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HUMAN LANDING SYSTEM STORABLE PROPELLANT ARCHITECTURE: MISSION DESIGN, GUIDANCE, NAVIGATION, AND CONTROL

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In preparation of the NextSTEP-2 Appendix H awards, NASA's Human Landing System (HLS) Program (HLSP) went through an abbreviated Design Analysis Cycle (DAC), or miniDAC, to characterize and quantify how a storable propellant design would affect and improve overall system performance of the Government reference design. As part of that miniDAC, HLSP ran multiple trade studies and assessments, including aspects of Mission Design, Flight Mechanics (FM), and Guidance, Navigation, and Control (GN&C). The storable propellant designs were assessed through the baseline mission targeting the Near Rectilinear Halo Orbit (NRHO) and several alternate staging orbits. This paper focuses on the FM and GN&C trades and assessments completed as part of the miniDAC as well as the GN&C elements of the miniDAC Master Equipment List (MEL) developed for the storable propellant lander configurations.

INTRODUCTION

Two basic vehicle architectures were examined in the miniDAC. The first was a three-element architecture (Transfer, Descent, and Ascent Elements) similar to the original government design from DAC-2 completed in mid-2019 (Ref. 1) , but with storable propellants instead of cryogenically cooled propellants. The three-element architecture launches each element separately, autonomously docking with each other in the lunar staging orbit. The second was a two-element architecture (Descent and Ascent Elements) that would launch integrated on a Super-Heavy class launch vehicle currently in development. Both architectures have their pros and cons, which will not be covered in detail in this paper, other than with respect to the mission design and GN&C.

The HLS baseline mission design, as provided in the Solicitation[*], utilizes the NRHO as the staging orbit for all of the HLS elements, to eventually aggregate at Gateway. However, for the first mission, HLS may dock directly with Orion rather than Gateway . No longer requiring a

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[*]https://www.nasa.gov/content/nextstep-2-omnibus-baa/

docking with Gateway on the first mission enables other possible mission architectures and staging orbits. Several staging orbits were examined as part of the miniDAC analysis, including circular and elliptical orbits of multiple sizes and orientations. These staging orbits were traded against one another and down-selected. Station-keeping, aborts, element disposal, the nominal mission, and the mission integration with Orion were assessed based on performance and operational constraints. Mission delta-V requirements were used to estimate the stage sizes for all of the HLS elements and update the MEL.

Pulsed engines are baselined to assist during powered descent for the storable propellant HLS configurations. The crew accelerations from this pulsing were evaluated and compared to crew health standards. All of the HLS designs are significantly larger that the Apollo lander and, for the miniDAC, they were baselined with the same reaction control system (RCS) (sixteen 110-lbf thrusters).

Because the ability for the crew to take manual flight control is an HLSP requirement, the handling qualities of the designs during powered descent and on-orbit docking were examined and compared to Apollo and the Altair lander . The advantages and disadvantages of the proposed propulsion systems are considered, and modifications that would improve performance are identified. Attitude control trades using thrust vector control (TVC), differential throttle, and/or RCS is also explored.

STORABLE PROP CONFIGURATION

Prior to the directive to return humans to the Moon by the end of 2024, previous government reference Design Analysis Cycles (DAC) focused on cryogenic propellant solutions for HLS. As a result of the new directive, the design team was asked to quantify the performance impacts associated with storable propellant solutions for the HLS design through an abbreviated DAC, or miniDAC. During the miniDAC, the design team found that the design efficiencies associated with the storable propellants and removal of cryogenic fluid management systems were not sufficient to offset the increase in propellant mass due to the propellants' lower specific impulse. Due to this design change, the GN&C team was tasked with determining the overall handling qualities and controllability performance of each vehicle based on the augmented assumptions. Some metrics for these studies had to be obtained from the HLS Handling Qualities group, however, study criteria will be further addressed in subsequent sections.

