Updated Supplemental Orbital Debris Response

1) How long will the applicant operate the spacecraft prior to starting end-of-life orbit lowering procedures?

RESPONSE: The spacecraft will be in operation for approximately two years before the end-of-life deorbit commences.

2) Please address the probability of damage from small objects as described in NASA-STD-8719.14B, requirement 4.5-2.

RESPONSE:

Requirement 4.5-2 limits the probability of spacecraft being disabled and left in orbit at the End of Mission (EOM). Specifically, it is concerned with the probability that systems critical to EOM disposal may be damaged over the operational lifetime or during EOM disposal (EOMD). The BW3 mission has both a primary and a secondary method for EOMD.

In the primary method, the onboard electric propulsion (EP) system will lower the orbit of the spacecraft until the drag that would be induced by pointing the normal of the array in the direction of the velocity vector (maximum drag) exceeds what can be generated by the thruster. At that point, a pitch-up maneuver points the array into the maximum drag orientation to accelerate the de-orbit. During this phase, the yaw of the array can be controlled through a network of highly distributed processors and actuators, making the system robust to failures.

The secondary method for EOMD, in the event of a thruster failure, would be to perform the pitch up maneuver immediately, and then use the (orientable) high-drag configuration for the entire de-orbit. While this extends the time to de-orbit, the entire EOMD sequence can still be achieved in under 5 years.

Because of the dual method of EOMD, and the highly distributed nature of the Attitude and Orbit Control System (AOCS), a rather large number of failures must occur before EOMD capability is jeopardized. The method outlined in 4.5-2 requires the use of the NASA DAS software to assess the probability that a critical component will fail by first associating that component with a "critical surface" area and then specifying the orientation of the area normal relative to the spacecraft frame, and finally specifying the attitude of the spacecraft. This then needs to be done over the entire mission and during EOMD.

To achieve the conditional failure probabilities, the failure probabilities (impact probabilities) for individual critical components were first found using DAS, and then these were combined by considering the simultaneous failures that would need to occur for any particular failure mode. Table 1 shows the output of using DAS to determine the probability of a small object collision per m^2 of area, per month of operation, from each principle direction at three different altitudes.

Table 1: Small body impact probabilities from DAS software.

Altitude	Probability [1,0,0]	Probability [0,1,0]	Probability [0,0,1]	Probability [1,0,0] SAP Shielding	Probability [0,1,0] SAP Shielding	Probability [0,0,1] SAP Shielding
550	2.69E-05	3.57E-02	1.20E-02	4.00E-08	1.69E-03	6.75E-05
450	7.13E-06	1.28E-02	2.74E-03	1.28E-08	3.30E-04	1.45E-05
350	3.75E-06	3.55E-03	9.29E-04	1.23E-08	7.83E-05	6.29E-06

Columns 2-4 show the probabilities for external components, and columns 5-7 for components that would be housed within an enclosure. The highlighted values are the maxima for each case, which are found to be in the velocity direction and at the highest operational altitude. These were used over the entire mission to provide the most conservative estimate.

Table 2: Total impact probabilities for AOCS components over 5 year period.

Component	Quantity	Area (cm^2)	Time (months)	worst case Prob
[redacted]	[redacted]	16	60	1.95E-07
[redacted]	[redacted]	640	60	3.13E-04
[redacted]	[redacted]	20	60	2.03E-04
[redacted]	[redacted]	80	60	4.88E-06
[redacted]	[redacted]	100	60	1.72E-08
[redacted]	[redacted]	875	60	1.31E-06
[redacted]	[redacted]	100	60	1.72E-08
[redacted]	[redacted]	100	60	1.72E-08
[redacted]	[redacted]	100	60	1.72E-08
[redacted]	[redacted]	100	60	1.72E-08
[redacted]	[redacted]	100	60	1.72E-08
[redacted]	[redacted]	100	60	1.72E-08
[redacted]	[redacted]	202	60	4.33E-02
[redacted]	[redacted]	16	60	4.40E-10
[redacted]	[redacted]	250	60	2.54E-03
[redacted]	[redacted]	100	60	0.00E+00
[redacted]	[redacted]	1	60	0.00E+00
[redacted]	[redacted]	1	60	0.00E+00
[redacted]	[redacted]	17.5	60	0.00E+00
[redacted]	[redacted]	4096	60	1.24E-13
[redacted]	[redacted]	4096	60	1.24E-13
			Total	5.22E-04

Table 2 shows a list of components that are involved in attitude control of the spacecraft during both nominal operations and deorbit. For each component, the number, representative impact cross-section, mission duration and cumulative impact probabilities are listed. If the components are redundant, the probability is for all components to fail, noting again that the worst case values from Table 1 are assumed. Because of the large amount of built-in redundancy, the failure tree is quite complex, so a few critical path components are highlighted for discussion.

