BlueWalker 3 Orbital Debris Mitigation

Pursuant to Section 5.64 of the Federal Communications Commission's ("FCC") rules,[1/] AST&Science LLC ("AST") provides this description of the design and operational strategy of the BlueWalker 3 ("BW3") spacecraft to demonstrate how it will mitigate orbital debris.[2/]

Executive Summary

The following document outlines how the AST BW3 mission has been designed to meet or surpass all of the Section 5.64 orbital debris mitigation rules. Specific assessments include a list of on-orbit objects and their associated de-orbit casualty risk, the probability of accidental explosions, the probability of collisions with large objects and the satellite de-orbit plan and analysis.

Analysis using the NASA DAS 3.0.1 software demonstrates that the primary spacecraft will naturally de-orbit within 4 years and presents a total casualty risk of only 1:19,700, surpassing the requirement of 1:10,000 by nearly a factor of two. A secondary object, the Launch Vehicle Adaptor, will naturally de-orbit within 8 years and presents a total casualty risk of only 1:12,500, also surpassing the requirement.

The possibility of explosions due to over pressure of propellant tanks, unanticipated mixing of fuel/oxidizer, or overcharging/damage of batteries has been mitigated in several ways. The propellant is an inert gas and the tank has a passive overpressure relief valves. The batteries are constantly monitored by highly redundant electrical power system modules and discharged/taken out of service if their operational parameters exceed acceptable thresholds.

The probability of collision with large objects, as demonstrated here, is no greater than that for a much smaller spacecraft due to the orientation of BW3 during operation (*i.e.*, flying edge-on). Nevertheless, AST shall maintain all best practices with regard to conjunction assessment, collision avoidance maneuvers using onboard propulsion, and sharing of ephemeris data.

Finally, despite meeting the de-orbit casualty risk for uncontrolled re-entry, AST will conduct a controlled deorbit of BW3 at end-of-life to ensure safe disposal of the spacecraft in accordance with IADC Guidelines.

I. Assessment of Debris Risk; List of Objects and Casualty Risk (§ 5.64(b)(1))

AST has assessed and limited the amount of debris released in a planned manner during normal operations, and has assessed and limited the probability of the BW3 becoming a source of debris by collisions with small debris or meteoroids that could cause loss of control and prevent post-mission

[1/] 47 C.F.R. § 5.64.

[2/] This filing is subject to a request for confidential treatment. Information highlighted in black is subject to that request and is redacted in this filing.

disposal.[3/] Table 1 presents the list of objects (mission-related objects or space debris) planned to be released as part of the nominal mission, including physical characteristics, orbital characteristics and predicted orbital lifetime. These total causality risks were generated using the NASA Debris Assessment Software (version DAS 3.0.1) using characteristics of the spacecraft and launch vehicle interface. This launch vehicle interface consists of the launch vehicle adapter and separation ring, which will be released by the spacecraft. Retaining this object with the spacecraft would interfere with the operation of the spacecraft payload. Furthermore, this object may not be kept with the launch vehicle for disposal as required by the launch vehicle provider. However, the casualty risk for this object complies with the 1:10,000 requirement, shall be released from the spacecraft outside of the GEO protected region, and shall not remain in LEO orbit for greater than 25 years. The expected lifetime of the launch vehicle adaptor (LVA) after release is at most 8 years, assuming mean solar activity levels and an average surface area for uncontrolled deorbit of 5.95 m^2. In this analysis, the maximum cross-sectional surface area of the LVA is 10.04 m2 and the minimum cross-sectional area is 1.85 m^2. The total mass is estimated at 150 kg.

Table 1 – List of objects for BW3

Object	***Description***	***Orbital Lifetime***	***Total Causality Risk[4/]***
ControlSat and Phased Array	Main body with phased array composed of individual Micron elements	4 years	1:19,700
Launch Vehicle Adapter and Separation Ring	Launcher to satellite interphase ring with hold-down and release mechanism (HDRM)	8 years	1:12,500

Since the ControlSat and Phased Array can be subdivided into its constituents, any individual component that survives re-entry with an impacting kinetic energy greater than 15 J will contribute to the total debris casualty area that results in the total casualty risk of 1:19,700. A summary of all surviving objects with their corresponding debris casualty areas and kinetic energies can be seen Table 2. Of the materials surviving re-entry, only the ControlSat, in entirety, and the high displacement joints (HDJs) of the LEO mechanical deployment system (LMDS) have sufficient impacting kinetic energy to contribute to the total debris casualty area *(as denoted by the red text in the table below)*.