Design Variants

The two primary focuses were a two element architecture (Ascent and Descent Element, or AE and DE) and a three element architecture (Ascent, Descent, and Transfer Element, or AE, DE, and TE). Each design was analyzed with storable propellant systems and two different descent propulsion system configurations: a three and six engine variant. Two and three element architectures were studied to investigate if either option was more likely to close given anticipated launch system constraints in 2024. The two element design potentially allowed the vehicle to be launched pre-integrated on the ground, but would require performance comparable to NASA's Space Launch System Block 1B and a similarly large fairing to launch the integrated HLS. The three element design would launch on three separate rockets but would potentially allow for flights on existing rockets provided by commercial launch partners. In the three element architecture, as each element arrived in NRHO, it would need to autonomously dock with the others before Orion and the HLS flight crew arrived.

Descent Engine Variants

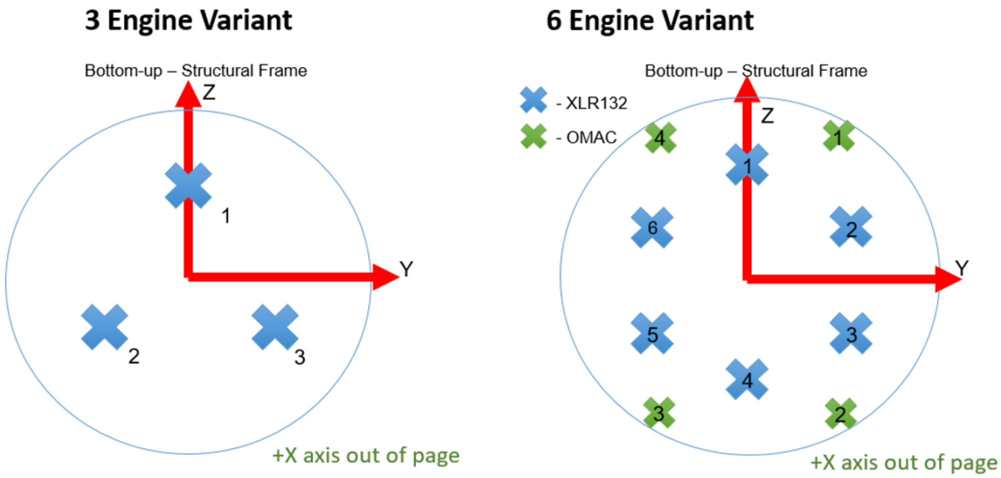

Figure 1. Descent Engine Variant Locations used for study

There were two engine design variants that were assessed on each element design, as shown in Figure 1. No TVC was assumed for this study. The three engine variant consisted of three 8000 lbf thrusters, based on an XLR132. Continuous deep throttling was assumed, allowing down to 20% throttle. The six engine variant consisted of six symmetrically placed 3754 lbf XLR132's* and four 1500 lbf Orbital Maneuvering and Attitude Control (OMAC) thrusters†. The OMAC's are part of the Reaction Control System (RCS) but were included to aid in descent phase characteristics. The XLR132's could throttle between 80% and 100%, while the OMAC's had only two engines pulsed at a time.

Reaction Control System Propulsion

The thruster locations were scaled proportionally based on the radius between element design variants. Since locations of each thruster were not given, the GN&C team had to provide this assumption. The RCS propulsion consisted of sixteen 100 lbf R-4D thrusters with clusters of four placed equilaterally about the descent element CG frame. Cardinal directions was assumed for each thruster on the cluster. A 40ms impulse width and a 300 sec Isp was assumed.