The component with the greatest probability of failure due to impact is the thruster (4%), which includes both internal and external parts. However, because of the ability to perform the second method of de-orbit, the thruster is not a critical component. The next highest are the controlSat

EMTs (0.25%), however they are also not in the critical path for attitude control during de-orbit due to the large redundancy in MEMTs distributed over the array. A possible bottleneck is seen to be in the communications path to the ground, where the UHF switch connects UHF receiver 1 and UHF receiver 2 to the FC. However, there are also Sband1 and Sband2 that can be used, so both paths would need to fail (1 in a billion chance). Of note are several highly redundant components (FCs - EMTs) distributed across the array. While they are not all redundant (a subset must be operational), the number of ways that the system can operate under significant component failures is so large that the contributions to system failure are negligible. The highest among them are shown at ~10^{-13}.

To add a final measure of conservatism, the individual component probabilities are summed, implying that any given failure listed in the table would constitute a failure to deorbit. As seen in the communication path example, this is not generally the case. Excluding the highlighted probabilities (already discussed) the resulting total probability is $5(10^{-4})$ or 1 in 2000. Even this highly conservative estimate is **well below the 1% threshold.**

3) Will there be coordination with the ISS and other appropriate agencies when deorbiting through the ISS altitude during end-of-life activities?

RESPONSE: Yes, every maneuver will be coordinated with the ISS, NASA, 18th Space Control Squadron or successor entity, the Joint Space Operations Center (JSpOC), and any other appropriate agency throughout the entire operation and deorbit time lifetime. AST is in the process of establishing a Space Act Agreement with NASA outlining the responsibilities, schedules and milestones associated with COLA activities during the entire mission lifetime.

4) On page 7 of the ODAR, it's stated that 0.075 kg of propellant is expected to be required for 2 expected maneuvers per year. Then, in Table 3, it is indicated that 0.085 kg of propellant will be budgeted for 6 maneuvers. Please provide an explanation for this discrepancy.

RESPONSE: The 0.075 kg in the body of the section was a typographical error and should match the propellant allocation in Table 3 below. The correct propellant allocation is 0.085 kg for all collision avoidance maneuvers. To clarify the wording, two maneuvers are expected per year. Primary operations are expected to last for around two years, with one additional year for deorbit. This propellant allocation is for the total mission duration, or three years, assuming two maneuvers per year.

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**	

5) What is the timeframe from receipt of a CDM to final determination regarding whether a maneuver will be required?

RESPONSE: Within 24 hours of receipt of an initial CDM (5-6 days out), a preliminary propulsive maneuver will be determined and scripted to begin no later than 24 hours prior to the predicted conjunction event. Within 24 hours of the predicted time of closest approach (TCA), the propulsion system can safely maneuver the spacecraft out of range of a 10 km uncertainty in the conjunction location as shown below in Figure 1, which should be much greater than the uncertainty within 24 hours of TCA. The ballistic coefficient of BW3 is quite high due to the very low projected area in the velocity direction, so the predicted ephemeris uploaded to Space-Track can be quite accurate even several days out. The uncertainty in the conjunction will then be driven primarily by the secondary object.

Over the 4-5 days prior to the conjunction event, the covariance of the secondary object's ephemeris and the probability of collision will be updated with every new CDM between the initial report and the conjunction event. The avoidance maneuver will be revised and remain in an "active" state until the probability of collision falls below 10^{-4} (0.01%) and tracks downward over several CDM updates. No later than four days prior to the expected conjunction, the electric propulsion system will undergo a state of health check and conditioning procedure. If the propulsion system is operating nominally, and the collision probability remains above the 0.01% threshold, the scripted maneuver will be executed as planned 24 hours in advance of the predicted TCA.

