(Preprint) AAS 20-589

SPACE LAUNCH SYSTEM ENGINE OUT CAPABILITIES

A. S. Craig[*], J. E. Hauglie[†], M. J. Hawkins[‡], J. J. Hays[§]

NASA's Space Launch System (SLS) is being developed with the primary purpose of returning people to the Moon and eventually landing people on Mars. With these lofty goals, ensuring mission completion is paramount even in the event of an in-flight mishap. One possible mishap is the loss of an engine in flight. While SLS was not required to show full engine out capability, the program took an "assess to" approach to see when the launch vehicle could complete the mission after an engine failure versus when the launch vehicle targets required a down-mode to an alternate mission target to ensure at least some flight objectives were complete, or at a minimum ensure safe return of Orion and the Crew. While this paper will focus on Artemis I, an uncrewed mission, some comparisons will be made to how the engine out capability will change for the subsequent Crewed flights of SLS and Orion.

INTRODUCTION

Very early in the SLS program, the decision was made to utilize Space Shuttle and Constellation developed hardware to the greatest extent possible. This meant that the Core would be similar to the Shuttle External Tank, the Solid Rocket Booster (SRB) would be derived form the Constellation 5 segment variant of the Shuttle 4 segment SRB's, and the Core engines would be the Space Shuttle Main Engine (SSME), otherwise known as the RS-25's. While the RS-25's have shown significant reliability, there is one flight, STS-51-F in 1985, where an engine shutdown due to a sensor failure. This required the Space Shuttle to perform an Abort to Orbit (ATO) and achieved a lower than planned orbital altitude.[1] Despite the multiple successful launches without an in-flight engine shutdown, National Aeronautics and Space Administration (NASA) will still prepare for the worst and have plans in place to correct for unplanned engine shutdowns.

The SLS Launch vehicle has four RS-25's on the Core and analysis has shown that the Block 1 configuration can recover from an unplanned engine shutdown in almost all phases of flight. Late in flight, SLS would press to Main Engine Cutoff (MECO) and continue on its nominal mission profile. Two other targets, an High Energy Alternate MECO Target (AMT-HI) for engine failures in the middle portion of flight, and an Low Energy Alternate MECO Target (AMT-LO) target for failures very early in flight rounds out the engine out capability of SLS. For Artemis I, the AMT-HI target was derived to ensure Orion could meet its high-speed reentry test objective for its heat shield by inserting the Interim Cryogenic Propulsion Stage (ICPS) and Orion stack into an orbit where the ICPS could insert Orion into a highly elliptical Earth orbit. The Artemis I AMT-LO target was derived to help Orion achieve all its other flight test objectives in Low Earth Orbit. The switching

[*] EV42/Guidance Navigation and Mission Design Branch, NASA, Marshall Space Flight Center, Huntsville, AL 35812
[†] Jacobs Engineering, Jacobs ESSCA Group, Huntsville, AL 35812
[‡] Jacobs Engineering, Jacobs ESSCA Group, Huntsville, AL 35812
[§] EV42/Guidance Navigation and Mission Design Branch, NASA, Marshall Space Flight Center, Huntsville, AL 35812

between targets is handled either through flight software or by Mission Control. Both the vehicle and Mission Control will monitor SLS's velocity on ascent to make the determination as to which Alternate MECO Target (AMT) the vehicle is capable of achieving. The AMT's are also derived so that the Core reentry, even in the event of an engine failure, has a high probability of impacting water instead of land.

PRELIMINARY ANALYSIS

Developing a trajectory with an unplanned engine shutdown in a trajectory optimizer requres a novel approach to the ascent trajectory problem. The key word in the trajectory is unplanned, so the trajectory is flying nominally up until the engine failure occurs. To do this, the trajectory must first be optimized assuming a nominal flight profile and then those steering angles "locked in" by using a lookup table to provide the thrust vector direction up until the engine failure. After the engine failure, steering controls are added to the optimization problem to allow the vehicle to loft and attempt to recover from the engine failure.

All preliminary analysis is completed in the Program to Optimize Simulated Trajectories (POST),[2] which is the main Three Degrees-of-Freedom (3-DOF) optimization tool used by the SLS program. First, a nominal mission trajectory is necessary to seed the engine out trajectory. This is developed using either a Dispersed Trajectory (TD) vehicle or in the case of Artemis I, the vehicle as it is expected to fly on flight day.[3] The key difference between the two is the TD vehicle is a statistically derived vehicle based on the expected manufacturing varations, where the Artemis I vehicle already has the known tag values for the engines, mass properties and booster propulsion characteristics. The benefit of using the Artemis I vehicle is greater accuracy on when the different Engine Out thresholds may be reached, however, the TD vehicle is a better representation of any SLS vehicle that may be assembled at Kennedy Space Center (KSC).

Another key ground rule that must be chosen is the launch month. Roughly the first two minutes of an SLS flight will be dominated by the two SRBs that provide the bulk of the thrust of the vehicle. One aspect of using SRB's is that their performance is very temperature dependent. That is, warmer temperature allow the SRBs to expend their total impulse in a shorter amount of time compared to a colder temperature. For example, an SLS flight in February would see less benefit from the SRBs compared to a flight in July. Another aspect of the month to month variation is winds. The prevaling winds in Febuary at KSC are typically tail winds, which helps performance, however, the prevailing winds in July are head winds. So while the higher July temperatures help the boost phase performance, the prevailing head winds somewhat offset that performance improvement and July does not see the greatest early engine out capability. Engine Out (EO) capability in February will be shown predominately since it is the performance driving month.

Once the base vehicle configuration and other key ground rules are selected, POST is used to optimize the nominal trajectory to the desired mission targets and the outputs of that trajectory are used to develop the steering lookup table. The POST input deck is reconfigured to instead of optimizing the steering profile, to follow the table explicitly up until the engine failure. Then depending on the phase of flight, controls are added to allow lofting for the vehicle to recover. For example, an engine failure late in flight would utilize the POST inertial Euler steering angles, primarily pitch, to guide the vehicle to the nominal mission target, however, an engine failure earlier in flight may use aerodynamic angles to loft the vehicle if an engine failure occurs during the vehicle's gravity turn. For these earlier failures, it is highly unlikely that SLS will be able to achieve its nominal insertion targets, so it would downmode to one of the two AMT's selected by the program. Also for the very early engine failures, the SLS Flight Software has the option to bias the Chi Table it used in the open loop boost phase guidance to provide some additional lofting of the vehicle. This is also refered to as an optimal Chi Table.

