NASA EXPLORATION MISSION 2 MISSION DESIGN

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Exploration Mission 2 (EM-2) will be NASA's first manned flight on the Space Launch System (SLS) and Orion Spacecraft. The mission has been changed from an SLS Block 1B configuration to Block 1. This change has necessitated a reexamination of the flight profile to determine what changes must be made in order to accommodate the reduced launch vehicle performance on the Block 1. Launch availability and orbital debris risk will be traded to find the best flight profile for both SLS and Orion.

INTRODUCTION

Exploration Mission 2 (EM-2) will be NASA's first crewed flight on the Space Launch System (SLS) and Orion Spacecraft. The flight manifest has changed this mission from an SLS Block 1B configuration to the Block 1. Due to a smaller upper stage, launch vehicle performance is reduced on the Block 1 compared to the Block 1B, which precipitated some changes to the mission profile. One such change is the loss of a 100 nautical mile circular parking orbit, which is below the major orbital debris bands in Low Earth Orbit (LEO). Instead, the Block 1 Core Stage must insert the Interim Cryogenic Propulsion Stage (ICPS) into an elliptical orbit that will fly through Micro-Meteoroid and Orbital Debris (MMOD) debris bands multiple times before inserting Orion into the High Earth Orbit (HEO). Orbital debris has increased significantly since the Apollo Program, so it must be considered for any crewed mission. This paper will evaluate placement of LEO & HEO apogee, timing of Perigee Raise Maneuver (PRM), Orion vehicle checkout prior to Trans-Lunar Injection (TLI), abort performance, and the MMOD risk against the overall launch availability and concept of operations to find an optimized mission design.

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The mission design will utilize Copernicus (Ref. 1),an n-body trajectory optimization tool, developed and maintained at Johnson Space Center in Houston, TX. Copernicus will use its plugin capabilities (Ref. 2 and 3) to call a database of SLS ascent trajectories developed in the Program to Simulate Optimized Trajectories (POST) (Ref. 4). The MMOD risk analysis will be done in the BUMPER 3 simulation with the appropriate MMOD databases. BUMPER calculates the number of failures in a deterministic fashion by computing the number of MMOD particles that exceed the ballistic limits of the different components on ICPS. Final mission design will be determined in an official SLS trade team with cross-program representation from the Orion program.

Figure 1. EM-2 on SLS Block 1B

EM-2 BLOCK 1B MISSION OVERVIEW

For the original EM-2 Block 1B mission, the Exploration Upper Stage (EUS) delivered the EUS/Co-Manifested Payload (CPL)/Orion stack to a 100 nmi Low Earth Orbit (LEO). The stack then loitered sufficiently long to perform preliminary checkouts of the Environmental Control and Life Support System (ECLSS) on Orion. After anywhere from 120 to 177 minutes in LEO, the EUS re-ignited, performing an Apogee Raise Burn (ARB) to place Orion onto a 24-hour High Earth Orbit (HEO). Shortly after the ARB, the EUS/CPL stack and Orion separated. After separation (and approximately 30 minutes after the end of the ARB), the EUS performed the TLI-1 burn, targeting the EUS/CPL stack to a lunar flyby for heliocentric disposal. After TLI-1, CPL separated from the EUS and re-targeted to an independent lunar arrival target. After coasting approximately 1 revolution in the 24 hour HEO (after ARB), Orion performed its own TLI-2 burn near the perigee of the 24-hour HEO, targeting an approximately 8-day (+/- 0.5 days) free return lunar flyby designed to return Orion back to a planned Earth Entry Iterface (EI) for a splashdown off the coast of San Diego, CA. The 24-hour HEO allowed for additional vehicle checkouts and provided for a low TLI-2 (dV) performance requirement, while aligning the scheduled waking crew time with the major (TLI-2) burn and the lunar close approach and flyby. After TLI-2, placeholders are reserved for

Outbound Trajectory Correction (OTC) and Return Trajectory Correction (RTC) burns on both the outbound (pre-lunar encounter) and return (post-lunar encounter) free-return trajectory legs, respectively. This mission profile will be referred to as the "Hybrid Free Return" mission. This profile was determined to be low risk to the Orion crew and was highly desired to remain as the flight profile given the change in manifest to the Block 1 SLS configuration. Figure 1 provides and illustrative annotated graphic overview of the Block 1B MTLI mission.

EM-2 BLOCK 1 MISSION OVERVIEW

With delays to the Block 1B development schedule that threatened the EM-2 launch schedule, a decision was made to use the ICPS for the first crewed mission. This change from a Block 1B (EUS upper stage) to Block 1 (ICPS upper stage) configuration, resulted in a number of impacts that dictated a new EM-2 Block 1 mission design and associated analyses and performance sensitivity studies to determine a preferred reference mission that would best support crew safety and mission success.

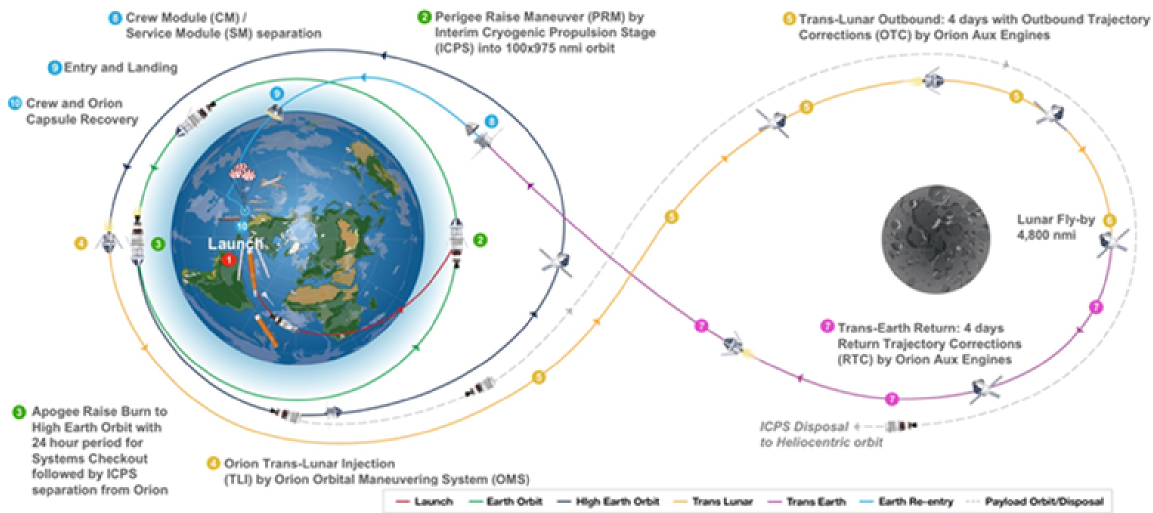

Figure 2. EM-2 on SLS Block 1

The starting mission profile for the trade study (Figure 2) depicts the EM-2 Block 1 mission. The first change was moving from the circular LEO utilized on the Block 1B to the same elliptical orbit used on EM-1, 975 x 22 nmi. After Core insertion, the ICPS/Orion stack coasts up to apogee where the ICPS performs its first burn to raise perigee altitude to at least 100 nmi. The stack then coasts to near the second perigee passage where the ICPS performs the ARB to the 24 hour HEO. From this point on, the mission profile is very similar to the Block 1B EM-2 mission except that the CPL has been removed from the manifest due to performance constraints on the SLS Block 1. Instead, small cubesats may be included at a later date, if vehicle performance permits.