BASELINE MISSION DESIGN (NRHO)

A general concept of operations for lunar landing missions was presented in Attachment A01 of the HLS solicitation, specifying that the contractor would deliver the uncrewed HLS to NRHO prior to the Artemis III mission. Per the solicitation, the uncrewed HLS can be launched as separate elements on multiple launch vehicles, if required by the architecture, followed by rendezvous, aggregation, and staging in NRHO. The solicitation details multiple trajectory options for arrival in NRHO. "Ballistic trajectories, which may range up to 120 days or longer, reduce the energy re-

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*http://www.astronautix.com/x/xlr132.html

†https://www.rocket.com/article/aerojet-rocketdyne-successfully-completes-launch-abort-engine-

quired for NRHO insertion" [*]. Fast transits can reach NRHO in a few days, "but may require a lunar flyby followed by a higher energy direct insertion to NRHO" [*].

The Artemis III crew will be transported to NRHO in an Orion spacecraft launched via the Space Launch System, where the uncrewed HLS will be staged. The nominal HLS mission will be to pick up the crew and mission materials in NRHO after rendezvous and docking with either Gateway or Orion. Following a go call for departure, the HLS will separate from either Gateway or Orion and begin its journey to the lunar surface[†]. The solicitation states that a phasing orbit of some kind will likely be necessary "to update navigation systems to support the powered descent and landing phase of the missions"[†].

Transits from NRHO to this phasing orbit vary in both duration and energy required, however a propulsive maneuver will be required to leave NRHO and to insert into this phasing orbit. A loiter in low lunar orbit (LLO), up to three revolutions, will likely be needed either for crew preparation for descent and/or navigation state updates to reduce error after the Lunar Orbit Insertion (LOI) burn. A nominal transit of 12 hours from NRHO to a 100 km circular LLO is anticipated, but trip time can vary based on final mission designs.[†]

The solicitation outlines four distinct phases for the descent from the phasing orbit to the lunar surface. The first phase consists of a Descent Orbit Insertion (DOI) burn to place the HLS in an orbit with a perilune sufficiently low to perform the second phase, Powered Descent Initiation (PDI). The PDI and Braking phase will slow the HLS into a surface-intercepting trajectory and arrest the HLS to a sufficiently low altitude to begin the Approach Phase, typically consisting of a pitch maneuver to allow for crew viewing of the landing site. The final phase, Terminal Descent and Touchdown, consists of the final vertical descent to the surface, achieving the desired velocity/attitude state for touchdown. The profile and duration of each of these phases will vary according to descent trajectory design. [†]

After the completion of the surface mission, the crew will use the HLS to ascend back up to Gateway or Orion in NRHO. Per the solicitation, this will also consist of four phases: a powered ascent phase similar to launch to orbit on Earth, a loiter period in a phasing orbit to target a return to the staged Gateway or Orion in NRHO, a cruise phase consisting of the transit from phasing orbit to NRHO, and a rendezvous and docking phase to connect with either Gateway or Orion. The ultimate duration and specific propulsive maneuver will vary with ascent profile design. [‡]

Following rendezvous and docking crew and cargo, including any samples or science instruments that are to be returned with the crew, will transfer into Orion for return to Earth using a standard Earth-return mission profile.

ALTERNATE STAGING ORBITS

Nominal Mission

HLS has the option of docking directly with Orion during the initial mission to the surface which allows the possiblity of an alternate staging orbit other than the NRHO. The alternate orbits assessed

[*] `https://www.nasa.gov/content/nextstep-2-omnibus-baa/`, page 5, Attachment A01, NextSTEP-2 BAA Appendix H

[†] `https://www.nasa.gov/content/nextstep-2-omnibus-baa/`, page 9, Attachment A01, NextSTEP-2 BAA Appendix H

[‡] `https://www.nasa.gov/content/nextstep-2-omnibus-baa/`, page 12, Attachment A01, NextSTEP-2 BAA Appendix H

include a 5500 x 100 km co-planar orbit, a 10,000 x 100 km co-planar retrograde orbit, a 7500 x 100 km polar orbit, and a 6500 x 100 km polar orbit where the line of apsides lies in the Earth-Moon plane (Co-planar Line of Apsides, or CoLA). The nominal mission for each of these orbits was assessed in order to evaluate feasibility and ultimately down select to the most favorable alternate orbit.

