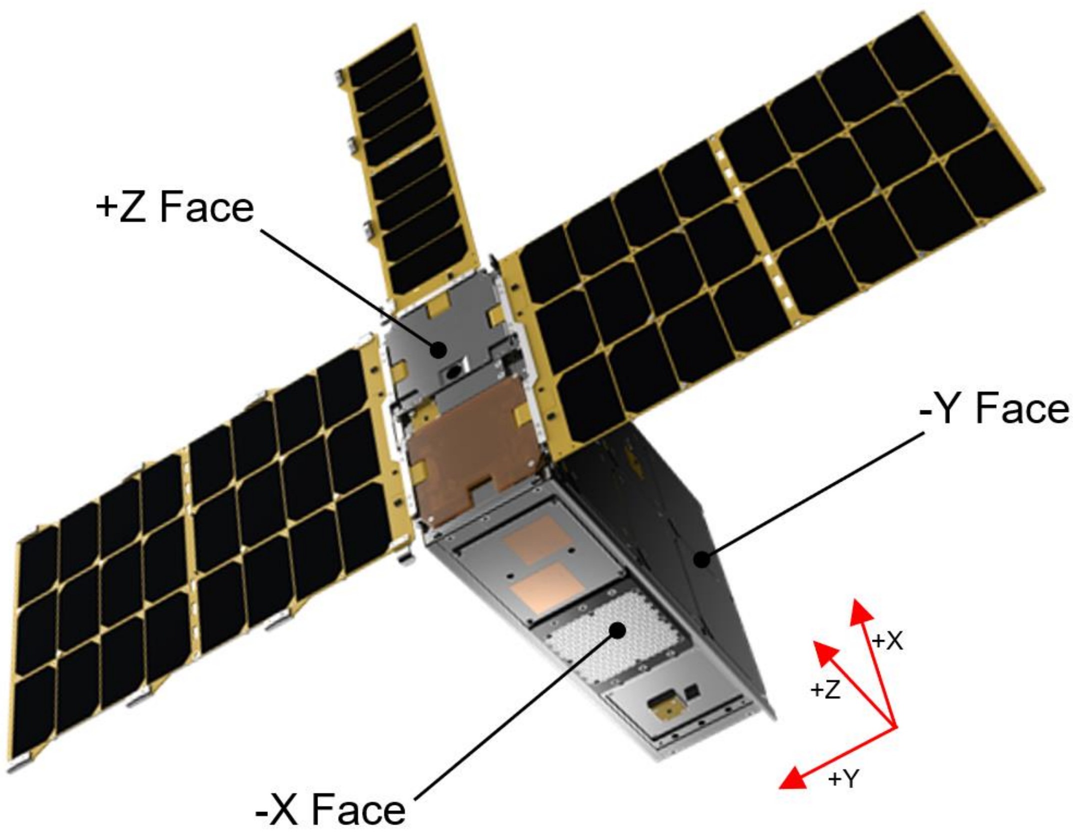

Figure 1: Cesium 6U spacecraft with solar panels deployed

Table 2: Cesium 6U spacecraft solar array dimensions

Solar Array	Dimensions
+X Solar Array	354.2 mm × 88.3 mm × 8.0 mm
+Y Solar Array	354.2 mm × 232.4 mm × 12.1 mm
-Y Solar Array	354.2 mm × 213.9 mm × 3.1 mm

Total spacecraft mass at launch, including all propellants and fluids: 14 kg

Dry mass of spacecraft at launch (minus all consumables and propellants): ~13.77 kg

Identification, including type, mass, and pressure, of all fluids planned to be on board: The battery modules (BM) are unpressurized standard COTS lithium-ion cells. The field-emission electric propulsion (FEEP) propellant is solid indium until it is heated to be used.

Description of all propulsion systems: Cesium Spacecraft will each utilize a FEEP thruster with solid indium propellant. There are no moving parts, and the propellant is in its solid state at room temperature.

Description of all active and passive attitude control systems with indications of the normal attitude of the spacecraft with respect to the velocity vector: Cesium Spacecraft maneuver capabilities utilize an ADCS system which features: 3 modular reaction wheels (RW), 3-Axis magnetometer, a star tracker, 6 coarse sun sensor inputs, 3 electromagnet torque rods, 3-axis MEMS accelerometer, 3-axis MEMS gyroscope, and a fully programmed ADCS computer providing each Cesium Spacecraft with turnkey attitude control capabilities and < 0.1° pointing accuracy. The primary attitude control modes for CM1 comprise of Detumble, Slew, Stabilization, and Pointing.