The ICPS performance limitations dictate a trajectory design similar to EM-1 (Ref. 5), along with the elimination of a CPL. While the EM-2 Block 1 Orion spacecraft weighs more than the EM-1 Block 1 Orion, its nominal free return flyby mission requires less propellant than the EM-1 mission which inserts and, subsequently, departs from a lunar Distant Retrograde Orbit (DRO) with powered lunar flybys on both the outbound and inbound trajectories.

The EM-2 Block 1B trajectory benefited from the EUS' capability to deliver an EUS/CPL/Orion stack to a 100 nmi circular phasing orbit, allowing it to select from both ascending and descending node Earth departures, thus garnering more nominal (unconstrained) launch opportunities. The Block 1 mission, with its more limited ICPS upper stage, takes advantage of the SLS Core Stage to deliver the ICPS/Orion stack to an elliptical parking orbit. This aids the ICPS in placing Orion in a 24-hour HEO and then performing TLI-1 to heliocentric disposal. Due to performance limitations, the elliptical orbit only provides ascending node departures, resulting in a loss of about half of the launch opportunities compared to the 100 nmi circular orbit.

The HEO orbit does provide for a relatively low Orion TLI-2 performance requirement, leaving significant remaining propellant to be applied in the event of an abort. Positive abort options for the crew are a top priority for this mission and much analysis to date and planned for the future will be brought to bear on the mission design to ensure the best possible options for returning a crew safely back to Earth in the event of a required abort.

EM-2 PARAMETRIC TRAJECTORY ANALYSIS

Prior to starting the broader EM-2 trajectory design trade study, a set of parametric analyses were performed in order to determine how sensitive payload performance is to various in-space trajectory design decisions. A large number of trajectory parameters were identified and studied in this phase. Those which were deemed to have a significant impact and were pursued further are described in more detail in the following sections.

Every study described below was performed by modifying the baseline EM-2 mission profile to accommodate the parameter to be studied. Only one trajectory parameter was varied from the baseline in each study. In this way, each study is independent from the other studies and provides performance sensitivities with respect to only the parameter in question. Each study was a point case, or set of point cases, run with a launch date of June 7, 2022.

Perigee Raise Maneuver Timing

The PRM is the first major engine burn executed by the ICPS. Its purpose is to raise the perigee of the Core stage insertion orbit to an altitude of 100 nmi. The standard technique for executing such a maneuver would be to center the burn on the insertion orbit apogee. However, various considerations with the Orion power subsystem and MMOD risk drove the team to study executing this maneuver earlier.

The team studied the impact to mission performance of placing the PRM at fixed durations 30 seconds, 1 minute, 2 minutes, 5 minutes, and 10 minutes after Core stage separation. For a PRM at apogee, the burn would occur approximately 43 minutes after Core stage separation.

The primary performance metric considered when evaluating the impact of the various PRM timing options was the additional amount of propellant required for ICPS and Orion to complete the mission. Here, as in other parametric studies, results were provided relative to the EM-2 baseline mission, which placed the PRM maneuver at apogee. Results are shown in Figure 3.

Not surprisingly, the amount of propellant ICPS must expend in order to complete the mission increases significantly as the timing of the maneuver shifts earlier. The impact to Orion is less severe, which is expected, as the ICPS is still placing Orion into approximately the same pre-TLI orbit in each case.

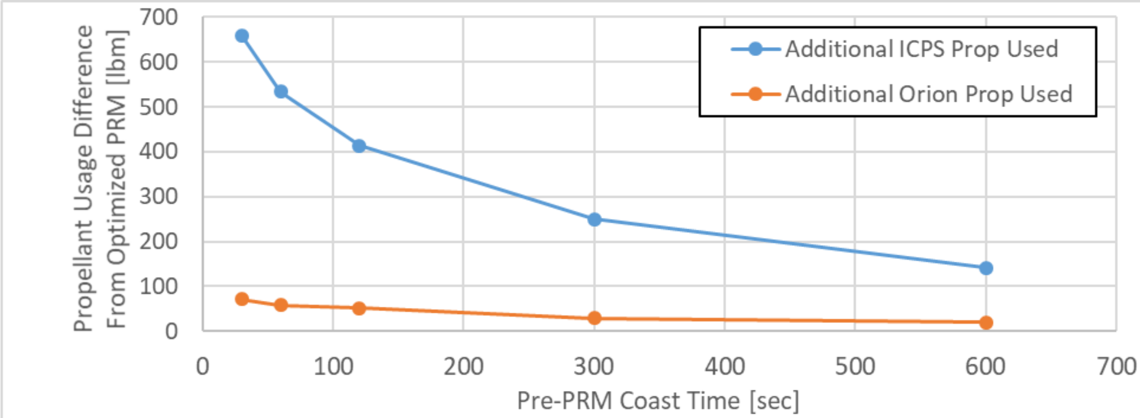

Figure 3. Impact of Propellant Usage for Various PRM Timings

Core Stage Insertion Apogee

An analysis was performed to study the impact to payload performance of altering the Core stage insertion orbit apogee altitude. The baseline EM-2 mission uses an insertion apogee altitude of 975 nmi. Also studied was an insertion apogee altitude of 1200 nmi. No other parameter was varied from the baseline EM-2 mission. Not surprisingly, increasing the insertion apogee results in less propellant expenditure by ICPS by roughly 1400 lbm. However, the higher insertion apogee also requires Core stage to burn over 8000 lbm of additional propellant, thereby leaving less remaining on the Core stage. The Core State insertion apogee is synergistic with the intermediate orbit period as those are the two main dials to balance the performance margins across the mission profile.

Intermediate Orbit Period

After performing the PRM, the ICPS executes an ARB to place the Orion spacecraft into a highly elliptical intermediate orbit. This maneuver is executed near perigee and raises apogee significantly. The baseline EM-2 mission sets the period of the intermediate orbit to 24 hours. The analysis team was interested in understanding how increasing the intermediate orbit period to 42 hours would impact mission performance. The 42 hour orbit was selected for study because it is the shortest orbit period after 24 hours in which the Orion crew could then nominally execute their Trans-Lunar Injection burn without interfering with other crew activities, such as the crew sleep period. Here, a greater orbit period translates into a higher intermediate orbit apogee. Also note that a higher intermediate orbit period (and therefore, apogee) implies that Orion will require less propellant to execute its TLI maneuver. As with other analyses, only the period of the intermediate orbit was varied in this study. No other parameter was varied from the baseline EM-2 mission. It is both interesting and significant to note here that the additional amount of ICPS propellant required to place Orion into the intermediate orbit is only slightly higher (~1500 lbm) than is gained by increasing the Core stage insertion orbit apogee to 1200 nmi, as shown in the previous study. Coupled together, these two parameter changes imply that mission analysts can effectively maintain existing propellant margins on ICPS while providing Orion with additional propellant margin. In effect, these two options (i.e., raising Core stage insertion apogee altitude and increasing the intermediate orbit period) together make use of both the Core state and ICPS capabilities to increase Orion propellant margin.

First Revolution Apogee Raise Burn

In the baseline EM-2 mission, the apogee raise burn (ARB) occurred on the second perigee pass of the mission. That is, after performing the PRM, ICPS coasts for more than one full orbit period before executing the ARB. However, multiple passes through apogee can non-trivially increase MMOD risk to the spacecraft. Therefore, it was of interest to the analysis team to understand how executing the ARB on the first perigee pass would impact mission performance. Iniitally, the primary constraint of the “First Rev ARB” case is that Orion requires a minimum of two hours after the PRM completes and before the ARB starts in order to check out the spacecraft prior to inserting into the intermediate orbit. The only way to achieve a coast duration of this magnitude is to use the PRM maneuver to significantly raise apogee while also achieving the required 100 nmi perigee altitude. Analysis showed that executing the PRM two minutes after Core stage separation and targeting a post-PRM apogee of 3140 mi would provide the required coast time, although at a fairly substantial impact to ICPS propellant usage at greater than 1700 lbm and a non trivial impact to Orion propellant usage at about 440 lbm.