3/ BW3 will occupy a circular orbit at an altitude between 500 km and 600 km. For purposes of the analysis here, AST uses the worst-case scenario of operations of 600 km.

4/ Based on the fragment survival analysis and total casualty risk to the population. *See* NASA NSS 1740.14 [RD4].

Table 2 – List of objects that survive re-entry

Object	***Quantity***	***Modeled Material***	***Debris Casualty Area (m2)***	***Kinetic energy***
ControlSat	1	Aluminum	2.67	147,993
[redacted]	[redacted]	Iron	33.5	12.7
		Aluminum	1.75	57.7
		Graphite Epoxy	184.27	0.06
Total Debris Casualty Area (m2)				4.42
Total Causality Risk				1:19,700

II. Probability of Accidental Explosions (§ 5.64(b)(2))

AST has assessed and limited the probability accidental explosions that would result in the spacecraft becoming a source of debris. The two sources of on-orbit explosions are propellant tanks and batteries, both of which will be continuously monitored throughout the spacecraft lifetime for failure modes. The batteries are continuously monitored by an electrical power system (EPS) module to avoid over-charging/discharging. Should battery operation fall outside of an acceptable range it will be discharged and taken out of service, removing any stored energy it contained. All batteries on-board the control satellite will have a 1.5 mm aluminum casing and will be thermally isolated to mitigate thermal loads. Additionally, the batteries will have protective circuitry to regulate safe and nominal voltage and current levels. The propellant for the electric propulsion system is an inert and non-reactive noble gas and does not present a source of energy conversion in the event of a gas leak. The pressurized propellant tank will be continuously monitored with downlink of state-of-health telemetry. Propellant safety measures include a system of pressure control and relief valves, with complete thermal isolation and temperature control. Any stored energy remaining at the spacecraft's end-of-life will be removed via depletion of the propellant tank and permanently discharging the on-board batteries.

III. Failures Leading to Debris: Collision with Large Objects Assessments (§ 5.64(b)(3))

The probability of a collision occurring between any two objects in Earth's orbit depends primarily on the likelihood of them passing close to one another -- a situation referred to as a conjunction -- but also on the apparent cross-sections of the two objects as projected along their relative velocity vector at the time of conjunction. So, all else being equal, the probability of a CubeSat colliding with something the size of the International Space Station is much greater than the probability of it colliding with a second CubeSat. Of particular importance is that it is the apparent cross-section projected <u>along</u> the relative velocity vector that matters in this calculation.

The vast majority of objects in low Earth orbit can be found in low eccentricity orbits. As a result, when two such objects pass near one another, the relative velocity vector lies very close to the local horizontal plane -- they do not tend to come from "above" or "below". This is true regardless of the relative inclinations of the two orbits, which would merely determine if the secondary object came from the "left" or "right" relative to the primary object's own direction of motion. In fact, among the operational satellites that will cross the BW3's orbital plane, the highest relative velocity angle above the local horizontal of <u>any</u> satellite is less than 4 degrees. *This means that it is the projected cross-section of the BW3 satellite within 4 degrees of the local horizontal plan that determines it collision probability.*

While the planform (as viewed from above) of the BW3 satellite is about 60 square meters in area, the cross-section of the satellite as viewed within the local horizontal plane is less than 1 square meter. This is because the BW3 satellite flies in the same orientation as a Frisbee -- although the satellite is not rotating as a frisbee would. In this "edge-on" flight orientation, the probability of a collision is therefore more than 60 times smaller than what it would be if the satellite were to fly perpendicular to its velocity vector. While the usual approach in calculating the probability of a collision is simply to use the largest dimension of the spacecraft as if it were the diameter of a spherical object, this would result in overestimating the probability of collision by more than a factor of 60.