The Detumble mode will be used after ejection from the launch vehicle, where Cesium Spacecraft will slow themselves to a controlled rotation using a B-dot control law, magnetometer (used to sense the rate of change in Earth's magnetic field), and the electromagnet torque rods to damp the Cesium Spacecraft angular rotation. Once the desired rotation rate is achieved, the system will then transfer to a slew mode to obtain a nadir pointing orientation and link to the ground station (Detumble & Connect operational Mode). During slew maneuvers, the primary actuators used are the 3 reaction wheels to orient Cesium Spacecraft and the torque rods to dump excess momentum continuously. Refer to Figure 2 for an illustration of Cesium Spacecraft with attitude orientations specified.

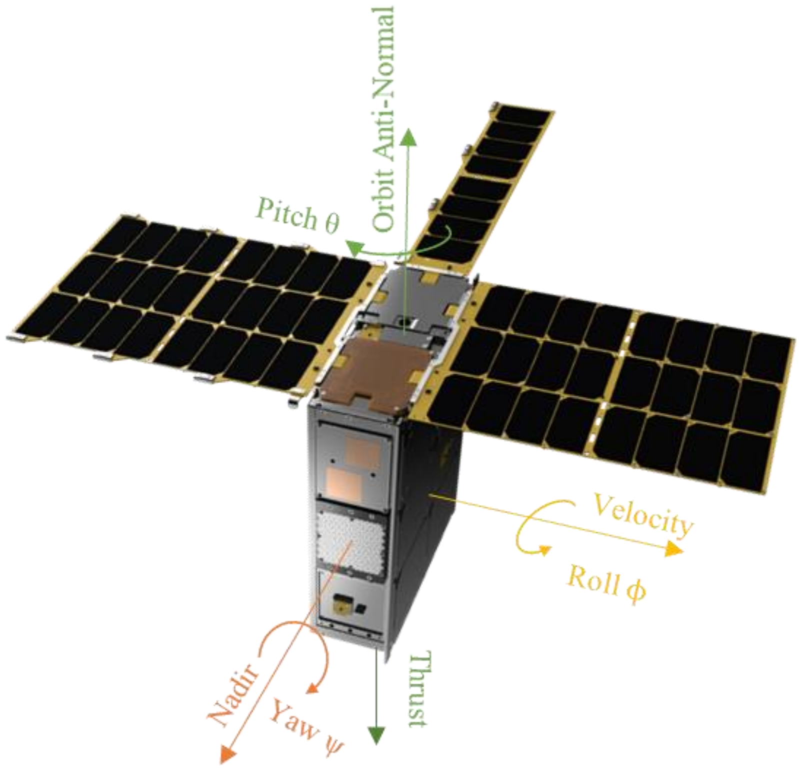

Figure 2: Cesium Spacecraft in normal operation mode

Sun pointing mode provides nadir pointing for the primary axis and rotates about the nadir axes to optimize solar pointing for the secondary axis. Nadir pointing is obtained through magnetic momentum management, sun sensor/magnetometer or star tracker attitude determination, and reaction wheel torque commands computed by a quaternion feedback control law.

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

Description of the electrical generation and storage system: The solar array system is used to collect solar energy from the Sun. It is deployed by an internal pin puller driver. The solar arrays can generate up to 56 W instantaneous power at the beginning of the mission.

Cesium Spacecraft Electrical Power System Module (EPSM) accepts input power from the solar panels and arrays, batteries, and external sources and converts it to output power at various voltage levels. Each BM is comprised of eight 18650 lithium-ion cells that provide electrical energy during the mission. BM electronics provide battery inhibits, first and second-level battery safeties (OV, UV, OC, OT, individual cell over-voltage, and others), a battery heater, and a “gas gauge” to provide up-to-date state-of-charge information on the batteries. The battery cells are recharged by the Electrical Power System (EPS).

Identification of any other sources of stored energy not noted above: N/A

Identification of any radioactive materials on board or make an explicit statement that there are no radioactive materials onboard: CM1 does not have any radioactive materials on board.

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

Potential causes of spacecraft breakup during deployment and mission operations: There are no plausible scenarios that would result in the spacecraft breakup during normal deployments of mission operations. No part of the spacecraft or solar panel deployment will generate debris in nominal operations. In nominal disposal operations, the spacecraft will burn up in the atmosphere. The most significant yet unlikely "breakup" may be caused by a battery explosion, which would still not generate any debris outside of the spacecraft. The possibility of this is very small, as can be seen in the Failure Mode and Effect Analysis (FMEA) below.