NOMINAL HYBRID FREE RETURN ON SLS BLOCK 1 MISSION SCANS

Feeding off the parametric analysis, the EM-2 Block 1 nominal and abort performance analysis focuses on mission availability for nominal missions and maximizing positive abort capability for off-nominal missions. The nominal performance assessment covered a scan duration of 1 year from June 7, 2022 through June 7, 2023. The scan design produces available mission epochs given vehicle performance capability and (preliminary) operations constraints such as: available ICPS and Orion translational propellant mass, mission propellant offloading, EI to splashdown downrange limits, solar eclipse duration limits, launch and landing lighting limits. These mission parameters and constraint limits are evaluated for variations in propellant offloading, post-MECO orbit apogee (with some perigee variation), and pre-TLI-2 HEO duration to assess the benefits or impacts that varying these parameters have on the nominal and abort capability.

As compared to the Block 1B configuration, Block 1 scans reflect a smaller population of available nominal missions, in a given year, than the Block 1B scans, due in large part to the greater performance capability of the EUS (Block 1B) vs the ICPS (Block 1). The EUS capability allows it to complete the ascent phase culminating in a circular (e.g., 100x100nmi) LEO with enough capability to subsequently perform an ARB of the entire EUS/CPL/Orion stack to a selected HEO and (after separation of the EUS/CPL stack from Orion) a subsequent TLI to a lunar flyby with a targeted heliocentric disposal destination. The EUS capability allows mission opportunities with either an ascending or descending node Earth departure trajectory. In contrast, the reduced performance of the ICPS forces a heavier reliance on the SLS Core Stage to deliver the ICPS/Orion stack to a higher energy, elliptical orbit (e.g., 975x22nmi to 1200x17nmi), which limits the available translunar trajectories to only the ascending nodes, as exhibited with the EM-1 mission design, which also uses the ICPS. The loss of descending node Earth departure availability with the ICPS (Block 1) accounts for reduced mission opportunities as compared to the more capable EUS (Block 1B) configuration. Mission Trades

The lesser performing ICPS necessitated a partial mission redesign and associated re-optimization of the trajectory(ies). The initial EM-2 Block 1 baseline mission design, based on the EM-1 crew study results, laid out the first candidate mission design. It included the following fundamental design parameters:

- Orion Mass: Not to Exceed Mass Allocation
- Orion Prop Load: 300 lbs offload (meet mass target)
- ICPS Performance: Per EM1 TLI data
- LEO Orbit: 2 revolutions (revs) at 975x100 nmi altitude
- Orion Array Deploy: Pre-PRM
- Secondary Payload: As available (not included in initial assessments)
- HEO Period: 24 hours

Table 1. Mission Design Options

Option	Traj. Case Num.	Insertion (nmi)	PRM Timing	Resulting LEO Apogee (nmi)	ARB Rev Start	HEO Period
A	1	975	apogee	975	2nd	24 hr
B	2	975	10 min	1200	1st	24 hr
	3	1200	apogee	1200	2nd	42 hr
C	4	1200	10 min	1450	2nd	42 hr
D	10	1200	10 min	1450	1st	42 hr
	5	1200	2 min	variable	1st	42 hr
	6	1200	2 min	3100	1st	42 hr
	7	975	apogee	975	1st	24 hr
E	8	1200	2 min	3100	1st	24 hr
F	9	1200	2 min	2000	1st	42 hr

Performance trades for this baseline resulted in the evolution of new (mission design) cases and subsequent performance trades to explore refined changes in the candidate mission designs. Performance trade studies provided mission availability with selected mission constraints for a number of mission design variations. This trade space (see Table 1) of 10 individual trajectory cases (9 identified cases with 1 case possessing 2 sub-options) provided a relatively comprehensive examination of the combination of potential mission design parameters via a set of performance trade studies intended to determine the number of viable nominal missions for a given mission design. Through the assessment of these "Traj. Case" mission candidates evolved a set of potential mission design options (i.e., Options A-F). Trajectory case 5 was intentionally left out as it ended up "seeding" case 6's LEO apogee target. Case 5 originally optimized the post-PRM apogee day to day, but issues with the optimization and time constraints on the trade study forced the team to abandon this case and move on to fixing the post-PRM apogee for case 6. Trajectory cases 3 and 6 were not selected for further refinement, so they were not sent forward as options.

Variations in selection mission parameters resulted in competing effects, benefiting or impacting different systems, thus making it a challenge to balance overall mission risk. An increase of the baseline 975 nmi (Core stage) insertion apogee to 1,200 nmi provides improvement in mission availability through decreased demand on ICPS performance and the use of existing Core stage ascent margin. The reduction of the time of execution of the PRM (from the baseline apogee passage to 2 or 10 minutes after Core stage MECO) results in an increase in LEO apogee and associated orbit period, which provides for a single rev in LEO prior to ICPS ARB. The increased orbit period provides reduced exposure to the MMOD field (by reducing LEO from 2 revs to 1 rev) while still providing sufficient time for Orion operations checkout (e.g., solar array deploys in a stable orbit with ample settling time to the next burn, a longer "quiescent" period for operations checkout). However, the early PRM also results in a decrease in mission availability due to ICPS performance impact. The number of post-PRM revs (1 vs 2) prior to ARB start, while decreasing MMOD field exposure, also decreases LEO operations checkout time. That said, a success-oriented mission could tolerate durations greater than 60 minutes and the single rev cases produced durations longer than 60 minutes. With regards to HEO size, expansion of the HEO period from 24 to 42 hours provides for a significant reduction in the Orion (TLI-2) propellant requirement, resulting in greater post-TLI-2 Orion abort capability due to greater propellant reserves. The larger HEO based mission results in slightly more frequent mission opportunities.

Trajectory Scan Methodology

The trade studies examined required the implementation of large trajectory scans. A one-year scan of nominal missions necessitated the use of (Python) scripting algorithms to manipulate the input decks (idecks) of the trajectory optimization software (i.e., Copernicus). The scripts allow trajectory analysts to automate the trajectory optimization process. However, though automated, the convergence of the year's worth of trajectory cases will take a lot of computer time. A well-tuned mission ideck could still require a day or so of computer execution time for a nominal mission set. After convergence of the trade study cases, a data extraction process translates selected data (e.g., mission epoch(s), vehicle and propellant mass data, mission phase durations, trajectory targets, orbit parameters, spacecraft thrust, burn propellant used, launch and landing lighting data, lunar transfer data, solar eclipsing data, etc.) into a spreadsheet format for further assessment.

Preliminary (orbit) parameter sensitivity studies help analysts to provide the trajectory optimization tool with settings that encourage good performance optimization. For example, previous studies showed that an equally weighted multi-objective function for ICPS propellant requirement with the Orion propellant requirement provides good performance for Orion while allowing for a reasonably good number of mission opportunities.

The construction of the ideck for a given mission design at a given launch epoch combines both the SLS (ascent to LEO) and Orion (TLI-2 and beyond) contributions to the mission as an integrated mission. The Copernicus tool provides the platform for (mission opportunity) trades study scans. To achieve an integrated mission design, the Copernicus ideck includes a "Hypergrid" data set that reflects ascent from launch to Core stage MECO and provides to the Copernicus simulation the Core stage MECO state and time as well as other items (e.g., deliverable ICPS and Orion mass, ascent margin for the Core stage). Copernicus uses the Hypergrid data to perform an integrated ascent/on-orbit trajectory optimization. Copernicus can interact with the Hypergrid as it optimizes the integrated trajectory. For example, Copernicus may use ascent margin in the Core stage to increase LEO apogee to reduce the propellant requirement on the ICPS or adjust the launch azimuth to maximize the duration of the launch window for a give mission day.