3-DOF RESULTS

With the prepatory work completed, it is now up to the analyst to churn through the trajectory, typically starting near MECO and working backwards. On the Block 1 configuration, it is not uncommon, for even near maximal payloads, to still achieve the nominal MECO target for a significant portion of the late trajectory. Due to the design of the SLS Block 1, it must start throttling late in flight to keep the vehicle acceleration within design limits. When an engine fails, and a Block 1 is left with three engines, it no longer needs to throttle to maintain those acceleration limits. Also in 3-DOF, up to 1/3 of the Flight Performance Reserve (FPR) is available to cover an engine failure. The assumption is the engine failure is already a statistically bad day, so some performance is released to cover the failure. Finally, 1000 lbm of propellant margin is withheld in 3-DOF to cover modeling differences between 3-DOF and Six Degrees-of-Freedom (6-DOF) simulations. This amount was later found to be very conservative, but deemed acceptable for the Artemis I analysis.

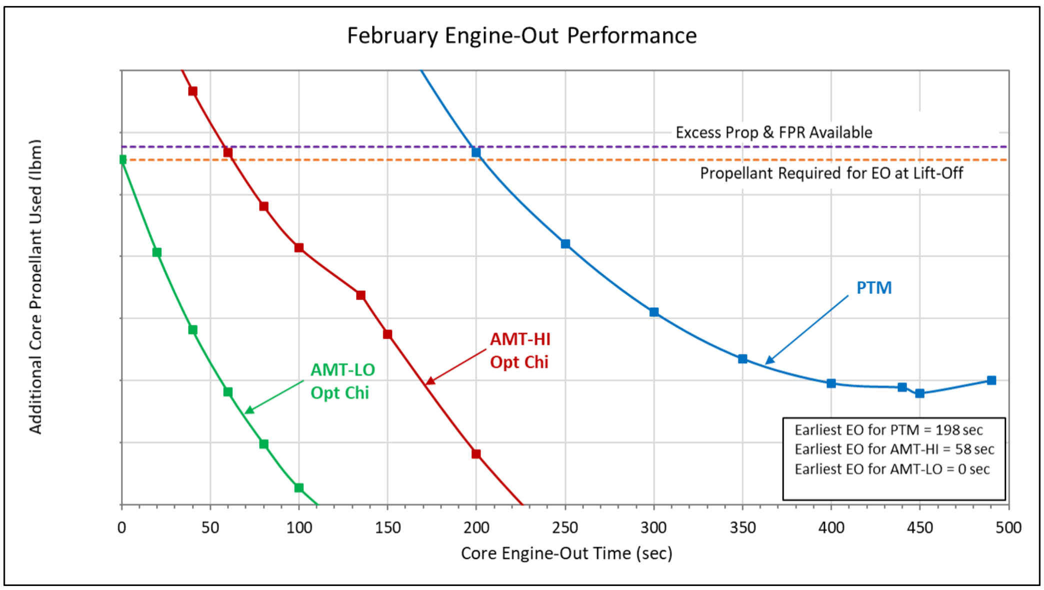

Figure 1. Artemis I Estimated February Engine Out Performance

Figure 1 gives a representative plot of what the Artemis I engine out capability would look like before other factors are taken into account. Reading this plot can be confusing at first, but the key is to look at it right to left. The X-Axis is time where an engine failure may occur, so the far right is the nominal MECO time. As the engine failures occur earlier in the trajectory, performance is relatively flat, due primarily to the late flight Core engine throttling, before impacts to the Press to MECO (PTM) performance starts to take hold. As the engine failure occurs earlier in flight, eventually PTM is no longer viable, so the vehicle would downmode to AMT-HI for failures between about 60-200 seconds. Prior to 60 seconds, the vehicle would select AMT-LO instead. For the two AMT targets, the benefits of an optimal Chi Table are already included in the plot. The two dashed lines represent the propellant required for EO capability at lift off and how much propellant is actually available to cover an EO. The former must be below the latter to show EO capability at liftoff. One final note is all EO times are estimates only. The actual thresholds for the AMT and PTM boundaries are determined through 6-DOF Monte Carlo analysis and not 3-DOF optimization.

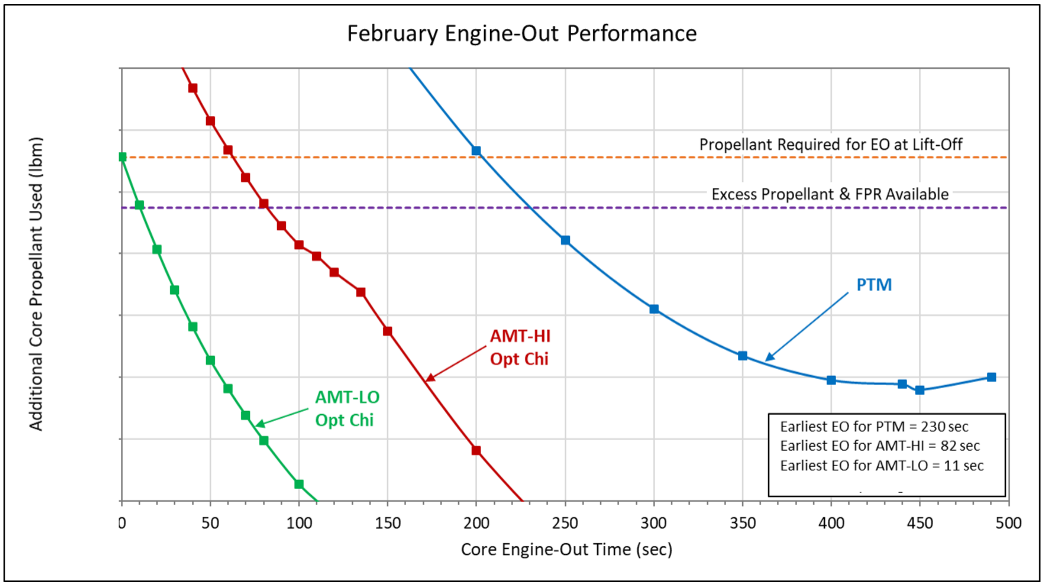

Figure 2. Artemis I Estimated February Engine Out Performance, Stuck Liquid Oxygen (LOX)

The performance presented so far represented the best case scenario where the crossover duct between the two LOX feedlines can fully compensate for the loss of an engine on one side. Realistically, this is not the case, and a significant amount of LOX may become stuck in the feedlines and become unusable for impulse. In that instance, the EO times shift to the right as shown in Figure 2 as the available propellant lines shift down, below what is necessary to protect for AMT-LO capability at lift off.