Summary of failure modes and effects analyses of all credible failure modes which may lead to an accidental explosion: A potential failure mode which may lead to an accidental explosion can stem from BM safety protections failure which would leave the batteries imperiled.

Detailed plan for any designed spacecraft breakup, including explosions and intentional collisions: CM1 has no plans for any intentional collisions and/or explosions.

List of components which are passivated at End of Mission (EOM) including method of passivation and amount which cannot be passivated: No components will be passivated at EOM.

Rationale for all items which are required to be passivated but cannot be due to their design: The lithium-ion batteries will not be passivated due to the negligible risk of impact/explosion. The low charges and small battery cells on Cesium Spacecraft power systems deem passivation at EOM redundant.

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

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:

Compliance statement:

Required Probability: 0.001.

Projected Probability: 0.000.

Supporting Rationale and FMEA details: Battery explosion or failure:

Effect: In the unlikely event that any of the Failure modes listed below were to occur, then the possibility of a battery explosion is present. If a battery cell does rupture, the small size, mass, and potential energy of the COTS batteries, most if not, all the debris should be contained within the vessel due to the lack of penetration energy.

Probability: Extremely Low, less than ~ 0.1%. The BM satisfies NASA flight safety program and JSC EP-WI-032 for use on the ISS and features over-voltage, over-current, under-voltage, and over-temperature protection.

Failure Mode 1: Short circuit

Mitigation 1: Primary battery protections that include: Cell over/under-voltage (OV/UV) protection that inhibits charging/discharging when the cells are in an over or under-voltage condition, Over-current (OC) charge/discharge protection that inhibits predefined charge and discharge limits.

Mitigation 2: Second level battery protection: If/when an over-voltage condition is detected on a cell, in addition to the main circuitry disconnecting the charge, discharge, and/or trickle-charge MOSFETs, the second-level protection discharges the battery pack through a 75 Ω power resistor. This 2 – 4 W discharge is intended to gradually reduce the affected cell voltage(s) while the cell over-voltage condition exists.

Mitigation 3: BM went through an internal short circuit (cells) testing with upper-temperature limits of +55 °C and lower limits at -15 °C and passed.

Failure Mode 2: Internal thermal rise due to orbital environment and/or high discharge rate

Mitigation 1: Each cell is individually wrapped with a conformal Kapton heater and a thermistor. The individual heaters combine to form a parallel combination of resistive heaters that are driven by the battery's positive and negative voltage terminals.

Mitigation 2: Over-temperature (OT) charge/discharge protection where charging/discharging inhibit are present in the event the cell temperature exceeds the predefined limits. Additionally, two heater controllers are present, one in the protection circuitry, and one in the SupMCU.

Mitigation 3: Second-level battery protection includes cell balancing where the BM automatically balances the cell voltages during/near the end of charging to maintain the overall cell balance to the pre-determined maximum cell voltage difference.

Failure Mode 3: Inoperative/defect in venting

Mitigation 1: Vent design in cells is constructed so that they relieve excessive internal pressure at a value and rate that will preclude rupture, explosion, and self-ignition. When evaluated, the BM passed per IEC 6213:2012 standard.

Failure Mode 4: Battery crush

Mitigation 1: The BM has passed crush testing, where crushing force was released upon with a maximum force of 13 kN ±1 kN has been applied resulting in no fire or explosion.

4.4-2, Design for passivation after completion of mission operation while in orbit about Earth or the Moon: CM1 does not intend on passivating any components after completion of the mission while in orbit about the Earth or Moon.

4.4-3, Limiting the long-term risk to other spaces systems from planned breakups: CM1 does not have any planned breakups, thus, requirement 4.4-3 is not applicable.

4.4-4, Limiting the short-term risk to other space systems from planned breakups: CM1 does not have any planned breakups, thus, requirement 4.4-4 is not applicable.

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 NASA DAS v3.1.0:

4.5-1, Limiting debris generated by collisions with large objects when operating in Earth orbit: Cesium Spacecraft collision probability per **DAS v3.1.0** is 1.7559 x 10^{-06}.

4.5-2: Limiting debris generated by collisions with smalls object when operating in Earth or Lunar orbit: Cesium Spacecraft collision probability per **DAS v3.1.0** is as follows:

Critical Surface: Thruster has a probability of penetration of 2.0883 x 10^{-06}.