Initial evaluation of the nominal mission from each alternate staging orbit targeted a specific landing site. Coplanar staging orbits targeted a (0°, 0°) landing site while polar orbits targeted (-89.9°, 0°).

Following Orion insertion in the staging orbit, an HLS-Orion rendezvous attempt and crew sleep phase occurs. This period accounts for 3 rendezvous attempts and 2 sleep periods lasting through approximately 9.5 orbits. Following the rendezvous phase, the HLS coasts to perilune at which point HLS performs a LOI burn to insert itself into the descent LLO. A loiter period of 3 to 4 revolutions in the LLO is completed to prepare for descent to the surface. The HLS coasts in LLO following the loiter to an optimal location to perform the PDI burn which initiates descent to the surface. For this analysis all burns were modeled impulsively so the surface descent phase was modeled as a PDI burn to reduce the perilune altitude of the orbit to 15.24 km followed by a coast to the landing site location and a final descent burn to remove the remaining velocity once the latitude and longitude of the landing site is reached. The magnitudes of the descent and ascent burns to and from the surface are not reported in this analysis since they are included in the model only to estimate the period of time from PDI to landing and lunar liftoff to LOI.

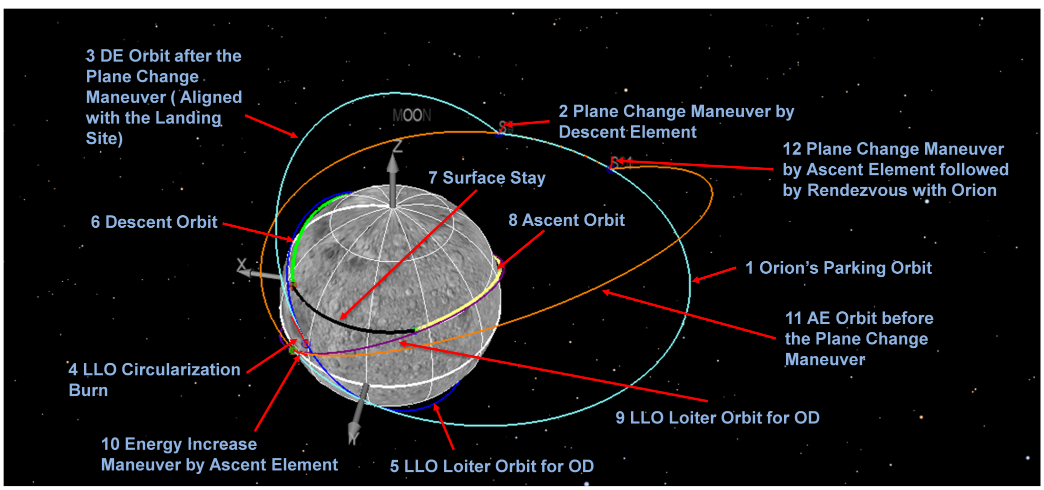

Figure 2. Concept of Operations for a near-equatorial lunar landing mission.

Surface stay time was adjusted to ensure the time Orion spent in the staging orbit is within the allowable limit provided by the Orion analysts. The ascent phase of the nominal mission is a similar set of burns as the descent phase but performed in reverse order. An ascent burn inserts the HLS into a 15.24 x 100 km transit orbit. After coasting to apolune the HLS completes an LOI burn to insert into a 100 km circular LLO. Another loiter period of 1 to 2 revolutions in the LLO is completed to prepare for the LLO to Orion transit. After loitering, HLS performs a Lunar Orbit Departure (LOD) burn to raise the apolune of the orbit and align for rendezvous with Orion. The HLS coasts on this orbit until it reaches Orion where it performs a rendezvous burn. Another Orion-HLS rendezvous phase that includes 3 rendezvous attempts and 1 crew sleep period occurs prior to Orion departure from the staging orbit. The basic Concept of Operations (ConOps) can be see in Figure 2 for the 5500 x 100 km co-planar orbit and is representative of the ConOps for all orbits.