Numerical trajectory optimization tools are notoriously sensitive to initial guesses for the problems they are working to optimize. A good initial trajectory "guess" will provide for better, faster convergence to an optimal solution for the overall trajectory. For performance trade studies, the trajectory scan setups use a "continuation method" whereby the event times and control parameter (optimization variables) values are shifted from a fully converged trajectory to one that is nearby in time (thus in orbit geometry). If the time step from a converged trajectory to the next desired trajectory epoch is sufficiently small to keep a sufficiently similar trajectory profile, then the optimizer will work the new problem to final converged optimization. The continuation method does provide a way to keep updated idecks for the next epoch in succession (in a scan) sufficiently well behaved that the optimizer can drive the initial guess to a final converged, optimal result. However, it does require sufficiently small enough jumps in epoch so as not to disturb the overall converged trajectory shaping.

Ongoing work at Johnson Space Center focuses on an initial guess generator (IGG) that uses semi-analytic orbit propagation algorithms to create a feasible trajectory at any given epoch. With a working IGG, the analyst could select any epoch-to-epoch magnitude shift for trajectory scans. This allows the analyst to focus on just the trajectories of interest without need to create less useful interim trajectories to keep to a small enough epoch shift so as to maintain trajectory optimization convergence.

After completion of a trajectory scan and subsequent data extraction, the initially unrestricted trajectory data can be filtered to reflect vehicle and/or operational constraints. The analyst can compare and contrast the number of nominal mission opportunities in a (for example, one-year) trajectory scan for an unrestricted dataset to one with added mission and vehicle configuration cutouts (e.g., ICPS usable propellant expended, Orion usable propellant expended, EI to splashdown downrange limits, eclipse duration exceedances, launch and/or landing lighting constraints, etc.). The post-processing of selected mission constraints allows the analyst to quickly assess the (mission availability) impact magnitude attributed to the various constraints.

Trajectory Case Descriptions

Four key factors were found to influence the results: insertion apogee, PRM timing, number of revolutions in LEO, and HEO period. Analyzing various combinations of the aforementioned factors led to six viable options (A-F).

Assumptions for the Mission Availability and Performance Analysis

- Orion has a maximum useable translational propellant mass.
- ICPS has a maximum useable translational propellant mass.
- ICPS performs a lunar flyby to heliocentric disposal, targeting a minimum C3 of 0.35 km^2/s^2 (with respect to the Earth-Moon barycenter) 10 days post Lunar Gravity Assist (LGA). The lunar flyby altitude is 260 km.
- Orion spends approximately 1 revolution in HEO.
- The Orion lunar free return flyby (from TLI-2 to EI) is approximately 8 days.
- The Orion Main Engine (OME) is used for TLI-2.
- The ICPS is used for the PRM, ARB, and TLI-1.
- All burns are modeled as finite burns.
- The objective function minimizes both Orion and ICPS propellant used. They are weighted evenly.

Mission Availability and Performance Analysis

Mission availability, LEO coast time, abort return time, and MMOD risk were used to compare the trajectory options. There are several vehicle checks that need to be performed while in LEO. This check-out time has a minimum of 60 minutes and a desired check-out time of 120 minutes. MMOD risk is tied to LEO as well. The more revolutions in LEO, the higher the MMOD risk. Furthermore, the larger LEOs tend to have a lower MMOD risk. The timing of the PRM is linked to the size of the LEO. The earlier the PRM, the higher the LEO apogee altitude. PRM timing also plays a role in Orion power generation. The sooner the PRM is executed after insertion, the sooner Orion can deploy the solar arrays. All of these need to be taken into consideration along with impacts to mission availability and the abort space in order to select the best mission design.

Option A had the highest mission availability of the cases examined. There were between 12-14 feasible launch opportunities per month. The feasible mission opportunities tend to be consecutive days each month. The average coast time in LEO was 190 minutes, which was greater than the desired LEO check-out time. This option has the PRM occurring approximately 45 minutes after insertion, which is more stressing for Orion power generation. Additionally, the 24 hour HEO causes Orion to use more propellant for TLI-2, which reduces the available propellant for aborts. Further, the lower LEO apogee altitude along with spending two revolutions in LEO increases the MMOD risk.

Option B was an attempt to provide Orion with more continuous checkout time in LEO by moving the PRM earlier in the LEO coast, or 10 minutes after the ICPS separation from the Core stage, while still performing the ARB near the first perigee passage. Checkout time is reduced in LEO from Option A to 78 minutes on average, or a minimum of 64 minutes across the yearly scan. For each lunar month, the launch period was also reduced to 10-11 days. The aborts situation remained similar to Option A due to retaining the 24 hour HEO. However MMOD risk was somewhat reduced due to reduction to 1 revolution in LEO.

Option C had similar mission availability to options B and D, with between 10-11 feasible launch opportunities per lunar month. The average LEO coast time was 211 minutes, which meets the desired LEO check-out time. For this option, the PRM was completed 10 minutes after insertion, which allows for the Orion solar arrays to be deployed early in LEO. Option-C used a HEO period of 42 hours. The larger HEO reduces the amount of propellant Orion uses for TLI-2, which increases the available abort propellant. Furthermore, like option-A, the stack spends two revolutions in LEO. However, the LEO apogee altitude is higher than option-A, so the MMOD risk is reduced slightly.

Options C and D are identical, accept option-D spends only one revolution in LEO. The mission availability was the same as option-C, with 10-11 feasible launch opportunities per month. With only doing one revolution in LEO, the average LEO check-out time was reduced to 81 minutes. This LEO time falls between the minimum and the desired check-out times. Option-D has the same benefit for Orion post-TLI-2 abort propellant available as option-C, since both use a 42 hour HEO period. In addition, the higher LEO apogee altitude along with spending only one revolution in LEO reduces the MMOD risk.

Option E provided the lowest MMOD risk of any of the assessed options, however it also had the lowest mission availability. The 24 hour HEO impacted the Orion abort capability in a similar manner to Options A and B. This option was designed to meet the desired Orion checkout time of 120 minutes, which is a result of the 3100 nmi apogee induced by the PRM at 2 minutes after Core separation. The scan for this Option did not run as well as the scans for the other options, however the trade team realized that this option would not be palatable due to the low mission availability. The results shown could have been improved on somewhat, but the team decided to work Option F instead.

Option F was a compromise between Options D and E in an attempt to get to a middle ground on checkout time, while retaining the best of the previous two options. It resulted in a reduced mission availability from Option D, but improved the MMOD risk and Orion abort capability due to performing the ARB on the first revolution in LEO and the 42 hour HEO period respectively. It also achieved a minimum of 90 minutes of checkout time in LEO, which was a compromise proposed by the Orion program.

Table 2. Launch Availability Summary

Option	Number of Days Over 1 year	Min/Max Launch Period Duration (days)	Average Launch Period Duration (days)	Min/Max Coast Time (min)	Average Coast Time (min)
A	195	13/15	14.3	166/193	183
B	139	10/11	10.2	64/86	78
C	138	9/11	10.2	202/216	211
D	149	10/12	10.9	66/88	88
E	>91	5/8	7.2	120/136	130
F	120	7/9	8.7	90/123	105

A summary of the mission design outputs for all options is provided in Table 2. The coast time for Option A represents the total time in LEO, including the coast to the PRM at apogee. The coast to apogee is approximately 45 minutes, so some Orion checkouts can be performed during that time. All other options utilize an early PRM, so the coast time starts at the end of the PRM as 10 minutes may not be sufficient time to perform any checkouts. A final scan was run on Option D to closeout the trade and resulted in slightly better numbers than Option C on launch availability. The analysts expect that with further refinement, Option C would end up being very close to Option D with respect to launch availability.