The critical surface meets the requirement of < 0.001 probability of penetration for large and small orbital debris (Requirement 4.5-1 and 4.5-2), as can be seen in Figure 3.

```
10 19 2020; 16:36:34PM    Processing Requirement 4.5-1:     Return Status :  Passed

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

**INPUT**

        Space Structure Name = Nightingale
        Space Structure Type = Payload
        Perigee Altitude = 550.000 (km)
        Apogee Altitude = 550.000 (km)
        Inclination = 97.600 (deg)
        RAAN = 0.000 (deg)
        Argument of Perigee = 0.000 (deg)
        Mean Anomaly = 0.000 (deg)
        Final Area-To-Mass Ratio = 0.0082 (m^2/kg)
        Start Year = 2021.000 (yr)
        Initial Mass = 14.000 (kg)
        Final Mass = 13.770 (kg)
        Duration = 5.000 (yr)
        Station-Kept = False
        PMD Perigee Altitude = 300.000 (km)
        PMD Apogee Altitude = 300.000 (km)
        PMD Inclination = 97.600 (deg)
        PMD RAAN = 0.000 (deg)
        PMD Argument of Perigee = 0.000 (deg)
        PMD Mean Anomaly = 0.000 (deg)

**OUTPUT**

        Collision Probability = 1.7559E-06
        Returned Message: Normal Processing
        Date Range Message: Normal Date Range
        Status = Pass

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

=============== End of Requirement 4.5-1 ===============

10 19 2020; 16:52:15PM    Requirement 4.5-2:  Compliant

=================================================
Spacecraft = Nightingale
Critical Surface = Nano Thruster
=================================================

**INPUT**

        Apogee Altitude = 550.000 (km)
        Perigee Altitude = 550.000 (km)
        Orbital Inclination = 97.600 (deg)
        RAAN = 0.000 (deg)
        Argument of Perigee = 0.000 (deg)
        Mean Anomaly = 0.000 (deg)
        Final Area-To-Mass = 0.0082 (m^2/kg)
        Initial Mass = 13.770 (kg)
        Final Mass = 13.770 (kg)
        Station Kept = No
        Start Year = 2021.000 (yr)
        Duration = 5.000 (yr)
        Orientation = Fixed Oriented
        CS Areal Density = 1.299 (g/cm^2)
        CS Surface Area = 0.0098 (m^2)
        Vector = (-0.186447 (u), 0.635185 (v), 0.749518 (w))
        CS Pressurized = No
        Outer Wall 1   Density: 10.911 (g/cm^2)  Separation: 15.000 (cm)

**OUTPUT**

        Probability of Penetration = 2.0883E-06 (2.0883E-06)
        Returned Error Message: Normal Processing
        Date Range Error Message: Normal Date Range

=============== End of Requirement 4.5-2 ===============
```

*Figure 3: **NASA DAS v3.1.0** Requirement 4.5 Assessment for Cesium Spacecraft, area-to-mass ratio of 0.008173 m^2/kg*

ODAR Section 6: Assessment of Spacecraft Post Mission Disposal Plan and Procedures

Description of spacecraft disposal options selected & preliminary plan for controlled re-entry: At CM1 EOL, Cesium Spacecraft will de-orbit by atmospheric drag within the 25-year requirement.

Assessment of spacecraft compliance with Requirements 4.6-1 through 4.6-4:

4.6-1, Disposal for space structures in or passing through LEO: Cesium Spacecraft meets all LEO reentry orbit criteria per **NASA DAS v3.1.0** as seen in **Error! Reference source not found.**.

```
04 14 2021; 10:58:32AM     Processing Requirement 4.6          Return Status :  Passed

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

          **INPUT**

                    Space Structure Name = Nightingale
                    Space Structure Type = Payload

                    Perigee Altitude = 550.000000 (km)
                    Apogee Altitude = 550.000000 (km)
                    Inclination = 97.600000 (deg)
                    RAAN = 0.000000 (deg)
                    Argument of Perigee = 0.000000 (deg)
                    Mean Anomaly = 0.000000 (deg)
                    Area-To-Mass Ratio = 0.008173 (m^2/kg)
                    Start Year = 2021.000000 (yr)
                    Initial Mass = 14.000000 (kg)
                    Final Mass =13.770000 (kg)
                    Duration = 5.000000 (yr)
                    Station Kept = False
                    Abandoned = True
                    PMD Perigee Altitude = 511.487960 (km)
                    PMD Apogee Altitude = 532.611225 (km)
                    PMD Inclination = 97.659921 (deg)
                    PMD RAAN = 11.473186 (deg)
                    PMD Argument of Perigee = 127.274029 (deg)
                    PMD Mean Anomaly = 0.000000 (deg)

          **OUTPUT**

                    Suggested Perigee Altitude = 511.487960 (km)
                    Suggested Apogee Altitude = 532.611225 (km)
                    Returned Error Message = Passes LEO reentry orbit criteria.

                    Released Year = 2034 (yr)
                    Requirement = 61
                    Compliance Status = Pass

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

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

*Figure 4: **NASA DAS v3.1.0 Requirement** 4.6 assessment for Cesium Spacecraft Area-to-Mass ratio of 0.008173 m^2/kg*

Moreover, as seen in Figure 5, using the **DAS v3.1.0** Science and Engineering, *Utility for Orbit Lifetime/Dwell Time*, Cesium Spacecraft have an orbital lifetime of 10.190281 years, which **satisfies the 25-year requirement.**

```
10 19 2020; 17:01:28PM Science and Engineering - Orbit Lifetime/Dwell
Time

**INPUT**

      Start Year = 2026.000000 (yr)
      Perigee Altitude = 550.000000 (km)
      Apogee Altitude = 550.000000 (km)
      Inclination = 97.600000 (deg)
      RAAN = 0.000000 (deg)
      Argument of Perigee = 0.000000 (deg)
      Area-To-Mass Ratio = 0.008173 (m^2/kg)

**OUTPUT**

      Orbital Lifetime from Startyr = 10.190281 (yr)
      Time Spent in LEO during Lifetime  = 10.190281 (yr)
      Last year of Propagation = 2036 (yr)
      Returned Error Message: Object reentered
10 19 2020; 17:01:41PM Science and Engineering - Orbit Lifetime/Dwell
Time
```

*Figure 5: **DAS v3.1.0** Cesium Spacecraft Orbital lifetime and Dwell time spent in LEO after EOL*

4.6-4, Reliability of post-mission disposal maneuver operations in Earth orbit: Cesium Spacecraft will not perform a post-mission disposal maneuver. Instead, will deorbit using atmospheric drag.

ODAR Section 7: Assessment of Spacecraft Reentry Hazards

Detailed description of spacecraft components by size, mass, material, shape, and original location on the space vehicle if the atmospheric reentry option is selected: NASA DAS v3.1.0 software analysis for Requirement 4.7-1 confirms all of the subcomponents of Cesium Spacecraft will demise before reaching the Earth's surface without causing any debris casualty area, except for the P5 STD Pin Puller which shows a debris casualty area of 0.399108 m^2 and an impact kinetic energy of 0.905823 J which **satisfies the 15 J requirement**.

Assessment of spacecraft compliance with Requirement 4.7-1:

4.7-1, Assessment of Debris Surviving Atmospheric Reentry: Per **NASA DAS v3.1.0** Cesium Spacecraft risk of human casualties is 1:100000000 with an impact kinetic energy less than 15 J, which satisfies Requirement 4.7-1.

ODAR Section 7A: Assessment of Spacecraft Hazardous Materials

Summary of the hazardous materials contained on the spacecraft using all columns and the format in paragraph 4.7.4.10.: There are no hazardous materials onboard Cesium Spacecraft.

Abbreviations & Acronyms

6U	6-Unit
ADCS	Attitude Determination Control System
BM	Battery Module
CM1	Cesium Mission 1
COTS	Commercial Off-The-Shelf
CS1	Cesium Satellite 1
CS2	Cesium Satellite 2
EOL	End of Life
EOM	End of Mission
EPS	Electrical Power System
EPSM	Electrical Power System Module
FCC	Federal Communications Commission
FEEP	Field-Emission Electric Propulsion
FMEA	Failure Mode and Effects Analysis
LEO	Low Earth Orbit
MEMS	Micro-Electromechanical System
MLT	Mean Local Time
MOSFET	Metal-Oxide-Semiconductor Field-Effect Transistor
NASA	National Aeronautics and Space Administration
OC	Over-Current
ODAR	Orbital Debris Assessment Report
OT	Over-Temperature
OV	Over-Voltage
RAAN	Right Ascension of the Ascending Node
RW	Reaction Wheel
SSO	Sun Synchronous Orbit
UV	Under-Voltage