Table 1. 5500 x 100 km Coplanar Orbit Integrated Mission Timeline, Mission Elapsed Time (MET) measured from Orion Arrival

Mission Phase	Event	Delta-V (m/s)	Duration (hr)	MET
Loiter in Staging Orbit	Orion arrives in 5500 x 100km Coplanar Orbit			0.00
	3 RPOD attempts + Checkout (+2 crew sleep)		82.39	82:23:11
	Integrated vehicle undock and backaway			82:23:11
	Transit to Perilune		4.0	86:22:56
Descent	Descent LOI Burn	443.9		86:22:56
	LLO Loiter (3-4 revs)		6.58	92:57:49
	DOI Burn	19.4		92:57:49
	Descent to surface burn sequence	TBD	0.95	93:54:42
	Surface Stay		35.44	129:21:10
Ascent	Ascent to LLO Burn Sequence	TBD	0.92	130:16:20
	Ascent LOI Burn	19.69		130:16:20
	LLO Loiter (1-2 revs)		3.16	133:26:03
	LOD Burn	432.42		133:26:03
	Transit 5500 x 100 km Coplanar Orbit		1.11	134:32:46
	Insert into 5500 x 100 km Coplanar Orbit	24.35		134:32:46
Loiter in Staging Orbit	3 RPOD attemps + checkout (+1 crew sleep)		36.35	170:53:34
	Orion undock and backaway			170:53:34

The mission duration for this timeline in Table 1 exceeds the allowable time in the staging orbit by 0.12 days. In order to achieve a timeline that fits within the allowable period either surface stay time would need to be reduced (by approximately one period of the staging orbit, approximately 0.32 days) or the rendezvous and sleep periods need to be further compressed.

Table 2. 10,000 x 100 km Nominal Mission Segments

	Descent				Ascent		
	Orion departure to LOI	Coast in LLO	DOI to Surface	Surface Stay	Surface to LOI	Coast in LLO	LOD to Orion arrival
Duration (Days)	0.047	0.324	0.040	0.95	0.038	0.329	0.032
delta-V (m/s)	606.6		TBD		TBD		625.4

Refinement of the mission timeline occurred following the initial assessment of the 10,000 x 100 km retrograde coplanar orbit shown in Table 2. This refinement added in Orion and HLS rendezvous phases and reduced the coast in LLO during ascent from 3 to 4 revolutions to 1 to 2 revolutions. The allowable Orion duration in the 10,000 x 100 km staging orbit could not accommodate the additional time so it’s timeline was not updated.

Table 3. 7500 x 100 km Polar Orbit Integrated Mission Timeline, MET measured from Orion Arrival

Mission Phase	Event	Delta-V (m/s)	Duration (hr)	MET
Loiter in Staging Orbit	Orion arrives in 7500 x 100km Polar Orbit			0.00
	3 RPOD attempts + Checkout (+2 crew sleep)		96.29	96:17:24
	Integrated vehicle undock and backaway			96:17:24
	Transit to Perilune		5.11	101:23:50
Descent	Descent LOI Burn	475.74		101:23:50
	LLO Loiter (3-4 revs)		5.91	101:23:50
	DOI Burn	19.4		107:18:19
	Descent to surface burn sequence	TBD	0.95	108:15:11
	Surface Stay		153.66	261:54:47
Ascent	Ascent to LLO Burn Sequence	TBD	0.95	262:51:36
	Ascent LOI Burn	19.37		262:51:36
	LLO Loiter (1-2 revs)		1.96	264:49:12
	LOD Burn	464.86		264:49:12
	Transit 7500 x 100 km Polar Orbit		1.32	266:08:34
	Insert into 7500 x 100 km Polar Orbit	96.07		266:08:34
Loiter in Staging Orbit	3 RPOD attemps + checkout (+1 crew sleep)		44.75	310:53:27
	Orion undock and backaway			310:53:27