The next refinement was to introduce a pitch polynomial to the optimal Chi Table optimization in POST and is discussed in more detail later. The method was first introducted in the Marshall's Aerospace Vehicle Representation in C (MAVERIC) simulations and then ported over to POST to assist the guidance analysts in generating the polynomials. The results shows a not insignficant improvement in EO times as see in Figure 3.

Finally, everything assessed so far has assumed fleet average RS-25's and not the actual RS-25's that are planned for Artemis I. A final assessment on the flight engines was performed to assess the sensitivity to the different tag values. The PTM, AMT-HI, and AMT-LO times for each engine as compared to the fleet averages can be seen in Figures 4, 5, and 6.

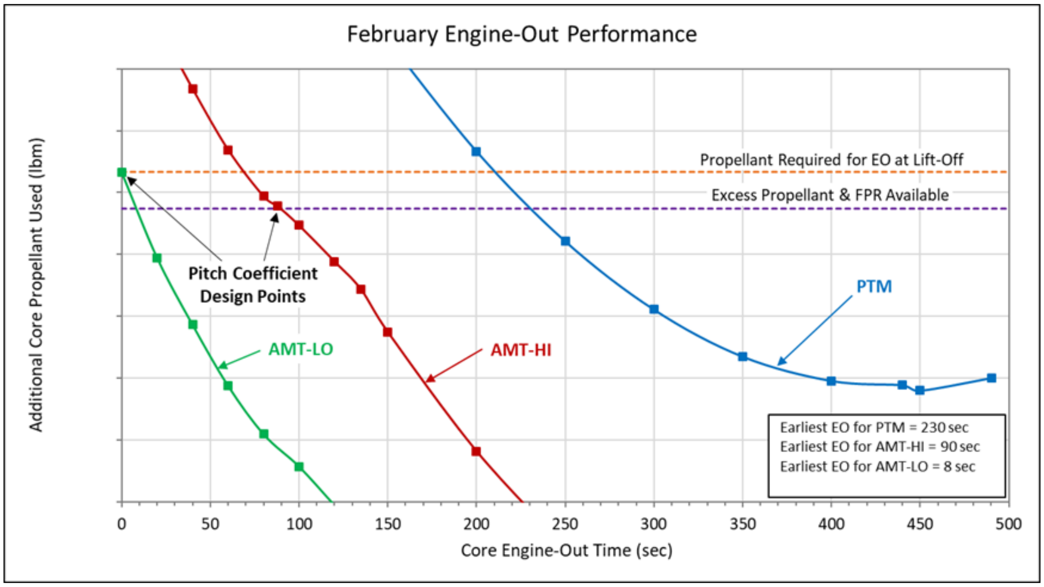

Figure 3. Artemis I Estimated February Engine Out Performance, Pitch Polynomial

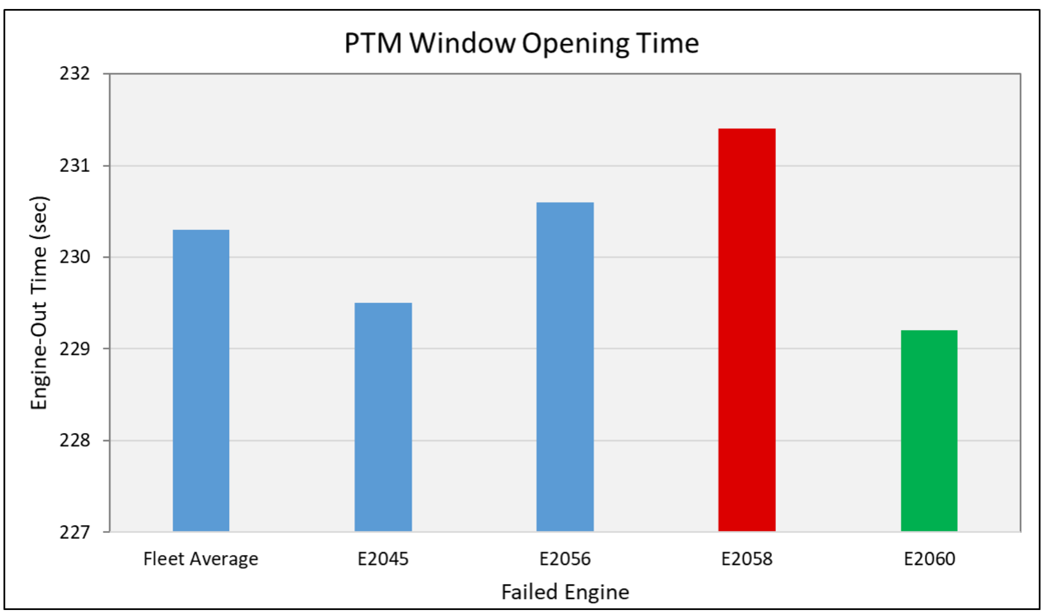

Figure 4. Artemis I Estimated February Engine Out PTM Peformance, Flight Tags

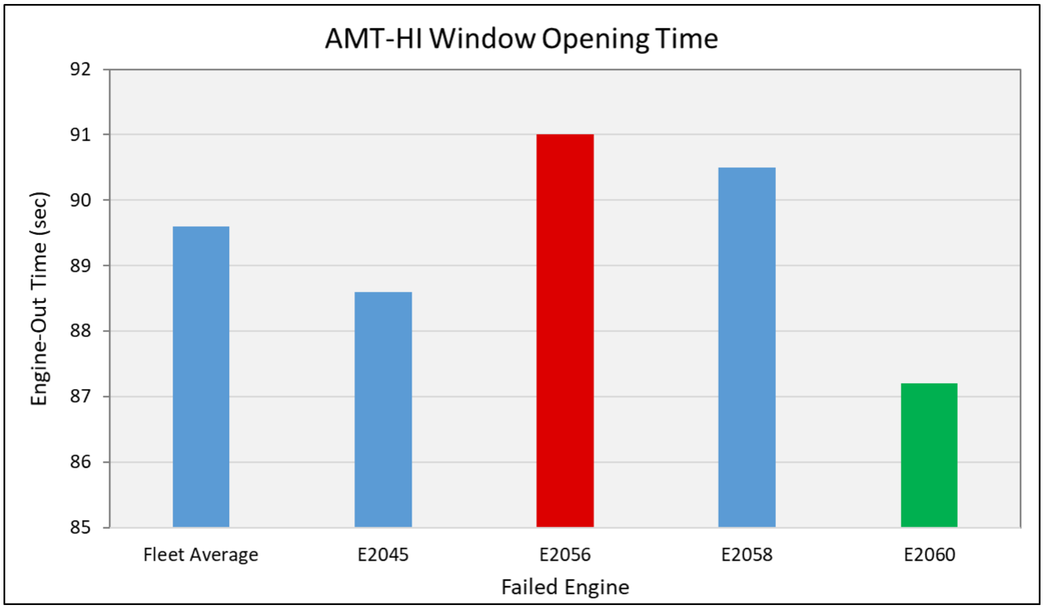

Figure 5. Artemis I Estimated February Engine Out AMT-HI Peformance, Flight Tags

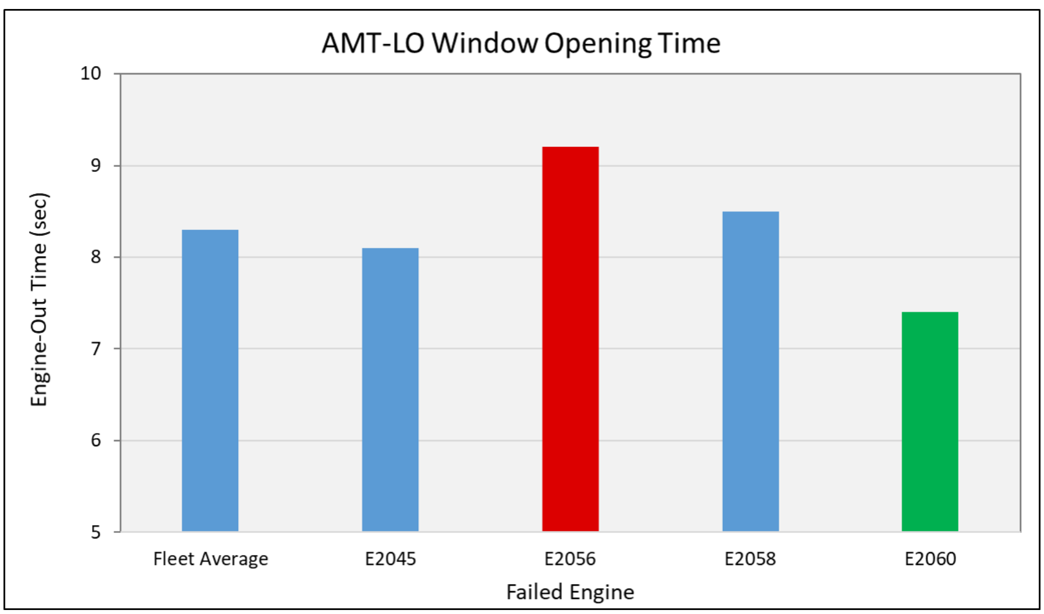

Figure 6. Artemis I Estimated February Engine Out AMT-LO Peformance, Flight Tags

GUIDANCE IMPLEMENTATION

For the SLS Artemis I mission, the onboard guidance system will be used for Boost Stage (BS) flight, defined as the portion of flight when the SRBs are used and provide the majority of the thrust, through Core Stage (CS) flight, when the SRBs have speareated and all thrust is provided by the four core engines. BS flight uses open-loop roll-pitch-yaw and throttle commands (called the Chi Table) as determined by the Chi Angle Optimizer (CHANGO)[4] tool. During CS flight, the Powered Explicit Guidance (PEG) algorithm is used. SLS uses an implementation of PEG derived from the Space Shuttle.

