3B5GSAT Orbital Debris Assessment Report (ODAR)

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This report is presented in compliance with NASA-STD-8719.14, APPENDIX A.

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DAS Software Version Used In Analysis: DAS 3.1.2.0

Version approved by:

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Record of Revisions

Revision	Date	Affected Pages	Description of Change	Authors
1.0	27 AUG 2021	All	Document creation	Alexis Martin

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Self-assessment of the ODAR

This ODAR follows the format recommended in NASA-STD-8719.14, Appendix A.1, sections 1 through 8 for the 3B5GSAT. Sections 9 through 14 apply to the launch vehicle ODAR and are not covered here.

Orbital Debris Self-Assessment Report Evaluation: Sateliot Mission

Table 1: ODAR Review Check Table

Reqm #	Launch Vehicle				Spacecraft			Comments
	Compliant	Not Compliant	Incomplete	Standard Not Compliant	Compliant or N/A	Not Compliant	Incomplete	
4.3-1.a	☐	☐	☐	☐	☒	☐	☐	
4.3-1.b	☐	☐	☐	☐	☒	☐	☐	
4.3-2	☐	☐	☐	☐	☒	☐	☐	
4.4-1	☐	☐	☐	☐	☒	☐	☐	
4.4-2	☐	☐	☐	☐	☒	☐	☐	
4.4-3	☐	☐	☐	☐	☒	☐	☐	
4.4-4	☐	☐	☐	☐	☒	☐	☐	
4.5-1	☐	☐	☐	☐	☒	☐	☐	
4.5-2	☐	☐	☐	☐	☒	☐	☐	
4.6-1.a	☐	☐	☐	☐	☒	☐	☐	
4.6-1.b	☐	☐	☐	☐	☒	☐	☐	
4.6-1.c	☐	☐	☐	☐	☒	☐	☐	
4.6-2	☐	☐	☐	☐	☒	☐	☐	
4.6-3	☐	☐	☐	☐	☒	☐	☐	
4.6-4	☐	☐	☐	☐	☒	☐	☐	
4.7-1	☐	☐	☐	☐	☒	☐	☐	
4.8-1	☐	☐	☐	☐	☒	☐	☐	

1. Mission Overview

Project Manager: Vladimir Zaitsev

Mission Overview: 3B5GSAT was launched on 22 March 2021, on a Sun Synchronous Orbit (97.4 degree inclination) at an altitude between 540-570 km.

ODAR Summary: All the debris generated in orbit are compliant with Requirements 4.3, there is no credible scenario for breakups, the collision probability with other objects is compliant with NASA standards, the estimated nominal decay lifetime due to atmospheric drag is 3.718 years (as calculated by DAS 3.1.2.0).

Launch: 3B5GSAT was launched on a Soyuz 2-1a Fregat from Baikonour on 22 March 2021.

Mission Duration: The nominal mission duration is 12 months. 3 years mission duration is the baseline for the spacecraft architecture, specifically affecting:

- Orbital decay and link time decay, mainly due to atmospheric drag.
- Degradation of battery cell capacity, mainly due to the number and characteristics of charge/discharge cycles.
- Degradation of solar cell power, mainly due to surface degradation by UV and atomic oxygen

Orbit Profile: The 3B5GSAT satellite was be deployed from the Russian 12U deployer developed for GK Launch Services by Capital Aerospace into a Sun synchronous orbit from which it will naturally decay due to atmospheric drag. The 3B5GSAT satellite has no propulsion and therefore does not actively change orbits.

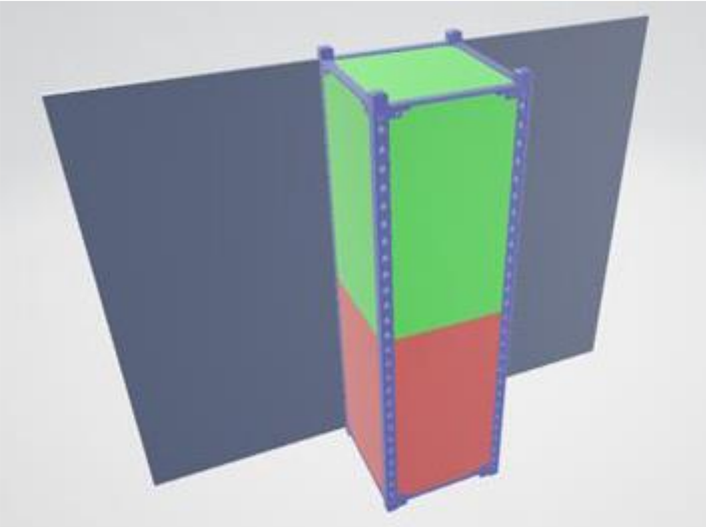
2. Spacecraft Description

Physical Description of the Spacecraft:

The main systems of the spacecraft apart from the payload are :

- OBDH: On-board Data Handling system
- ADCS: Attitude Determination and Control System
- EPS: Electric Power System
- COMMS: Telecommunications system

A 3U CubeSat configuration with double deployable solar panels (see model in Figure 1) is considered with a mass of 4.2kg:



Hereunder are the ranges of the cross-sectional area considered for the analysis:

- 3U with double deployable solar panels: [0.01, 0.15] m^2

Total satellite mass at launch, including all propellants and fluid: 4.2 kg

Dry mass of satellite at launch, excluding solid rocket motor propellants: 4.2kg

Description of all propulsion systems (cold gas, mono-propellant, bi-propellant, electric, nuclear):

The satellite is not equipped with any propulsion systems. Attitude control is performed by means of reaction wheels and magnetorquers.

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 satellite dos not embark any fluids.

Fluids in Pressurized Batteries:

None. The energy storage is implemented via unpressurised standard Lithium-Ion rechargeable battery cells with a total nominal capacity of 45.6 Wh. The EPS converts and distributes the incoming and stored power into four low voltage rails (two 5V and two 3.3V with an absolute maximum power rating of 27.5W and 18.2W respectively) and two high power rails on battery voltage. The distribution is done using controllable switches with latch-up protection.

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

The ADCS system handles all necessary sensors and actuators to provide attitude knowledge and control in all three axes. The gyros, the torquers and the reaction wheels are integrated in the PC104 stack. Magnetometer, redundant magnetometer, coarse and fine sun sensors are placed on different locations in the spacecraft.

There are 8 coarse sun sensors. They are small photo-diodes which are glued on each face of the spacecraft, plus on the deployable panels to cover the stowed configuration. One fine sun sensor sits on the main sun facing solar panel, the other one on the zenith closure panel.

De-tumble mode stabilises the spacecraft after orbit injection, using magnetorquers. In case of battery depletion before reaching a stable attitude, the mode can be paused and resumed after the batteries have recharged.

The main modes during routine operations are “Nadir/Sun” and “Nadir/Ground Station”. They keep the payload antenna nadir pointing, whilst rotating the spacecraft around the nadir axis.

“Nadir/Sun” assures that the solar panels are illuminated as good as possible and “Nadir/Ground Station” increments link budget margin and the average time pass.

Slew manoeuvres and compensation of short term disturbances is done with the reaction wheels. To prevent them from reaching their maximum speed, they are offloaded regularly using the magnetorquers. Offloading manoeuvres do not affect the pointing capabilities of the system.

3. Spacecraft Debris Released during Normal Operations

Requirement 4.3-1: Debris passing through LEO, released debris with diameters of 1mm or larger

No release of debris will occur during the lifetime of the 3B5GSAT satellite. All deployments use a frangibolt and motor-based systems that do not generate any debris. Additionally, there is no probable scenario for unintentional debris generation.

Result for Requirement 4.3-1: COMPLIANT

Requirement 4.3-2: Debris passing near GEO

There will be no intentional release of debris during the lifetime of the mission, as Sateliot's mission is contained in Low Earth Orbit.

Result for Requirement 4.3-2: COMPLIANT

4. Spacecraft Intentional Breakups and Potential for Explosions

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

No risks identified other than a malfunction in the EPS that could lead to short circuits and overheats with, in the worst case, the risk of an explosion of the batteries. With that regard, to prevent any malfunction, both current and temperature are monitored in and out of each of the battery modules at all time. Loads will be disconnected from the battery module in case a threshold current or temperature is reached with both over and under voltage protections.