Table 4. 6500 x 100 km CoLA Orbit Integrated Mission Timeline, MET measured from Orion Arrival

Mission Phase	Event	Delta-V (m/s)	Duration (hr)	MET
Loiter in Staging Orbit	Orion arrives in 6500 x 100 km CoLA Orbit			0.00
	3 RPOD attempts + Checkout (+2 crew sleep)		84.86	84:51:30
	Integrated vehicle undock and backaway			84:51:30
	Transit to Perilune		4.79	89:38:53
Descent	Descent LOI Burn	524.85		89:38:53
	LLO Loiter (3-4 revs)		7.34	96:59:26
	DOI Burn	19.4		96:59:26
	Descent to surface burn sequence	TBD	0.95	97:56:19
	Surface Stay		104.29	202:13:36
Ascent	Ascent to LLO Burn Sequence	TBD	0.95	203:10:26
	Ascent LOI Burn	19.40		203:10:26
	LLO Loiter (1-2 revs)		2.48	205:38:57
	LOD Burn	470.80		205:38:57
	Transit 6500 x 100 km CoLA Orbitt		4.34	209:59:07
	Insert into 6500 x 100 km CoLA Orbit	22.53		209:59:07
Loiter in Staging Orbit	3 RPOD attemps + checkout (+1 crew sleep)		36.35	246:19:55
	Orion undock and backaway			246:19:55

Surface Aborts

Of the four alternate orbits assessed, two were showing the most promise and selected for more detailed analysis on aborts from the surface: the 5500x100 km altitude posigrade, near equatorial orbit (6.8 deg inclination in the Moon-fixed reference) and 6500x100 km altitude polar CoLA orbit (96 deg inclination in the Moon-fixed reference frame) (see Figure 3) .

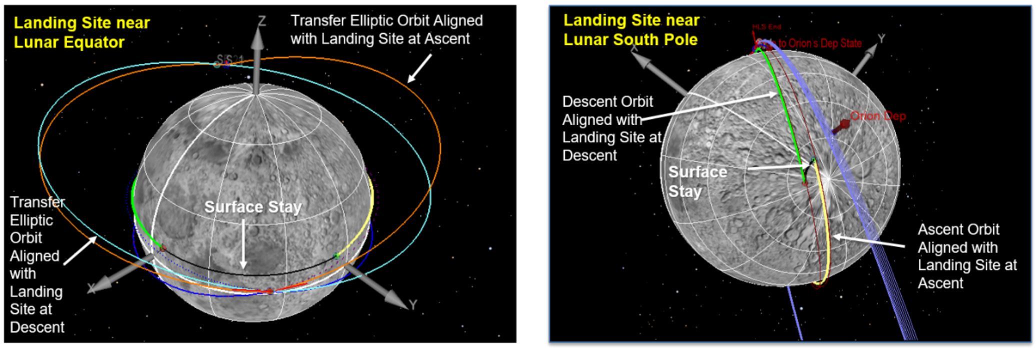

Figure 3. Coplanar and polar orbit based lunar-landing mission scenarios (shown in the Moon-Centered inertial reference frame)

An analytic technique was used to approximate the DE delta-V requirement to depart the (Orion) elliptic parking orbit (EPO) and land on the lunar surface as well as approximating the AE delta-V requirement to return to the EPO for rendezvous with Orion and subsequent return of the crew to Earth. The Copernicus* trajectory design and optimization computer program was used to provide the optimized numerical solution. The nominal mission performance contains all burns (plane change, circularization, DOI) to get the DE to the surface and the AE from the surface, back to the EPO.