AST has assessed and limited the probability of BW3 becoming a source of debris by collisions with large debris or other operational space stations. The probability of collision with large debris can be estimated using the NASA Debris Assessment Software (version DAS 3.0.1). However, because BW3 presents such a small cross-section relative to its mass, it falls outside of the range of parameters that are allowed as inputs to DAS. So, to make a highly conservative estimate of the collision probability, AST used the full planform area of BW3, as if the satellite were actually flying normal to its velocity vector. Even under this assumption, the probability of collision was $3.66(10^{-4})$ (1 in 2,732) which is compliant with the current regulations. *However, the actual probability given the edge-on flight configuration would be closer to $6.1(10^{-4})$ (1 in 164,000), well within compliance by a significant margin.*

Even this calculation is an overestimate as it assumes that the spacecraft will make no effort to perform collision avoidance maneuvers during its operational lifetime. However, the spacecraft also will have the capability to be maneuvered to avoid collisions with large objects, limiting even further the probability that the spacecraft will become a source of debris. Satellite station-keeping within a given orbital plane is monitored by both a primary and secondary flight operations location, one in Midland, TX and the other in College Park, MD. GPS receivers and batched least squares orbit determination methods provide precise satellite positions, known to (much) less than 10 m, that will be used to maintain a precision ephemeris for each spacecraft. The satellite will be registered with the 18th Space Control Squadron or successor entity prior to deployment. Its initial deployment, ephemeris, and planned maneuvers will be shared with the 18th Space Control Squadron or successor entity, the Joint Space Operations Center (JSpOC), other operators with assets flying at the same orbital altitude, and may also be made publicly available. Provided information is made available on new launches that might cross the operational altitude of the AST satellites, these can be monitored, however there is no possibility of action should the launch vehicle malfunction and cross paths with one of the AST satellites. New spacecraft can be placed in a lower altitude, transfer orbit before reaching the final position. Conjunction warnings will be provided by JSpOC and LeoLabs. A third-party analysis software will assess the probability of collision as it evolves over time up to seven days preceding the conjunction at which point a maneuver decision will be made based on the assessment of the collision probability.

The spacecraft will contain an on-board electric propulsion system that is required to provide a collision avoidance maneuver within a 24-hour window of an identified probable conjunction event. Any uncertainty in the time or distance of closest approach with a secondary object will be driven by its state uncertainty, given the position precision described above. The collision avoidance process is illustrated in Figure 8. The state vectors of crewed space stations (*e.g.*, the ISS) and visiting vehicles are known very accurately, so the collision probability will drop off very rapidly outside of a predicted close approach distance of about 50 m. Given a collision probability that exceeds the acceptable threshold, the time required to put sufficient distance between the objects at the time of closest approach will be relatively short, and certainly less than a day. Should the propulsion system fail during critical mission operations, collision avoidance maneuvers can still be performed using the [redacted] to maneuver the spacecraft into the high-drag configuration. The altitude would be lowered sufficiently to avoid collision with the secondary object before returning to its operational configuration. While the spacecraft cannot be returned to its initial orbit, this ensures that collision avoidance capabilities are maintained through the mission, while still maintaining an orbit sufficient for continuation of the mission.