In addition to Open-Loop Guidance (OLG), which uses the Chi Table, and Closed Loop Guidance (CLG), which uses PEG, the guidance system includes all steering functionality, and additional auxiliary algorithms. The steering system converts Chi Table and PEG commands into body rate and attitude commands for the Flight Control System (FCS). The auxiliary functions include a throttle manager, an acceleration limiter based on Shuttle's G-LIMIT algorithm, an engine cutoff algorithm, a mass estimation algorithm, and a targeting algorithm for launch window effects.

A number of algorithm modifications have been made throughout the guidance Flight Software (FSW) code to handle an EO. In the event of an EO, the guidance system may not be able to reach the nominal target set. The desire is to continue on to the nominal MECO targets, called PTM. If PTM is not feasible, two alternative target sets, AMT-HI and AMT-LO, are available for automatic downselection. The exact targets are designed to allow some flight test objectives to be completed while avoiding CS re-entry footprint concerns. For a BS EO, the pitch commands from the Chi Table will be augmented with an empirical delta polynomial. For a CS EO, only the potential downselect is needed, and PEG will re-target if needed. The AMT-HI and AMT-LO target sets are available for manual selection for contingency scenarios.

In the event of an EO, the guidance system is responsible for autmoatically responding and producing a new trajectory that keeps the vehicle safe, with the desire to maximize availability of PTM, followed by AMT-HI, and ultimately ensuring that AMT-LO is feasible even for an EO at liftoff. This section will discuss the process of setting boundaries between the different target orbits, then describe additional design considerations to respond to stressing EOs during BS flight, then describe modifications to the throttle algorithm to ensure safe flight.