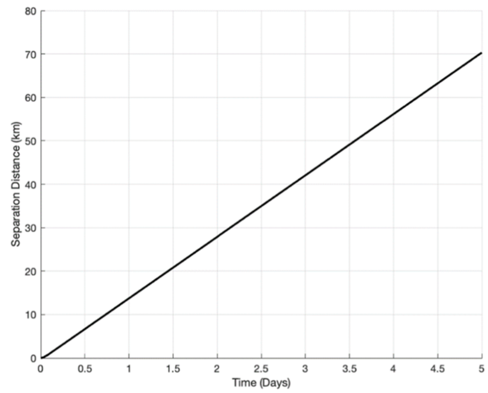

Figure 1: Satellite separation distance from original trajectory position.

6) What is the timeframe from receipt of a CDM to final determination regarding whether a maneuver will be required in the event of a propulsion system failure and a high-drag configuration will be utilized to perform avoidance?

RESPONSE: Initially, by the scenario above, the nominal maneuver is planned using the thruster. At no later than the TCA-4 day point, the electric propulsion system undergoes a state of health check and conditioning procedure. If it fails this check, a contingency high-drag maneuver will be planned while attempts are made to bring the thruster online.

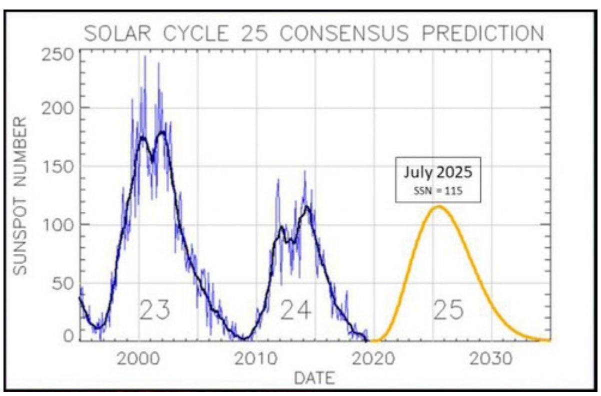

Figure 2: NOAA / NASA Solar Cycle Prediction for upcoming Cycle 25.

The required duration of the high-drag maneuver will depend on the level of solar activity at the time and its influence on the atmospheric density. A direct evaluation of the density will be known at the time from tracking long-term drag effects, but for now a prediction will have to suffice. Figure 2 shows the predicted level of solar activity over the next cycle from 2020-2035. Figure 3 shows the drag versus altitude using data from the most recent cycle. Solid lines show the edge-on drag at minimum and maximum solar activity and dashed lines show the normal drag (maximum) for the same limits.

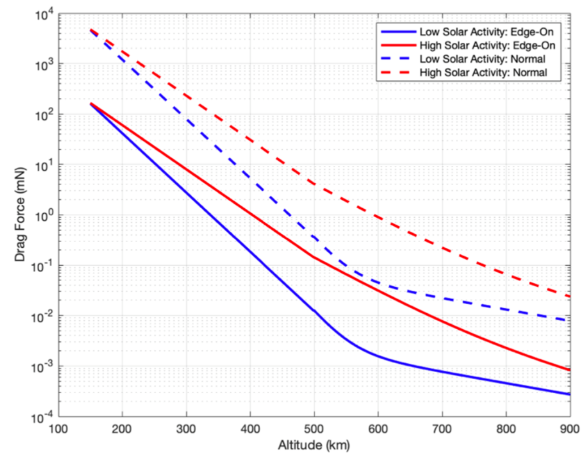

Figure 3: Atmospheric drag on BW3 satellite as a function of orbital altitude

Table 4 shows the approximate predicted drag levels versus year from 2020 to the predicted solar maximum in 2026. The nominal operation for BW3 is 2022-2024, not including de-orbit, over which time the drag is predicted to vary from 0.28 mN to 0.76 mN. Figure 4 shows the velocity vector separation distance from the predicted conjunction point vs. time for an assumed arithmetic mean value (over the solar cycle) of 0.76 mN, which would correspond to predicted 2024 solar activity. It can be seen that to reach the same 10 km separation as the nominal thruster maneuver requires just under 1.2 days, or about 28 hours. In 2022, at a drag of 0.28 mN, this separation distance would be achieved in just over 3 days, or about 75 hours. The TCA-4 day health check of the thruster then provides 33% margin for the high-drag maneuver if needed. Since the actual drag levels will be accurately known, if it is identified that for some reason the drag levels are lower than predicted, the CONOPS can be adjusted accordingly. The spacecraft can be pitched into the high-drag configuration within one orbit, or about 90 minutes, so this has no significant impact on the schedule.