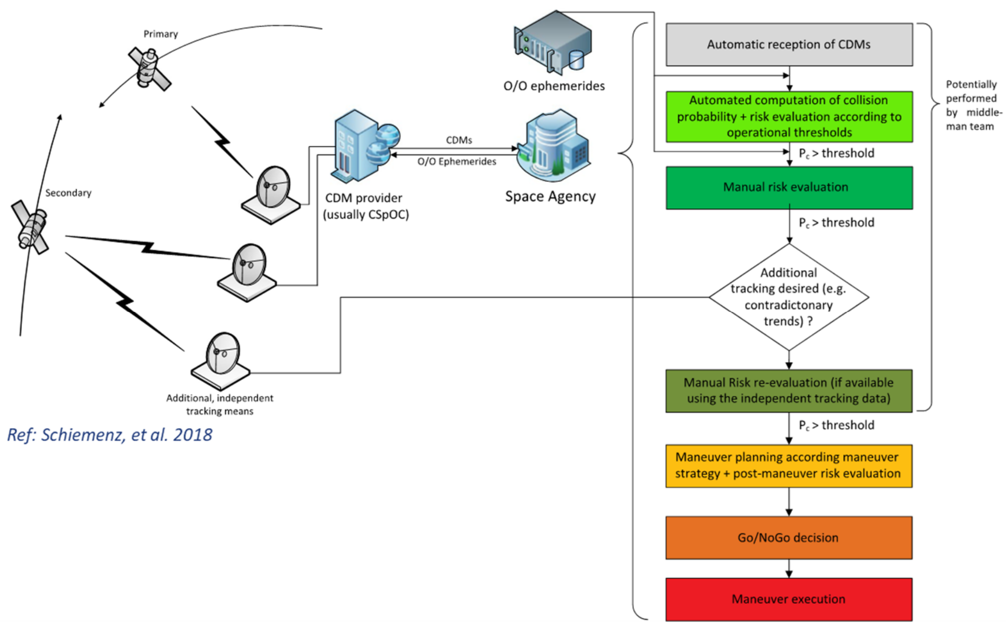

Figure 8 – Collision avoidance system

In the event that a conjunction event exceeds the probability threshold, the satellite's electric propulsion system will be capable of providing an altitude changing maneuver in order to mitigate the conjunction event. By raising the altitude, the satellite's period will increase, causing an in-plane separation from its original trajectory as illustrated in Figure 9. Assuming an altitude change of 100 m, and a 10% operational duty cycle for the thruster, the corresponding change in separation distance can be seen in Figure 10. The 10% thruster duty cycle is a result of operating the thruster only while in solar view with the satellite oriented to provide tangential thrust in-line with the velocity vector. This represents the worst-case scenario for the thruster operating conditions to ensure sufficient distance can be placed between the satellite and its original position.

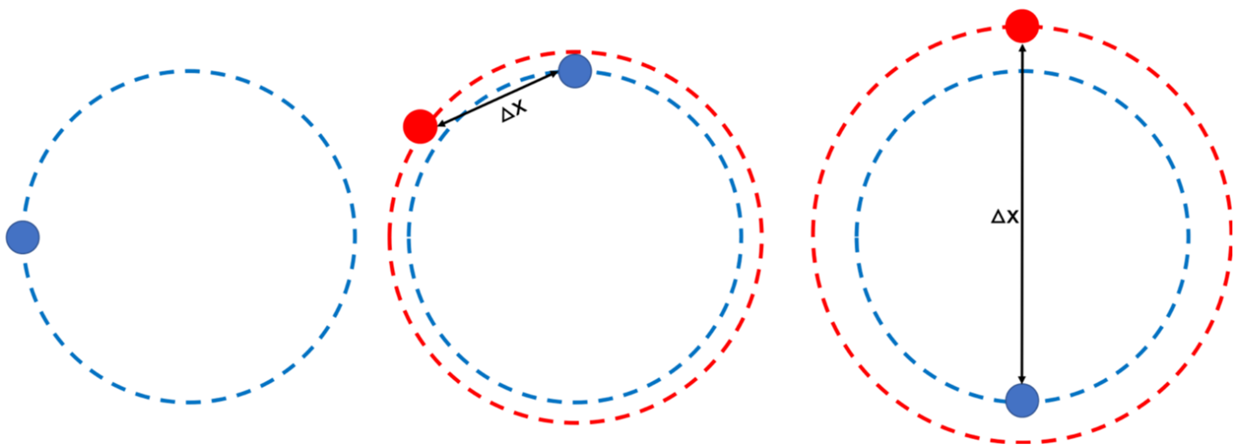

Figure 9 – Conjunction separation via phasing.

Assuming that a conjunction possibility is identified with three days of advanced knowledge, approximately 40 km of separation distance can be placed between the satellite and the debris object. With an approximate uncertainty of 10 km, this provides sufficient separation distance to avoid a collision. Once the satellite is sufficiently out of range of the debris object, the electric propulsion system can perform an altitude lowering maneuver in order to return it to its initial location.