The primary guidance EO design task is to set the boundaries between AMT-LO and AMT-HI, and between AMT-HI and PTM. The boundary between two orbits is set as an inertial velocity trigger. Although it is typically more convenient to think of mission time as the boundary, inertial velocity is preferable since it is directly related to the energy achieved and the energy that remains to be made up to get to an orbit. Figure 7 shows a schematic of the velocity triggers within ascent flight. Since the velocity triggers are subject to change, the timing is notional and not to scale. The three orbits are shaded with traditional stoplight colors to emphasize the preference for higher-energy targets.

Simulations from POST are used to determine the approximate boundaries between achievable orbits. Ultimately, the boundaries need to be confirmed via Monte Carlo simulation. Typically, orbit insertion requirements for SLS are assessed at a 3 σ level. However, for failure scenarios, a less restrictive 2 σ level is imposed. EO scenarios tend to be especially sensitive to Monte Carlo dispersions, so the 2 σ assessment allows reasonable boundaries that are not unduly influenced by the significant engine failure on top of an already low-performing case.

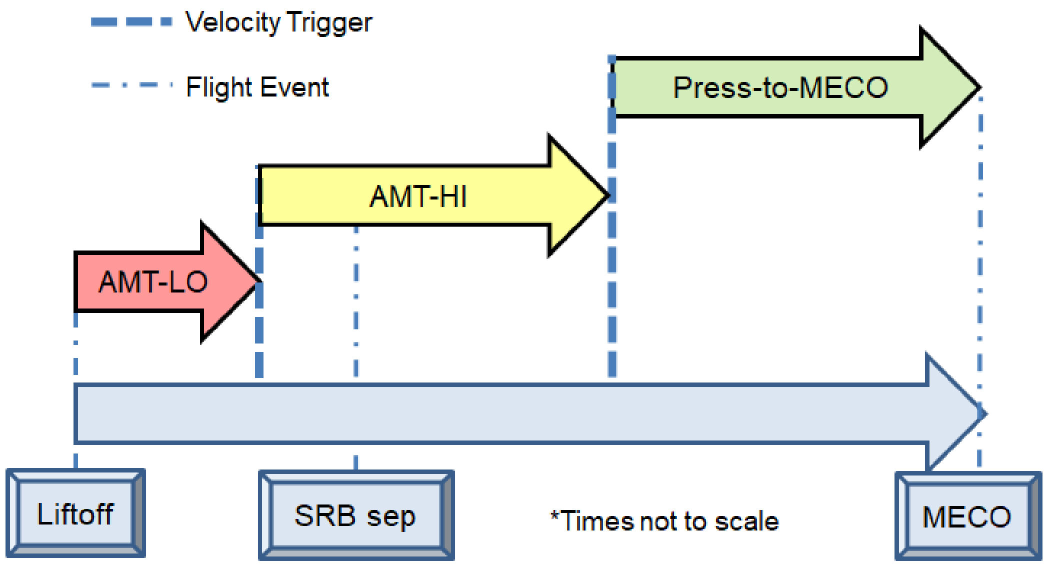

Figure 7. Engine Out Velocity Trigger Schematic

Velocity triggers are desired that ensure the vehicle can reach the selected orbit for all launch dates, and for launches at any time within the daily launch window. The three largest drivers of EO performance are the time of the EO, the launch time within the daily window, and the Propellant Mean Bulk Temperature (PMBT) for the launch. For a given launch, the earlier the EO occurs, the more difficult it is to achieve a given orbit. When setting the velocity boundaries, then, EOs at only three times need to be assessed. The earliest possible EO, at liftoff, is selected to ensure that AMT-LO can be achieved. For the boundary between AMT-LO and AMT-HI, only EOs immediately after the velocity trigger need to be assessed. Similarly, for the boundary between AMT-HI and PTM, only EOs immediately after the PTM velocity trigger need to be assessed.

A second driver of EO performance is the launch time within the daily window. Launches that are away from the optimal launch time require more fuel to steer to the optimum plane. Additional fuel is also needed to make up for the effects of the lost engine. For example, a flight with a late EO can still achieve the nominal MECO target, but will use more fuel than the same flight without an EO. Extra propellant is reserved for launch window effects, and further extra propellant is reserved for EO. To ensure that the selected targets are achievable for all launch times, the design is assessed for the launch time that uses the most fuel for launch window effects. This will be at either the open or the close of the window, and is typically at the open. Testing at the edge of the window ensures that extra propellant for launch window effects cannot also be used for EO effects.

A third driver of EO performance is the SRB PMBT. A colder SRB tends to have lower performance. To ensure that the selected targets are achievable for any given launch day, EO performance is assessed with low PMBT SRBs.

Finally, monthly and seasonal effects affect overall vehicle performance, as well as EO performance. SLS has the ability to use a different set of selected parameters for each launch day, to capture daily effects like orbital plane targeting. The velocity triggers can be designed for each launch day. However, since monthly effects are a driver of EO performance while daily effects are less significant, it is expected that one set of velocity triggers will be carried for each launch month.