The two-burn minimum-energy (delta-V) surface abort analysis of a specific landing site at a latitude of 10°and a longitude of 0°, is shown in Figure 4. For this case, the two burns include the energy change (apoapsis increase) followed, after a coast to the apoapsis of the orbit, by a plane change burn at apolune , targeting to match the Orion EPO. For the lunar mission to this landing site, the total ascent delta-V cost can vary from approximately 500 m/s to approximately 800 m/s. The plane change delta-V follows a similar pattern to the total delta-V cost, suggesting that the energy change (apoapsis raise) delta-V cost is relatively consistent for the variation in surface stay as shown in Figure 4.

A minimum-energy scan of the total delta-V needed for the ascent sequence, taking the AE from post-powered ascent to the (Orion) 5500x100 km EPO, is shown in Figure 5 for selected surface stays (1 through 4 days). These contours provide insight into the landing site abort cost for a given abort time.

Similar to the minimum-energy case, the surface abort performance scan results for the minimum-time case were conducted for the range of landing sites from -15°to 15 °latitude and -60°to 60°longitude. The min, max, and mean delta-V cost of the ascent sequence in case of the two abort modes, minimum-time and minimum-energy, are summarized in Table 5 for surfaces stays from 1 to 4 days.

*https://www.nasa.gov/centers/johnson/copernicus/index.html

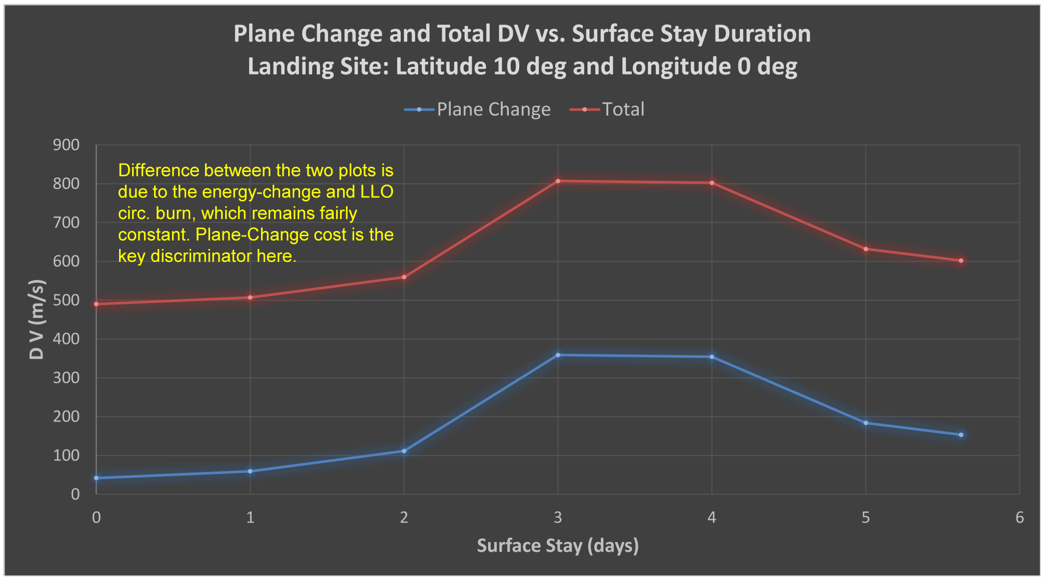

Figure 4. Plane change and Total delta-V cost for early surface abort launch for a landing site latitude and longitude of 10°and 0°, respectively.

The second alternate orbit evaluated is the 6500x100 km polar CoLA orbit with approximately a 96 degree inclination in the Moon-Fixed reference frame. The descent sequence includes only two burns: a single burn at periapsis to do an insertion into a 100x100 km altitude phasing orbit and the DOI (which reduces the periapsis of the phasing orbit to 15.24 km). The ascent sequence also includes two burns: the ascent circularization (raising the 15.24 km periapsis of the 100x15.24 km post-powered ascent transfer orbit) and a single periapsis burn to insert into the 6500x100 km Orion CoLA EPO. The mission design is illustrated in Figure 6. The abort mode examined focuses on early surface departure (launch). It should be noted that the minimum-time and minimum-energy abort scenarios turn out to be the same for this near-polar mission as the required plane changes are much less compared to the near-equatorial mission.