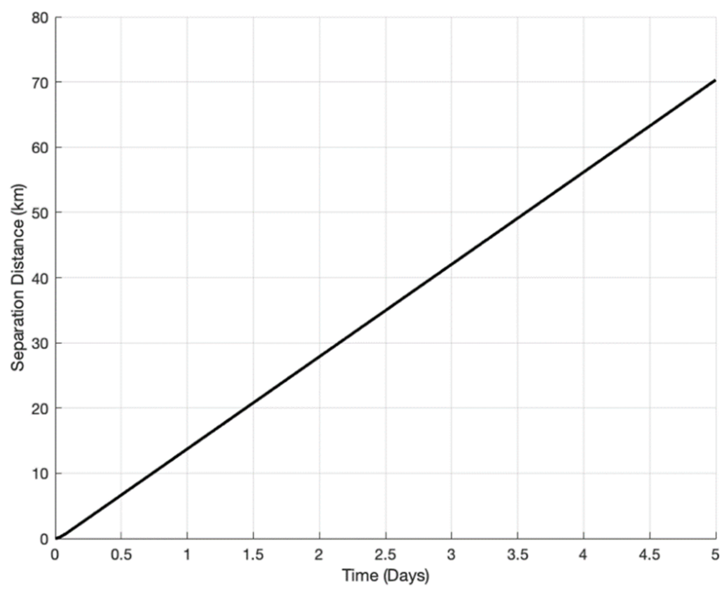

Figure 10 – Satellite separation distance from original trajectory position

Total Impulse and Propellant Mass Allocations

Based on historical debris mitigation maneuvers assessed by NASA, the BW3 baseline is two maneuvers per year with a total propellant mass of 0.075 kg of propellant total. Combining this propellant mass with the mass allocation for the deorbit maneuver, the impulse and propellant budget can be seen in Table 3.

Table 3 – Impulse and propellant mass budget for BW3.

	Total Impulse	Propellant Mass	Description
Deorbit	87 kNs	7.4 kg	Lower altitude to 440 km
Collision Avoidance	1 kNs	0.085 kg	Raise altitude by 100 m then return to orbit. 6 total maneuvers budgeted.
Orbit Maintenance	2 kNs	0.17 kg	Station-keeping as needed.
Margin	10 kNs	0.85 kg	Unforeseen required maneuvers.
Total	**100 kNs**	**8.5 kg**	

IV. Post Mission Disposal Plans: Satellite Deorbit Plan and Analysis (§ 5.64(b)(4))

The BW3 satellite decay time is derived from the drag profile of the satellite. Atmospheric densities for high and low periods of solar activity are obtained from the NRLMSISE-00 model. The satellite geometry is that of a nadir-facing plane with a 10 m aperture targeted for a 600 km altitude, circular orbit and operating over a 2-year mission lifetime. The spacecraft consists of an array of Microns and a ControlSat, with a total allocated mass of [redacted]. The coefficient of drag is assumed to be 2. For the spacecraft either flying edge-on or with the array normal to the velocity vector, the atmospheric drag as a function of altitude for low and high levels of solar activity can be seen in Figure 1. The purpose of flying with the array normal to the velocity vector is to determine how the drag force might assist with deorbiting the satellite. The corresponding orbital decay starting at an altitude of 600 km using the drag profile of Figure 1 can be seen in Figure 2. Averaging the solar activity across the orbital decay, the satellite can be expected to deorbit naturally in over 25 years. This demonstrates the necessity of a planned deorbit to ensure the spacecraft is deorbited within 25 years of End-of-Life (EOL) to be in compliance with IADC Guidelines for Space Debris Mitigation. To ensure a timely and controlled deorbit at end of life, the satellite will be equipped with an electric propulsion system.