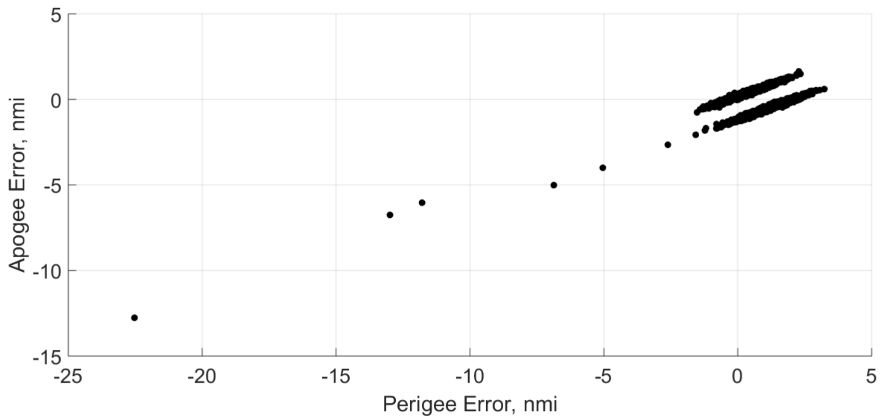

Figure 8. Apogee and Perigee Errors to AMT-LO for EO at Liftoff

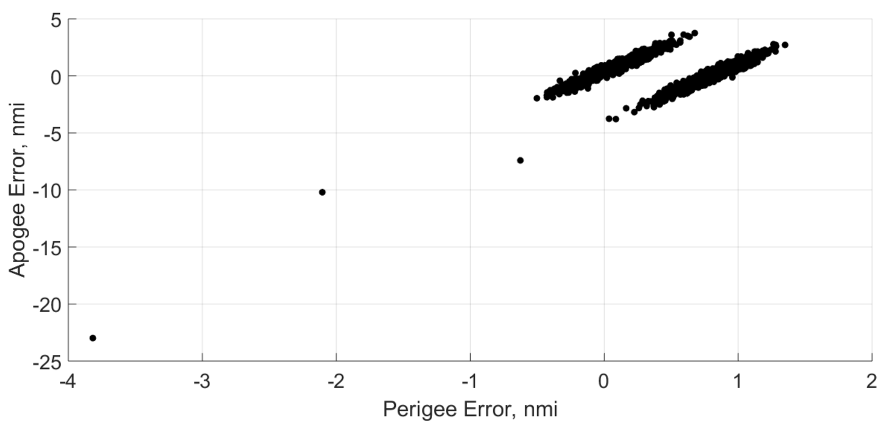

Figure 9. Apogee and Perigee Errors to AMT-HI for EO at Velocity Boundary

To judge the suitability of the velocity triggers, Monte Carlo runs are performed for the earliest EO times, for a launch at either the open or close of the daily window, with low PMBT. Figures 8 and 9 show a representative distribution of apogee and perigee errors for EOs at liftoff (targeting AMT-LO) and at the time of earliest AMT-HI for a typical design month. The guidance targeting near MECO is affected by which of the four engines failed, and typically the final apogee and perigee fall in two families depending on which side the engine was on. With the additional stresses due to the EO, it is expected that a small number of low-performing seeds will fall outside of the desired target error range, and are acceptable with the 2σ criterion. The main clusters of cases fall well within typical insertion accuracy requirements.

The AMT-LO and AMT-HI orbits can be better achieved for early EOs with additional BS guidance modifications. Based on EO trajectory studies with POST, it was discovered that a simple second-order polynomial augmentation of the open-loop pitch profile can allow the vehicle to gain significant performance benefits. The desire to allow AMT-LO for EOs at liftoff, with the desire to have AMT-HI available as early as possible, must be balanced with the need to not unduly stress the vehicle with additional maneuvering. Constraints include ensuring the total angle of attack, α_{total}, is kept low between Mach 0.8 and Mach 2.0, called the buffet-critical Mach region. The peak Q - α_{total} during BS flight must be kept low. The angle of attack at SRB separation must be kept within prescribed limits. Finally, the core re-entry footprint must be within acceptable bounds.

For a more complete description of the BS pitch modifications, including an innovative method for designing the pitch augmentaiton polynomial, see Reference (5).

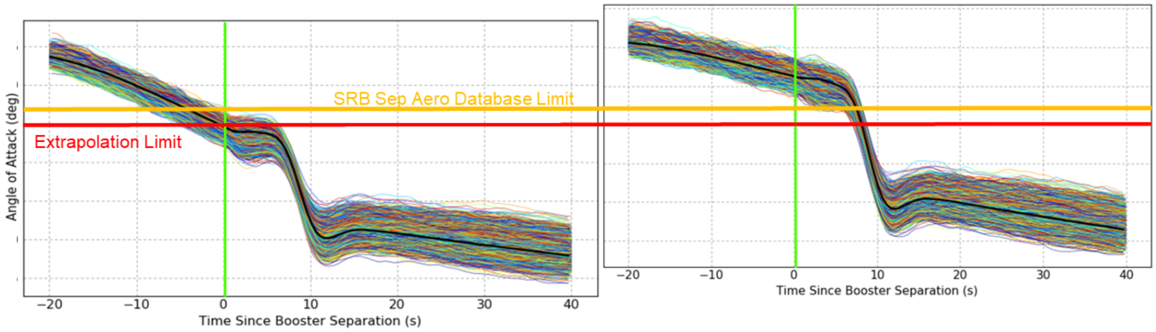

Figure 10. Angle of Attack at SRB Separation for Early (left) and Refined (right) Designs

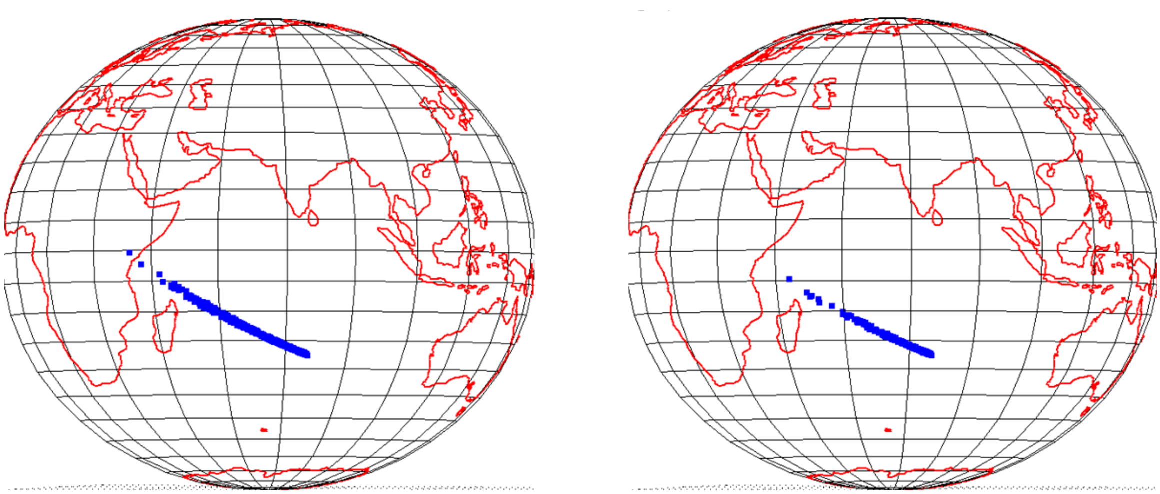

Figure 11. Approximate Predicted Core Impact Points for Early (left) and Refined (right) Designs