ORION ABORTS ASSESSMENT

An analysis of the abort space was performed for three phases of the mission: LEO, HEO, and post-TLI-2 to lunar flyby. The general abort return time constraint is 6 days from abort declaration to splashdown, which allows 24 hours to develop an abort plan. This leaves a maximum return time of 5 days, which was the target for this analysis. The driver for this return time would be a cabin depressurization where the astronauts would be required to stay in their suits. Too much time in a spacesuit can lead to medical issues (e.g. sepsis), so it is important for the vehicle return to Earth as quickly as possible.

Assumptions for the Abort Study

The following are the basic assumptions that apply to the abort analysis for all three phases of the mission:

- Return time from abort declaration to entry interface was minimized. This time included an

Orion coast prior to performing the maneuver. For several cases, Orion coasted to a more optimal location for the abort maneuver causing an overall decrease in return time.

• The OME was used for the abort maneuvers.

• Orion was allowed to burn all available remaining propellant after removal of the 136 kg (300 lbm) propellant offload, flight performance reserve, Outbound/Return Trajectory Corrections, Orbit Maintenance, Attitude Control System, Worst 1 Fault Tolerant Allocation, and Burn Time Integration.

Assumptions for the HEO Abort Study

The following are the basic assumptions for the HEO abort study:

• A six month scan was conducted that ran from June – December 2022.

• The aborts targeted the EI target line, which sets up Orion to splashdown off the coast of San Diego.

• Two HEO periods were examined: 24 and 42 hour. The 24 hour HEO used a 975x100 nmi LEO and the 42 hour HEO used a 1200x100 nmi LEO.

• Both cases stay in the HEO for approximately one revolution.

24 hour HEO Aborts

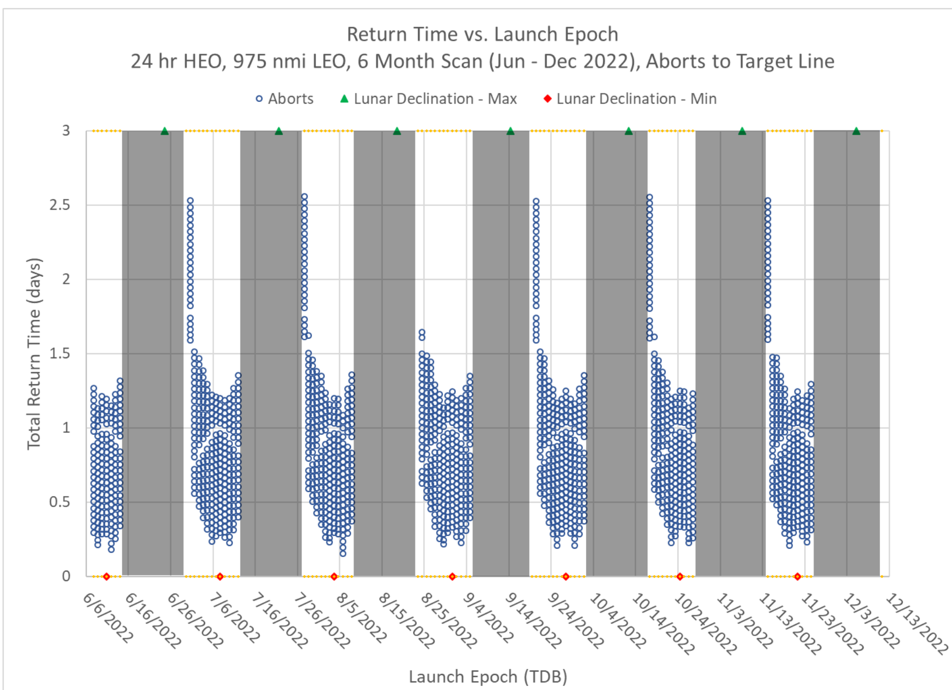

Figure 4. 24 hr HEO total return time vs. launch epoch. Shaded regions show unavailable epochs due to violation of the maximum allowable ICPS propellant limit.

Aborts to the target line were available for about half of each month (Fig. 4). There is a natural alignment between the non-available HEO aborts and otherwise unavailable epochs (due to violation of the maximum allowable ICPS propellant limit). The abort return duration (abort declaration to EI) ranged from 0.151 – 2.561 days, with a mean return time of 0.885 days.

42 hour HEO Aborts

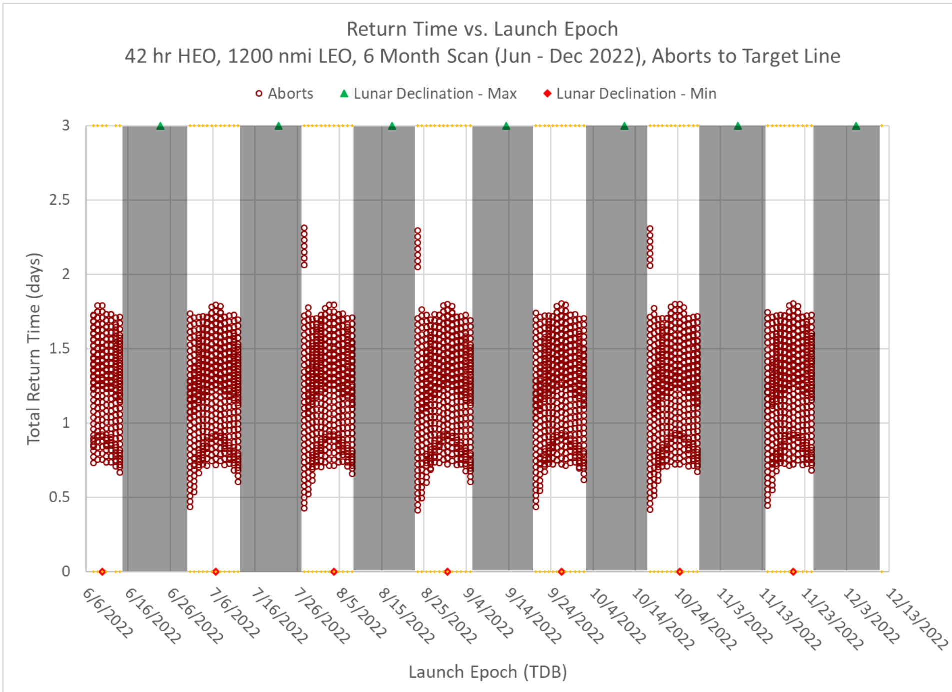

Figure 5. 42 hr HEO total return time vs. launch epoch. Shaded regions show unavailable epochs due to violation of the maximum allowable ICPS propellant limit.

Aborts to the target line were available for every epoch throughout the 6 month scan (Fig. 5). The epochs with the longest return times align with the unavailable epochs (due to violation of the maximum allowable ICPS propellant limit). The abort return duration (abort declaration to EI) ranged from 0.414 – 2.315 days, with the mean return time of 1.244 days.

Assumptions for the Post-TLI2 to Lunar Flyby Abort Study

The following are the basic assumptions for the post-TLI2 to lunar flyby abort study:

• The aborts targeted the EI target line, which sets up Orion to splashdown off the coast of San Diego.

• Post-TLI2 aborts for two HEO periods were examined: 24 and 42 hour. The 24 hour HEO used a 975x100 nmi LEO and the 42 hour HEO used a 1200x100 nmi LEO.

• Both cases stay in the HEO for approximately one revolution.

• Both direct and lunar flyby returns were analyzed.