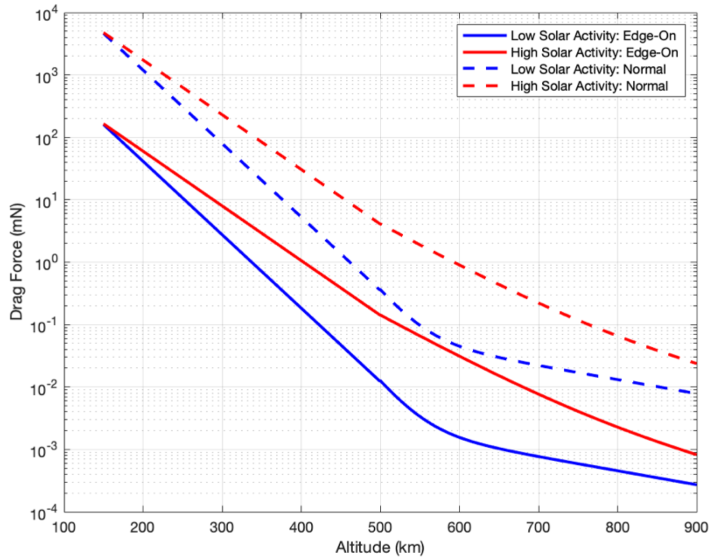

Figure 1 – Atmospheric drag on BW3 satellite as a function of orbital altitude

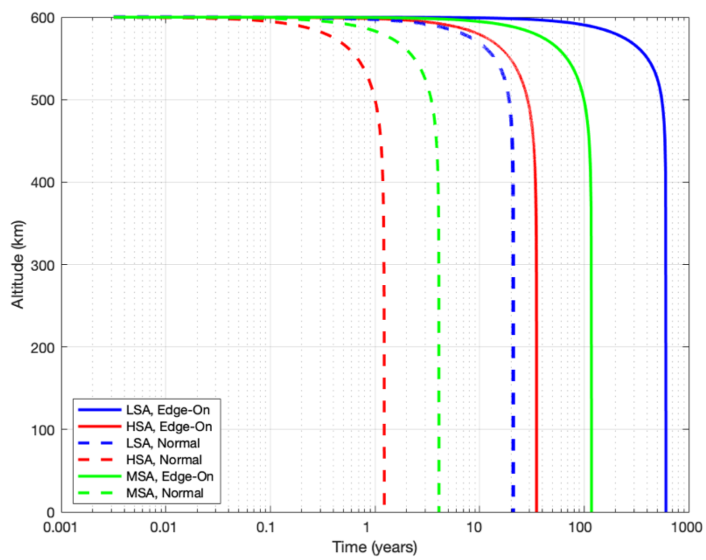

Figure 2 – Natural deorbit time (without propulsion) for the BW3 satellite

To mitigate propellant mass considerations and minimize the deorbit time of the satellite, the satellite can be rotated from flying edge-on (low drag) to flying normal to the velocity vector (high drag) at an altitude at which the drag on the array is comparable to the thrust provided by the electric propulsion system. These low drag and high drag configurations are illustrated in Figure 3. The fastest deorbit time occurs when the altitude at which the drag on the array is identical to the thrust provided by the electric propulsion system. In addition to the propulsion unit, the flight computer has both hardware and software redundancy, and [redacted]. Once the desired disposal altitude is reached, the electric propulsion system can be switched off, and the satellite is then pitched using these [redacted] until the array is normal to the velocity vector. Figure 4 shows the deorbit time as a function of the electric propulsion cutoff altitude. The maximum deorbit time would be approximately three times larger than the times provided in this table due to power constraints limiting operation of the thruster to periods of direct solar view. Using this approach, the deorbit time is well below the 25-year disposal time required to be in compliance with IADC Guidelines for Space Debris Mitigation.

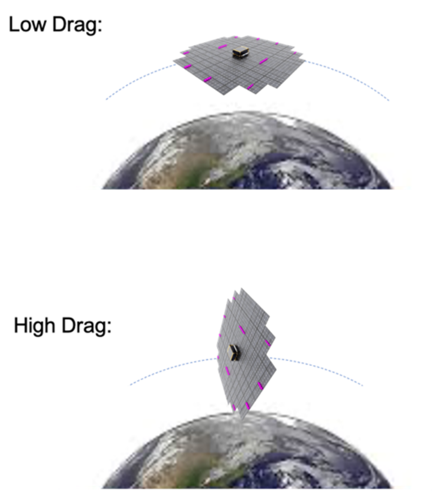

Figure 3 – Low and high drag configurations

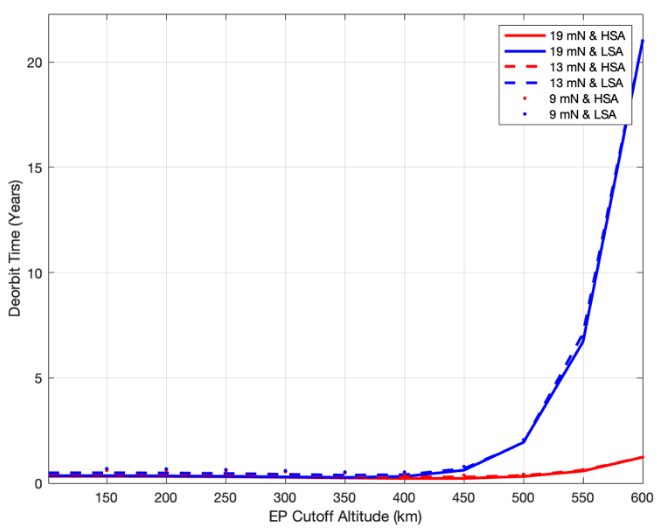

Figure 4 – Deorbit time as a function of the electric propulsion (EP) cutoff altitude