An example of an early design iteration and a final refined design for a typical launch month can be seen in Figure 10 and Figure 11. Figure 10 shows the angle of attack near SRB separation, along with the desired limit from the aerodynamics database, and the hard limit based on extrapolating the database. For this example case, the angle of attack at SRB separation was the limiting constraint, and it is seen that the refined design stays above the desired limit. Figure 11 shows the approximate core impact points for the two cases using a simplified impact model. Predicted impact points avoid land masses for the refined design. A high-fidelity tool is used to verify the footprint requirements during final assessment of the guidance design. Proper design of the BS polynomials ensures that the EO maneuvers can improve performance to orbit while staying within other constraints.

Besides the selection of AMTs boundaries and open-loop pitch modifications, the algorithms to command vehicle throttle are affected by an EO. Modifications to these algorithms are discussed below.

The throttle manager algorithm determines the proper throttle setting during flight. To maximize performance to orbit, the vehicle will typically use the maximum design throttle level. The throttle manager polls a number of algorithms and will return the lowest desired throttle level.

For the Artemis I mission, up to five throttle-down events are expected. During the region of maximum dynamic pressure (Max Q), a pre-determined "throttle bucket" is included in the Chi Table by the CHANGO algorithm. Prior to SRB separation, the engines are throttled down to reduce bolt tension. An acceleration limiting algorithm, G-LIMIT, is active during both BS and CS flight. Finally, the engines are planned to shut down from the minimum throttle setting after a certain dwell time. A throttle manager algorithm calculates when to command the throttle down to the minimum setting. Due to the presence of the SRBs, separate G-LIMIT parameters are used during BS and CS flight. Early Artemis I mission design typically required active BS acceleration limiting. With updated trajectory designs and mission requirements, it is now not expected that BS acceleration limiting will be required. The size of the throttle bucket is affected by seasonal temperature variations, among other factors, and a throttle bucket is not always required. Throttle-downs for SRB separation, CS acceleration limiting, and MECO are all expected for every mission.

The loss of a core engine naturally affects the throttle manager, as the same throttle setting now only represents three-quarters of the thrust. To account for these effects, a number of updates were made to the throttle manager to account for an EO.

The throttle manager will follow the throttle commands from the Chi Table until the end of the throttle bucket. Often, the remaining three engines operating at full throttle will still produce less thrust than four engines did while throttled down. However, if the throttle bucket calls for a particularly low throttle value, the throttle manager will limit the throttle to ensure the intended maximum thrust is produced. After the end of the throttle bucket, all three engines will return to full throttle.

The throttle level for SRB separation is set by the liftoff and separation team. A separate value for EO is loaded onto the vehicle, and the throttle manager will choose this value if an EO is known at the time of SRB separation.

Finally, additional logic protects against unstable throttle oscillation if an EO occurs during the end-of-flight G-LIMIT phase.

The G-LIMIT algorithm attempts to hold a constant maximum acceleration level, using feedback on the throttle level. The acceleration corresponding to a given throttle level slowly changes during flight. This correspondence is reasonably constant compared to how quickly the G-LIMIT algorithm operates, allowing throttle to be used in place of thrust. Towards the end of ascent flight, as the fuel tanks empty and the vehicle becomes lighter, the G-LIMIT algorithm produces throttle commands in a downward stairstep shape, leading to approximately constant acceleration. The PEG algorithm expects the final phase of flight to be a constant acceleration phase, relying on the G-LIMIT algorithm to actually enact this.

Loss of a core engine represents a major step change in vehicle dynamics, and severs the link between throttle and thrust. Absent any specific EO considerations, the throttle manager would quickly calculate that maximum throttle is required, as the measured vehicle acceleration is now significantly less than expected and new throttle commands take a finite amount of time to achieve. Before the G-LIMIT phase, and during the early portion of it, there are no concerns as the maximum thrust from three engines is less than the desired level. Very late in the G-LIMIT phase, undesired throttle-up commands are quickly replaced by the final throttle down to the minimum level. How-

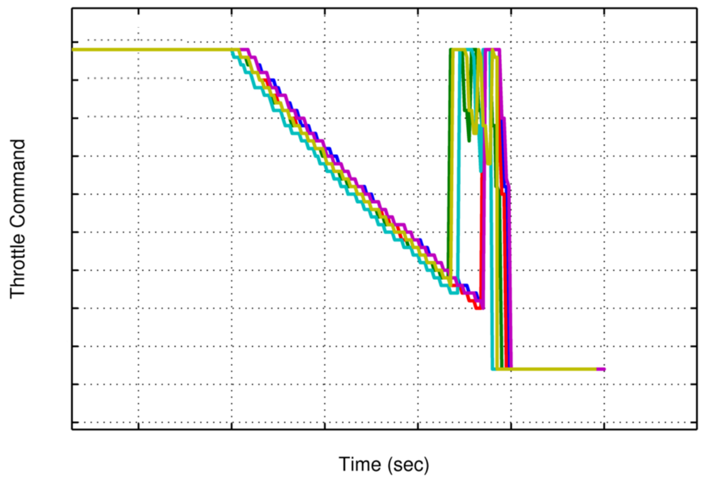

Figure 12. Unstable Throttle Commands, Prior to EO Specific Logic

ever, during the middle of the G-LIMIT phase, an EO would cause the throttle command to quickly reach the maximum, significantly overshooting and allowing far too high an acceleration. Figure 12 shows the throttle commands experienced after an EO during the G-LIMIT phase in the absence of EO logic. Figure 13 shows the resulting acceleration experienced for the same cases.