All failure modes below might theoretically 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 the selected 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: Extremely Low. It is believed to be a much less than 0.1% probability that multiple independent (not common mode) faults must occur for each failure mode to cause the ultimate effect (explosion).

Failure mode 1: Internal short circuit

Mitigation 1: Qualification and acceptance shock, vibration, thermal cycling, and vacuum tests followed by maximum system rate-limited charge and discharge will 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.

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

Mitigation 2: Cells will be tested in lab for high load discharge rates in a variety of flight like configurations to determine if the feasibility of an out of control thermal rise in the cell. Cells will also be tested in a hot environment to test the upper limit of the cells capability.

Combined faults required for realized failure: Spacecraft thermal design must be incorrect AND external over current detection and disconnect function must fail to enable this failure mode.

Failure mode 3: 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 3: This failure mode will be negated by a) qualification tested short circuit protection on each external circuit, b) design of battery packs and insulators such that no contact with nearby board traces is possible without being caused by some other mechanical failure, c) obviation of such other mechanical failures by proto-qualification and acceptance environmental tests (shock, vibration, thermal cycling, and thermal-vacuum tests).

Combined faults required for realized failure: An external load must fail/short-circuit AND external over-current detection and disconnect function must all occur to enable this failure mode.

Failure mode 4: Inoperable vents.

Mitigation 4: Battery vents are not inhibited by the battery holder design or the spacecraft.

Combined faults required for realized failure: The manufacturer fails to install proper venting.

Failure mode 5: Crushing.

Mitigation 5: 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 6: Low level current leakage or short-circuit through battery pack case or due to moisture-based degradation of insulators.

Mitigation 6: 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 7: Excess temperatures due to orbital environment and high discharge combined.
Mitigation 7: The spacecraft thermal design will negate this possibility. Thermal rise will be 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 over-current monitoring and control must all fail for this failure mode to occur.

Result for Requirement 4.4-1: COMPLIANT

Requirement 4.4-2: Design for passivation after completion of mission operations while in orbit about Earth or the Moon

At the end of the mission the spacecraft will be commanded to a dormant mode in which all subsystems will be powered off, other than the EPS and the COMMS system, which by design cannot be disconnected to avoid single points of failure.

Any rotating or movable part shall be fixed and any relative movements shall be blocked at the end of operations and before re-entry.

The COMMS transmitting capabilities will be inhibited and the system will be deemed passivated.

Result for Requirement 4.4-2: COMPLIANT

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

There are no planned breakup during the mission.

Result for Requirement 4.4-3: COMPLIANT

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

There are no planned breakup during the mission.

Result for Requirement 4.4-4: COMPLIANT

5. Spacecraft Potential for On-Orbit Collisions

Requirement 4.5-1. Limiting debris generated by collisions with large objects when operating in Earth orbit

The worst case initial orbit of 3B5GSAT is a circular orbit at an altitude of 550 km and an inclination of 97.4 degrees. The following mass-to-area ratios are considered:

- Mass of 4.2kg and minimum cross-sectional area ($0.01m^2$): 500kg/m 2 = $0.002m^2$ /kg.
- Mass of 4.2kg and maximum cross-sectional area ($0.15m^2$): 28kg/m 2 = $0.036m^2$ /kg..

The computed probability of collision with large objects for the satellites is 1.6586e-06 below the maximum acceptable probability of 0.01.

Result for Requirement 4.5-1: COMPLIANT

Requirement 4.5-2. Limiting the probability of damage from small objects when operating in Earth or lunar orbit

There are no sub-systems which in case of damage will prevent the satellite disposing by natural re-entry as it is intended to.