Periodic orbit maintenance of the spacecraft will be performed, if needed, using the propulsion system. The apogee and perigee of the spacecraft within each orbital plane will be maintained within +/- 2 km of their target altitude unless tighter constraints are required. Inclination will be bound to within +/- 1 deg of the target value. Finally, in terms of the post-mission disposal plans for the spacecraft at end-of-life, including the quantity of propellant that will be reserved for post-mission disposal maneuvers, the primary end-of-life plan is a powered re-entry. This will use a combination of first orbit lowering using onboard propulsion and then drag control using the spacecraft attitude and orbit control system (AOCS). Specifically, the on-board electric propulsion system will provide the means of de-orbiting the spacecraft to an altitude at which the maximum spacecraft drag exceeds the propulsive capability. The array can be pitched to the maximum drag orientation and then yawed around nadir to throttle the drag and control the deorbit rate. Using an atmospheric density model with uncertainty (the NRLMSISE-00 model), the impact location of any surviving parts of the spacecraft at re-entry will be continuously evaluated with an uncertainty ellipse. As the satellite falls farther into the atmosphere, the uncertainty in the density will decrease, as will the uncertainty ellipse for re-entry. If the center of the re-entry ellipse shifts too far from the desired location, the rotation of the array so that it is edge-on to the velocity vector will effectively halt the de-orbit process (reducing the drag force), allowing the center of the re-entry ellipse to shift along the ground track to the desired location. When the desired projected re-entry location is reached, the array is again rotated back to the high-drag configuration and the de-orbit continues. Sufficient propellant will be maintained throughout the mission to provide the deorbit maneuver at end-of-life. This maneuver, in accordance with NASA standard NASA-STD-8719.14B for all debris mitigation practices, shall lower the altitude to approximately 440 km, at which point the array can be pitched with magnetic torquers that can control the exposed surface array, effectively throttling the drag. This drag force is larger than the deorbit thrust provided by the electric propulsion system and provides a more controllable descent below 440 km. The allocated propellant mass as a function of the altitude at which the high drag configuration will be used can be seen in Figure 5. Should a critical failure of the propulsion system occur, the spacecraft can be reconfigured to its high drag configuration (array normal to velocity vector) and still deorbit the spacecraft within the required 25 years as seen in Figure 2.

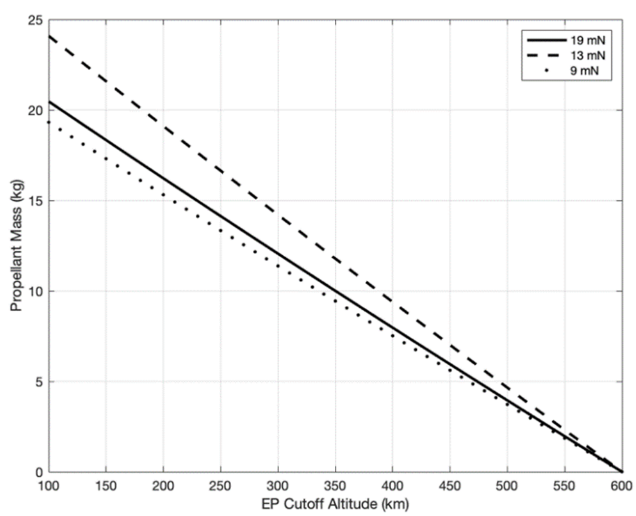

Figure 5 – Propellant mass requirement as a function of the electric propulsion (EP) cutoff altitude.

In the event of a critical mission failure before deployment, an orbital decay analysis was performed to investigate the deorbit time of the spacecraft in its stowed configuration. The deorbit time for various levels of solar activity can be seen in Figure 6. Considering the effects of the solar cycle on the levels of solar activity, the stowed configuration would be expected to deorbit in approximately 100 years by assuming mean solar activity levels throughout the orbital decay. Given that the natural deorbit time exceeds the 25-year requirement, a deorbit plan will be implemented for the spacecraft in the stowed configuration. Maintaining the same impulse and propellant mass budget discussed further in Section 2.7, the stowed satellite can be deorbited to 420 km by exhausting the entire propellant supply. From this altitude, the satellite will naturally deorbit in approximately 6 years. Since the thruster will not be aligned with the center of gravity, a combination of the on-board [redacted] will need to be used in order to compensate the torques imparted by the thruster. The deorbit times as a function of the final altitude that the thruster can achieve can be seen in Figure 7.