To prevent the possibility of throttle overshoot during the G-LIMIT phase, logic was designed in the throttle manager to return the throttle to some higher value as soon as the FSW is notified of the engine out, and to impose this value as the new maximum allowed throttle level. Two immediate candidates for the new value are 133% of the value at the EO event, which would return the vehicle to the pre-EO thrust level, or 100% of the value at the EO event, which would ensure that the vehicle cannot command too high a throttle value. On investigation, 133% proved to be too high. Since the G-LIMIT algorithm causes a downward trend in throttle level, by the time this level is reached it is too large, and the vehicle's acceleration briefly exceeds the desired maximum level. The 100% option ensures that the vehicle does not exceed the desired acceleration level. However, this causes the acceleration to drop below the desired level, and typically does not reach the desired level before MECO. This non-constant acceleration can affect the PEG algorithm during the crucial final moments as it recomputes the trajectory to account for the EO. In practice, a value between these two extremes is chosen that restores the desired acceleration as quickly as possible without ever exceeding it. Figure 14 shows the throttle response and vehicle acceleration for these three options. Figure 15 shows the acceleration spike for 133% and the acceleration deficit for 100%. Note that the ideal factor is chosen based on Monte Carlo dispersions across all EO times, and the case shown here does not fully return to the previous acceleration level.

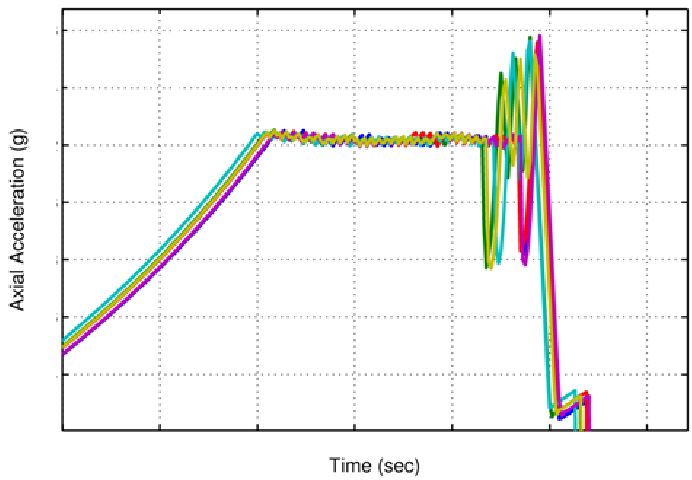

Figure 13. Unstable Accelerations, Prior to EO Specific Logic

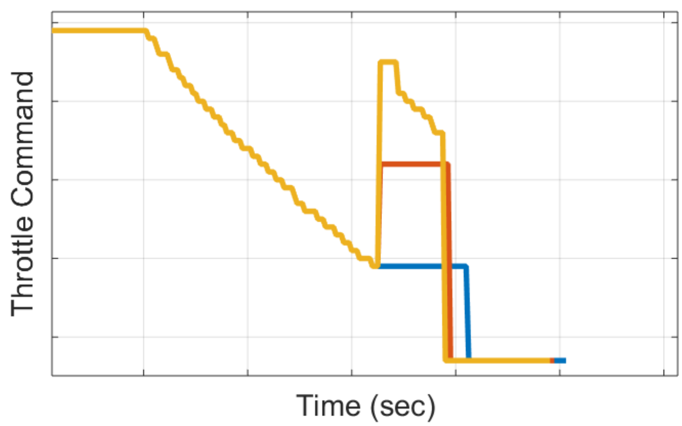

Figure 14. Throttle Commands with EO Specific Logic Designs

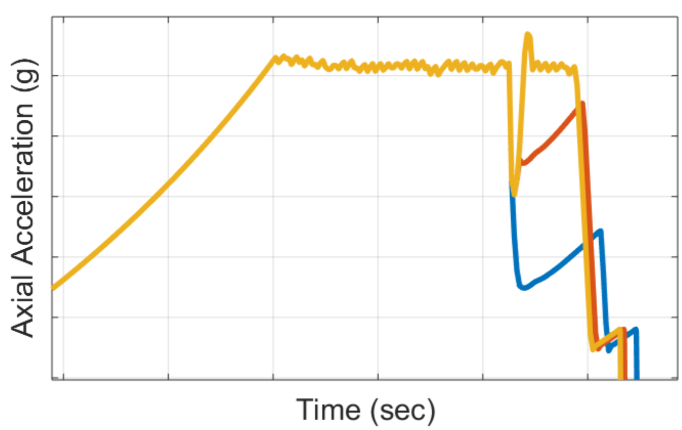

Figure 15. Accelerations with EO Specific Logic Designs

CONCLUSION

The data presented focused on the Artemis I mission and provides an idea of what EO capability the SLS vehicle will have for its inaugural flight. Considering that the first flight does not stress SLS vehicle performance as much as subsequent flights will, this EO analysis will need to be completed for each flight. The important takeaway is the SLS program is planning for contingencies and can show that the launch vehicle has a significant amount of EO capability despite not being designed explicitly to protect for engine failures.

REFERENCES

[1] R. D. Legler and F. V. Bennett, *Space Shuttle Missions Summary*. NASA Johnson Space Center, September 2011. NASA/TM-2011-216142.

[2] S. Striepe, *Program to Optimize Simulated Trajectories (POST2), Vol. 2: Utilization Manual, Ver 3.0*. NESC, NASA Langley Research Center, May 2014.

[3] W. Stein, S. Thompson, T. Statham, and S. Craig, "THE IMPLEMENTATION OF MAXIMUM LIKELIHOOD ESTIMATION IN SPACE LAUNCH SYSTEM VEHICLE DESIGn," AAS/AIAA Space Flight Mechanics Meeting, January 2019. AAS 19-249.

[4] M. Hawkins, N. Ahmad, and P. Von der Porten, "CHANGO: A Software Tool for Boost Stage Guidance of the Space Launch System Exploration Mission 1," *AAS 19-726*, 2019.

[5] M. Hawkins and N. Ahmad, "Guidance Modifications and Enhancements for Space Launch System Block-1 in Support of Artemis I and Beyond," *AAS 20-634*, 2020.