• The feasible missions (missions that meet the ICPS propellant limit) of the worst performing month (least amount of post-TLI2 Orion propellant available) for 24 and 42 hour HEO scans were examined. The worst month for the 24 hour HEO missions was May 2024. The worst month for the 42 hour HEO missions was June 2024.

24 hour HEO Post-TLI2 Aborts

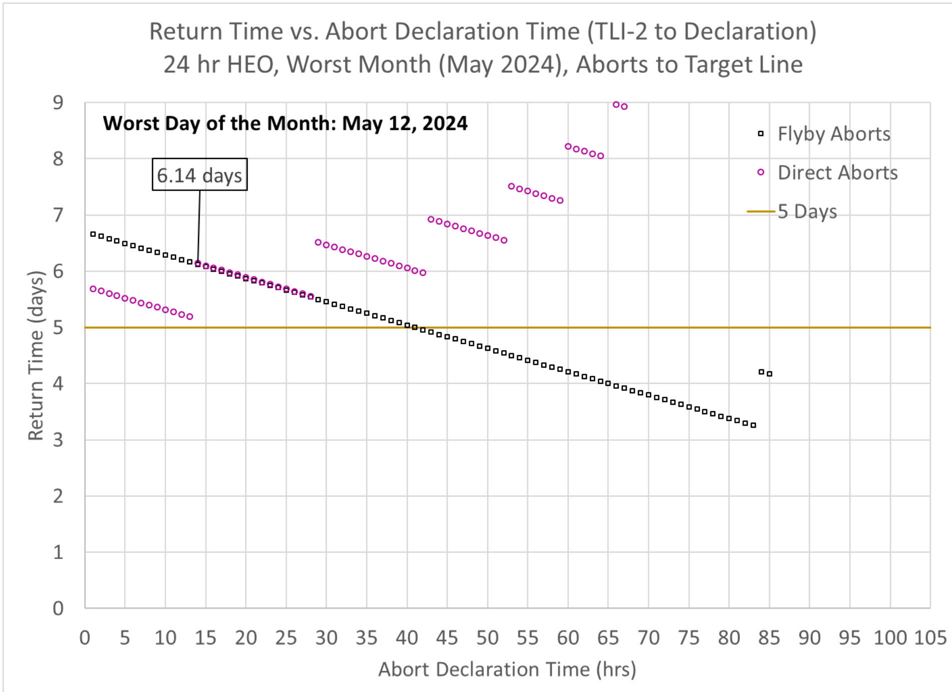

Figure 6. 24 Hour HEO Post-TLI2 abort return time vs. abort declaration time from TLI-2. Worst performing day for Orion.

The maximum return times ranged from 6.14 days on the worst day to 4.77 days on the best day of the worst month. For the first 14 hours after TLI-2, a direct abort was the quickest option to return to Earth on the worst day for Orion performance (Fig 6). For the best day case (Fig 7), a direct abort was the quickest return option for the first 26 hours after TLI-2. After that point, doing a lunar flyby return was faster. The worst day of this scan shows that there are no returns under the 5 day limit until approximately 40 hours after TLI-2.

42 hour HEO Post-TLI2 Aborts

The maximum return times ranged from 5.44 days on the worst day to 4.52 days on the best day of the worst month. For the first approximately 36 hours after TLI-2, a direct abort was the quickest option to return to Earth on the worst day for Orion performance (Fig 8). For the best day case, a direct abort was the quickest return option for the first 39 hours after TLI-2 (Fig. 9). After that point, doing a lunar flyby return was faster. The worst day of this scan shows that there are no returns under the 5 day limit for about a 10 hour block starting approximately 37 hours after TLI-2.

MICROMETEROID AND ORBITAL DEBRIS ANALYSIS

MMOD risk has increased steadily since the beginning of manned space ventures. The source of this increase is the orbital debris portion of the risk. Figure 10 reveals the growing trend for tracked orbital debris objects with sizes of 10 cm or greater. NASA's Orbital Debris Program Office predicts there are over 500,000 objects 1 cm and larger in Earth orbit—a scale factor of over 30 compared to trackable objects. The deliberate destruction of the Fengyun-1C Chinese weather satellite in 2007

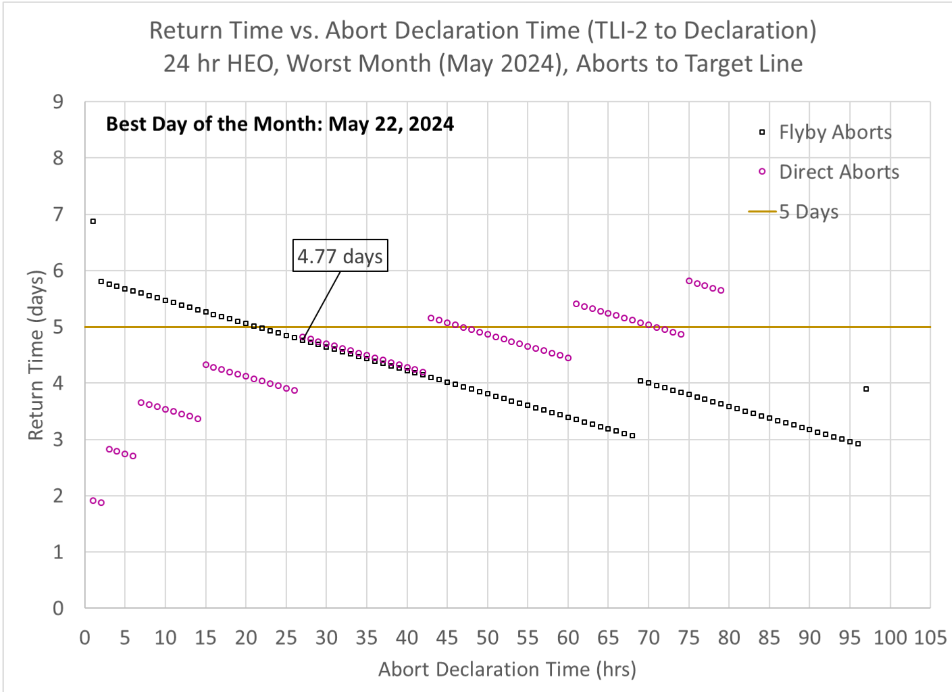

Figure 7. 24 Hour HEO Post-TLI2 abort return time vs. abort declaration time from TLI-2. Best performing day for Orion.

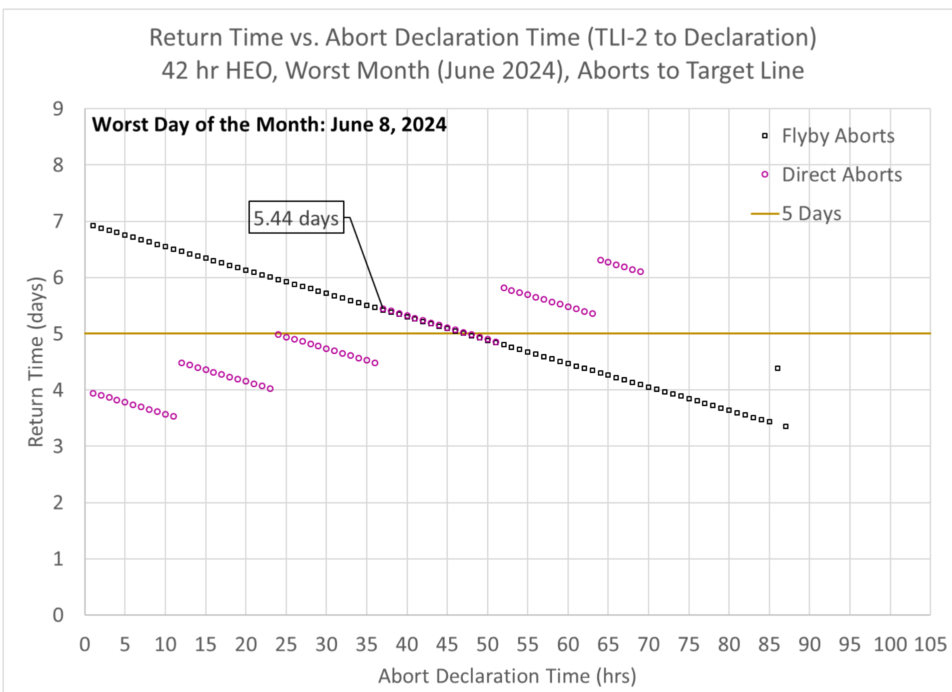

Figure 8. 42 hour HEO Post-TLI2 abort return time vs. abort declaration time from TLI-2. Worst performing day for Orion.