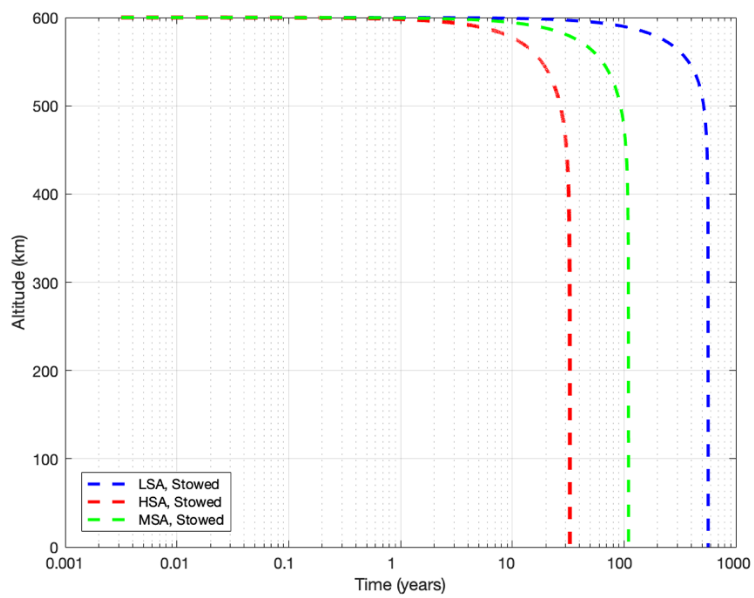

Figure 6 – Orbital decay of BW3 in the stowed configuration.

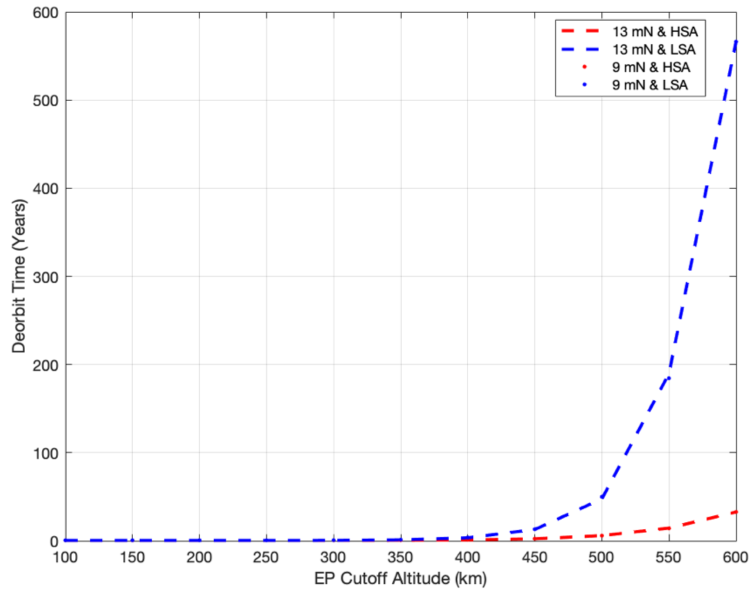

Figure 7 – Deorbit time as a function of final altitude achieved by the propulsion system.

Atmospheric Re-Entry

In the event the AOCS system fails, resulting in an uncontrolled descent of the BlueWalker 3 mission, a casualty risk assessment using the NASA Debris Assessment Software (version DAS 3.0.1) indicates a total risk assessment of 1:19,700, which is lower than the 1:10,000 requirement. Of the debris that does not demise before reaching the surface, those substantially contributing to the total casualty area are well below the 15 Joule kinetic energy requirement. Those components above the 15 Joule requirement contribute a total debris casualty area characterized by the 1:19,700 casualty risk assessment. The total casualty risk assessment for the launch vehicle adapter, which will be released during the deployment state, is 1:12,500.

* * *

CERTIFICATION OF PERSON RESPONSIBLE FOR PREPARING
ENGINEERING INFORMATION

I hereby certify that I am the technically qualified person responsible for preparation of the engineering information contained in this application, that I am familiar with Part 5 of the Commission's rules, that I either prepared or reviewed the engineering information submitted in this application, and that it is complete and accurate to the best of my knowledge and belief.

Raymond J. Sedwick

Dr. Raymond Sedwick
Chief Scientist, Space Systems
AST&Science LLC
2901 Enterprise Lane
Midland, TX 79706
(432) 276-3966