Table 4: Predicted atmospheric drag on BW3 satellite at 550 km over the next half cycle

Year	Solar Activity	Normal Drag force (mN)
2020	Min	0.10
2021		0.17
2022		0.28
2023	Mean (geometric)	0.46
2024	Mean (arithmetic)	0.76
2025		1.26
2026	Max	2.09

It should be noted that although the satellite is pitched up into its high-drag configuration, the satellite can be easily rotated around its nadir direction (yaw) in this configuration. This allows the satellite to present its minimum projected area along the relative velocity vector at any updated time of closest approach.

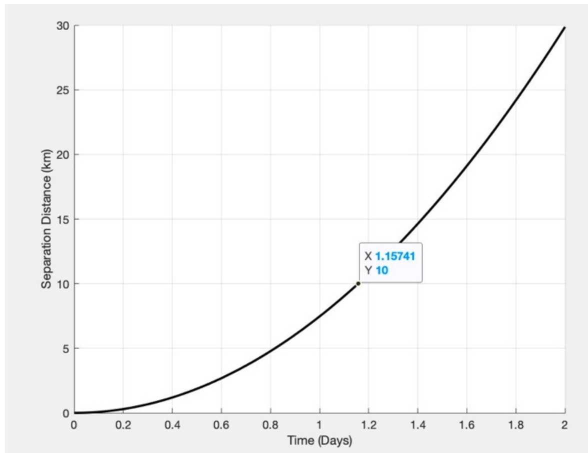

Figure 4: Velocity-track displacement versus time at mean atmospheric density

7) What are the targeted reduction in probabilities of collision when planning an avoidance maneuver when using propulsion and when using a high-drag configuration?

RESPONSE:

The targeted Pc threshold to execute a maneuver is $>10^{-4}$ within 24 hours of TCA to use the propulsion system, or within 48 hours of TCA to use the high-drag configuration. If the time required for the high-drag maneuver to reach this level is predicted to exceed 48 hours due to unexpectedly low drag conditions, then the time will be adjusted accordingly.

SUMMARY

The BW3 system has been designed with a high degree of component redundancy to minimize the likelihood that anticipated small body impacts will significantly increase the risk associated with safe operation and EOM disposal. As noted in the response to Question 3, AST is in the process of establishing a Space Act Agreement with NASA outlining the responsibilities, schedules and milestones associated with COLA activities during the entire mission lifetime. This entails continuous and ongoing communication and coordination with NASA and other spacecraft owner/operators during the nominal two-year mission and the safe and responsible removal of the spacecraft from orbit at EOM. The BW3 system specifications and CONOPS will support these activities in accordance with the NASA Spacecraft Conjunction Assessment and Collision Avoidance Best Practices Handbook and as specified in the aforementioned Space Act Agreement.

As with the large body collision probability analysis, the DAS results show that small object impacts within the orbital shell of the spacecraft exceed those coming from the radial direction by several orders of magnitude. In the case of the large body collision analysis, the flight orientation of the spacecraft -- flying in the orientation of a frisbee -- greatly reduces the collision cross section in the relative velocity direction during potential conjunctions. A similar situation exists here, however rather than relying on this reduction for the calculations, the worst case particle flux has been assumed over the entire mission for all analyses. This would imply that the actual probabilities would be even lower than what has been presented by several orders of magnitude.

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I HEREBY CERTIFY THAT I AM THE TECHNICALLY QUALIFIED PERSON RESPONSIBLE FOR THE PREPARATION OF THIS UPDATED SUPPLEMENTAL ORBITAL DEBRIS RESPONSE, AND THAT IT IS COMPLETE AND CORRECT TO THE BEST OF MY KNOWLEDGE AND BELIEF.

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