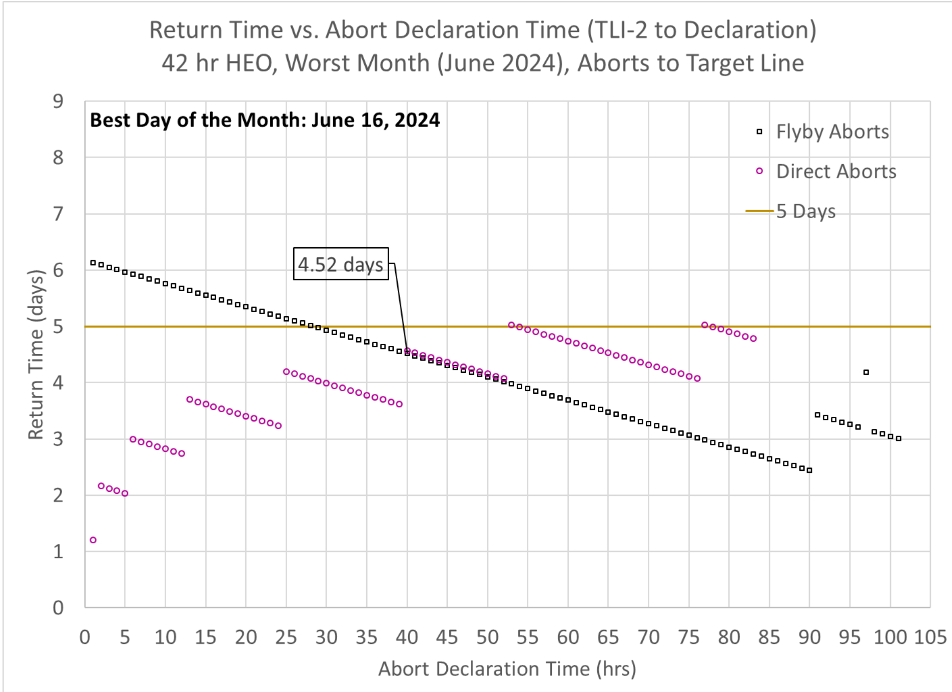

Figure 9. 42 hour HEO Post-TLI2 abort return time vs. abort declaration time from TLI-2. Best perfoming day for Orion.

was a major contributor, and the accidental collision of the Iridium 33 and Cosmos-2251 satellites in 2009 also boosted the current risk. With more than 2.5 million kilograms of objects now in Earth orbit, such accidental collisions are projected to be a major source of orbital debris in the future.

The increased MMOD risk was investigated in 2015 for the EM-2 mission utilizing the Block 1B configuration of SLS. This study resulted in the reconfiguring of the EM-2 mission profile into a LEO apogee of 100 nautical miles (nmi) to remain under the main orbital debris bands until TLI. These regions of greatest orbital debris density are in the altitude range of 300 to 1000 nmi and cause the meteoroid influence to be very low by comparison. Unfortunately, the SLS Block 1 configuration with the ICPS does not have the performance to achieve this orbit profile. As a result, the trade study described herein was executed to minimize MMOD risk while maintaining acceptable performance and other associated risks for the combined SLS and Orion system.

The MMOD risk was determined utilizing a model consisting of the Bumper 3 (Ref. 7) simulation code with the appropriate embedded elements of ballistic limit equations (BLE's) as well as the ORDEM 3.0 orbital debris environment and the MEM r2 2.0.2.1 micrometeoroid environment representations. This tool set, along with a finite element representation of the ICPS and specified flight trajectory parameters, estimates the risk of a damaging strike due to micrometeoroids or orbital debris during flight. Bumper calculates the number of failures in a deterministic fashion by computing the number of MMOD particles that exceed the ballistic limits for each element of the Finite Element Model (FEM), and calculates the total number of failures by summing the individual elements. An estimate of the probability of mission failure is determined from the total number of failures using Poisson statistics. The Bumper code has been the standard in use by NASA and contractors to perform meteoroid/debris risk assessments since 1990. NASA Johnson Space Cen-

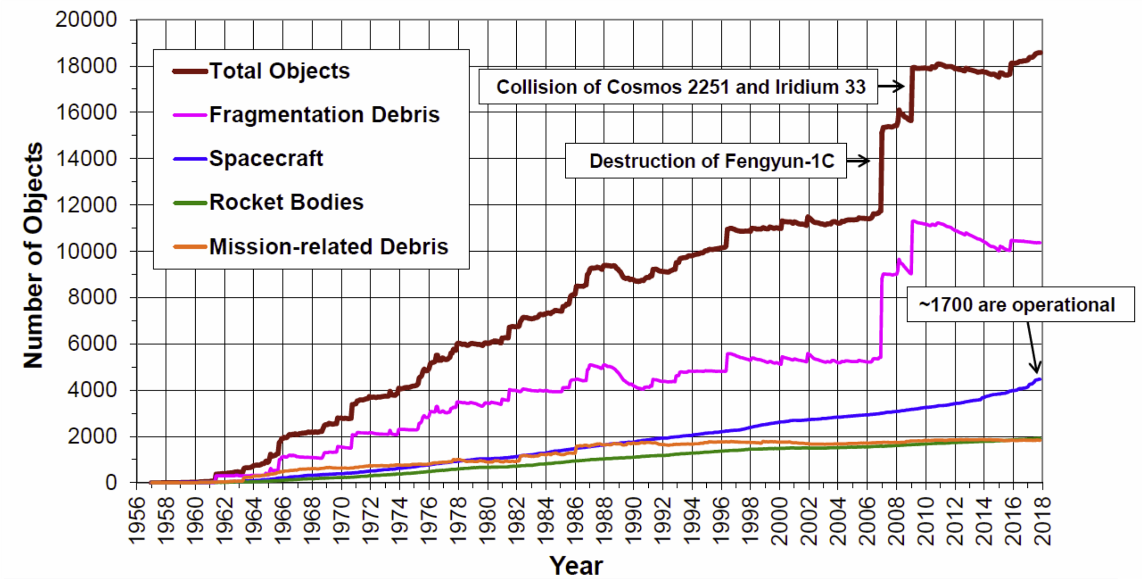

Figure 10. Number of Objects in LEO (greater than 10cm), 1956-2018 (Ref. 6)

ter (JSC) has applied BUMPER to risk assessments for Space Station, Shuttle, Mir, Extravehicular Mobility Units (EMU) space suits, and other spacecraft (e.g. LDEF, Iridium, TDRS, and Hubble Space Telescope).