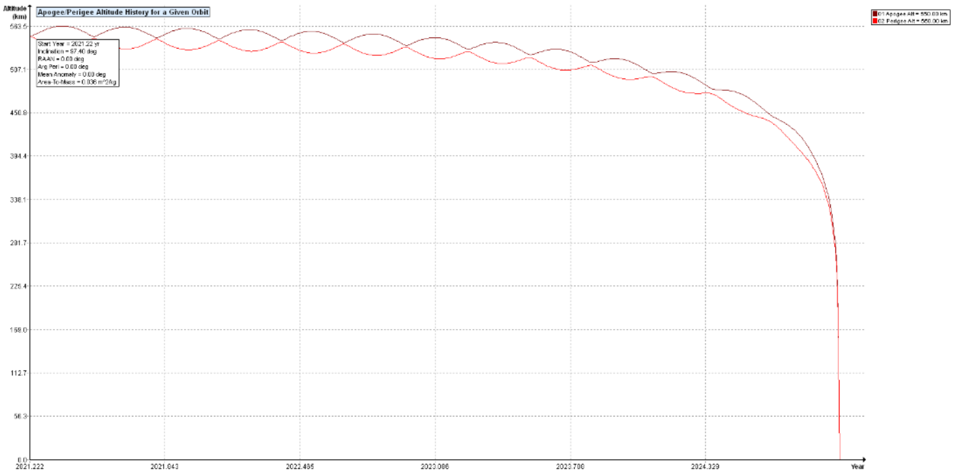
Result for Requirement 4.5-2: COMPLIANT

6. Spacecraft Post-mission Disposal Plans and Procedures

An uncontrolled re-entry with a complete burn out in the top layers of the atmosphere (above 50km) is expected with a mission of this characteristics. Therefore, no manoeuvres, sails, tether systems or similar are considered to accomplish the disposal of the spacecraft.

Requirement 4.6-1. Disposal for space structures in or passing through LEO

The altitude of the satellite is computed from its worst case initial circular orbits at the altitude of 550 km, in its end of mission configuration. The area to mass ratio for the tumbling spacecraft is used (0.036 m^2/kg). (i.e. Minimum mass (4.2kg) and maximum cross-sectional area (0.15m^2): 28kg/m^2 = 0.036m^2 /kg)



Result for Requirement 4.6-1: COMPLIANT

Requirement 4.6-2. Disposal for space structures near GEO

There are no space structures near GEO involved in this mission.

Result for Requirement 4.6-2: COMPLIANT

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

There are no space structures between LEO and GEO involved in this mission.

Requirement 4.6-3: COMPLIANT

Requirement 4.6-4. Reliability of post-mission disposal operations in Earth orbit

In best-case scenario, the cross-section area will be 0.15 m^2. Taking into account a SSO insertion of 550km altitude, the satellite will re-enter after 3.718 years.

In nominal operations, the cross-section area will range from 0.03m^2 and 0.15 m^2. With a nominal orbit starting at 550km of altitude, a mass of 4.2 kg and considering an average cross-section area of 0.09 m^2, the satellite will re-enter after 4.89 years

Result for Requirement 4.6-4: COMPLIANT

7. Spacecraft Reentry Debris Casualty Risks

Requirement 4.7-1. Limit the risk of human casualty

The risk of human casualty was computed by DAS v3.1.2 for an uncontrolled reentry to be 1:100000000 for 3B5GSAT. This is the lowest output of the DAS software. Considering 0 energy makes it to the ground during reentry, the risk of human casualty is effectively 0. The results are summarized in the table below.

Object	Risk of Human	Subcomponents	Demise	Total Debris	Kinetic
Name	Casualty	Object	Altitude (km)	Casualty area (m2)	Energy (J)
3B5GSAT	1:100000000			0.00	0.00
		EPSA1	72.3	0.00	0.00
		EPSA2	73.1	0.00	0.00
		OBDH	75.8	0.00	0.00
		ADCS Computer	76.7	0.00	0.00
		COMMS TT&C	72.6	0.00	0.00
		COMMS MB	75.1	0.00	0.00
		Antenna	75.8	0.00	0.00
		Reaction wheels	76.3	0.00	0.00
		Solar panels	78.0	0.00	0.00
		Closure Panel L	78.0	0.00	0.00
		Closure Panel S	78.0	0.00	0.00
		Torque rods	76.1	0.00	0.00

Result for Requirement 4.7-1: COMPLIANT

8. Collision risk posed by tether systems

Requirement 4.8-1. Mitigate the collision hazards of space tethers in Earth or Lunar orbits

No tethers are to be used in Sateliot mission.

Result for Requirement 4.8-1: COMPLIANT

END of ODAR for Sateliot Mission

Approved: [Author's name, Position, Date]