An overarching assumption associated with the use of this model is that the ORDEM 3.0 and MEMR2 files properly define the debris and meteoroid environments, respectively. The validation of ORDEM 3.0, the driving environment to quantify MMOD risk for ICPS, is covered in section 3 of Reference 8. MMOD modeling limitations are detailed for the Bumper code and the two environment files within their respective supporting documentations. Section 2.3 of Reference 7 outlines the limitations associated with the modeling of the MMOD strikes. Section 1.3 of Reference 9 includes the limitations associated with the representation of the meteoroid environment while section 1.2 of Reference 10 discusses the limitations with regard to the representation of the orbital debris environment. But other assumptions which drive the accuracy of the assessments are related to the selected failure criteria for each component modeled. The following paragraph details the selection of the failure criteria for the ICPS systems—the primary contributors to the overall system Loss of Mission (LOM) risk.

Hypervelocity impact testing at NASA's White Sands Test Facility (WSTF) provides the primary source of failure criteria data. The results of these tests provide general insight as well as a source for the tuning of damage estimate algorithms. For ICPS four pressurized containers were assumed to contribute to MMOD strike risk: the LOX tank, the LH2 tank, the Helium tanks, and the Hydrazine tanks. The LOX, LH2 and Hydrazine tanks were modeled as metal tanks with a partial penetration failure criteria based on testing which has shown that this level of penetration initializes spalling on the interior surface of the tank wall. This spalling is assumed to either weaken the pressure vessel sufficiently to allow the propagation of cracks or to generate contaminating debris inside the tanks. Either of these conditions can lead to LOM. The Helium tanks are composite overwrapped pressure vessels (COPVs) with graphite-epoxy wrappings over a metal liner. Prior analyses assumed failure criteria to be penetration to the outer surface of the liner. Validation testing of this assumption at

WSTF revealed that the tanks retained structural integrity and did not leak until the liner was also penetrated. So penetration through the COPV liner became the updated LOM failure criteria for the ICPS COPVs. It should be noted that COPVs on ICPS are connected such that a leak in one tank will deplete the complete system. Assorted additional assumptions include the following. The engine nozzles were assumed to be relatively robust with regard to MMOD strikes, so they were not included as risk contributors. Also, the total exposed area of pressurized piping as well as avionics was assumed negligible and therefore not modeled. The MMOD risk to Orion and the Service Module was assessed separately from ICPS, but with the same modeling approach. In both the Orion and SLS assessments, the structures of the neighboring systems were modeled as shielding.

Option	Traj. Case Num.	ICPS Risk (1/x)		Orion Risk (1/x)		Total Risk (1/x)	
		LOM	LOC	LOM	LOC	LOM	LOC
A	1	185	70,800	793	879	150	868
B	2	369	136,000	1,090	1,200	276	1,190
C	4	254	92,900	827	913	194	904
D	10	536	183,000	1,240	1,390	374	1,380
E	8	762	291,000	1,080	1,160	447	1,160
F	9	603	218,000	1,160	1,270	397	1,270

Figure 11. MMOD Analysis Results

The MMOD results for the various trades performed for the EM-2 study are detailed in Figure 11. The ICPS risk covers the period between separation from Core stage and separation from Orion. The Orion risk covers the entire EM-2 mission profile. As shown, the LOM risk is driven by the ICPS risk, while the Loss of Crew (LOC) risk is largely determined by the Orion risk. The ICPS LOC risk is based on assuming that the service module can successfully abort Orion during a leak from the LOX, LH2 or hydrazine tanks. These risks are quantified using an algorithm developed by Schonberg (Ref. 11) during an extensive review of tank burst testing. Schonberg's algorithm reveals that tank pressure is a major driver in determining burst probabilities. The relatively low pressure levels of these tanks result in very low burst risks. The helium COPV tanks are at high pressures, however, and testing has shown that resulting leak rates are sufficient to exceed the ICPS reaction control system. As a result, an MMOD strike sufficient to cause leakage in a COPV is considered the failure criteria for both LOC and LOM. The COPV testing also revealed, however, that the risk of COPV leaks were much lower than previously estimated. As a result, the LOC risks for COPV remained very low.

As previously detailed, the LOM failure criteria for the tanks other than the COPVs on ICPS are based on interior spalling. This criteria in conjunction with mission profiles flying through the main debris regions resulted in estimates of unacceptable LOM risks. In particular, the Option A mission configuration, which represents the baseline orbit previously established for an EM-2 mission using a Block 1 configuration, resulted in an estimated MMOD LOM risk of 1 in 185 and LOC risk of 1 in 868. This baseline orbit is a two orbit LEO with an apogee of 975 nmi and therefore traverses the highest debris density bands for extended periods. Option B retains a relatively low apogee (1200 nmi) but eliminates the second LEO revolution. This improved the MMOD risk to 1 in 276 for LOM and 1 in 1,190 for LOC, but a reduction in check out time due to the single orbit made this an unacceptable option. Option C represents an attempt to reduce the MMOD risk by extending the apogee to 1450 nmi, but since it retained the two revolution LEO for check out purposes, the risk estimates for LOM were back to unacceptable levels—1 in 194. As anticipated, the two revolution LEO profiles caused multiple passes through the high debris density regions resulting in unacceptable LOM results. In an attempt to alleviate this impact, the remaining trades eliminated the second revolution in LEO. Options E and F represent attempts to reduce MMOD further and extend check out times with highly elliptical orbits. The theory is that these orbits would cause optimal orientations and reduced high density debris exposure times for the ICPS but approach the flight times of the lower apogee two revolution missions. The results show that these approaches did reduce the LOM levels. The 1/447 LOM for the 3,100 nmi apogee was the best seen during the trade. The launch availabilities were shown to be unacceptable, however, so Option D was reviewed with a very good balance of MMOD risk with the other factors. This option, with an apogee of 1450 nmi and a single revolution LEO, resulted in a LOM estimate of 1/374 and a LOC of 1,380. The LOC was the lowest crew risk in the study as a result of an optimal balance of ICPS LOM and Orion LOC factors.

FINAL RECOMMENDATIONS AND OUTCOME

The trade study concluded that Option F was the best choice of mission design for EM-2, with Option D coming in a close second. These two options balanced the overall mission risk when everything was taken into account. Option F had the second best MMOD risk overall at the cost of lower launch availability. Option D had slightly higher MMOD risk compared to Option F, but had greater mission availability. Ultimately, the Joint Integration Control Board (JICB) selected the Option D mission design for use in the EM-2 Mission Analysis Cycle (MAC) due to the higher mission availability. The JICB felt that the change in MMOD risk was not significant enough between the two options to sacrifice up to 3 days of mission availability per lunar month. The trajectory analysis teams at Marshall Space Flight Center and Johnson Space Center have since produced an initial reference trajectory for the SLS EM-2 MAC that will be used by both Orion and SLS to confirm the mission design fits within the capability of both vehicles.

	Insertion (nmi)	PRM (min)	Apogee (nmi)	Rev	HEO (hrs)	Msn Avail (dd/mm)	Total PRA (1 in x LOC)	MMOD Stack (1/x LOM/LOC)	Abort Capability	LEO Ops/Perf.
A	975	45	975	2	24	12-14	188	150/868	Degraded post-TLI	Required + Desired + IVP Stressing
B	975	10	1150	1	24	10-11	199	276/1,190	Degraded post-TLI	Required only
C	1200	10	1450	2	42	10-11	189	194/904	Improved post-TLI	Required + Desired
D	1200	10	1410	1	42	10-12	~205	374/1,380	Improved post-TLI	Required + IVP benefit
E	1200	2	3100	1	24	5-8	198	447/1,160	Degraded post-TLI	Required + Desired + IVP benefit
F	1200	2	2000	1	42	7-9	201	397/1,270	Improved post-TLI	Required + Desired + IVP benefit

Figure 12. Final Trade Summary

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