A landing site optimal for the CoLA EPO orbit, with a latitude of -80.97 degrees and a longitude of -99.08 degrees, is chosen to analyze the abort performance for a range of surface stay duration from 0 to 4.45 days. The maximum duration between Orion's arrival and departure from this EPO (CoLA orbit) is 9.327 days, because of which the maximum surface stay duration is 4.45 days, which corresponds to the nominal case. Figure 7 shows the early surface abort results for descent and ascent delta-V cost for the chosen landing site for a range of surface stay durations. The descent sequence total delta-V cost is not affected by the early surface abort as expected. However, the ascent sequence total delta-V cost shows a variation with surface stay duration ranging from 474 m/s to 492 m/s. It is noted that these results are specific to the particular landing site used and will vary for different landing sites. The time duration between the post-abort EPO insertion and Orion's departure from the EPO varies from approximately 5 days to 0.5 days when the surface stay duration changes from 0 to 4.45 days.

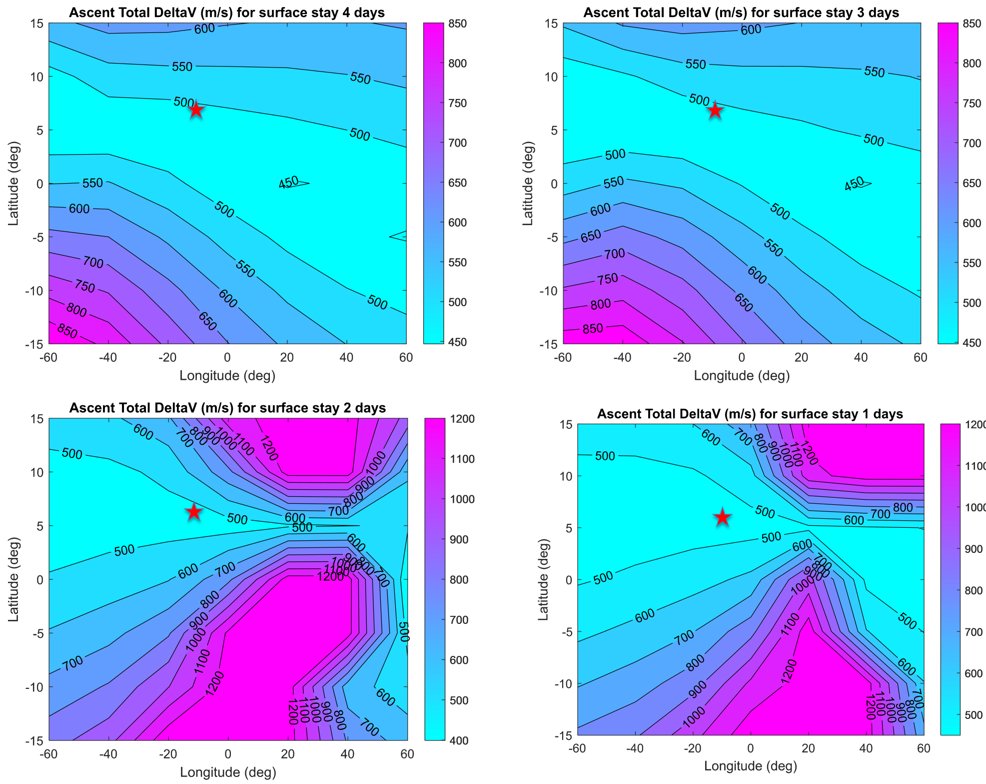

Figure 5. Ascent sequence total delta-V cost for a lunar landing from a 5500x100 km (Orion) EPO with overlaid example reference landing site (latitude = 6.77 deg, longitude = -9.83 deg) with minimum descent delta-V cost, for a 4-day surface stay scenario

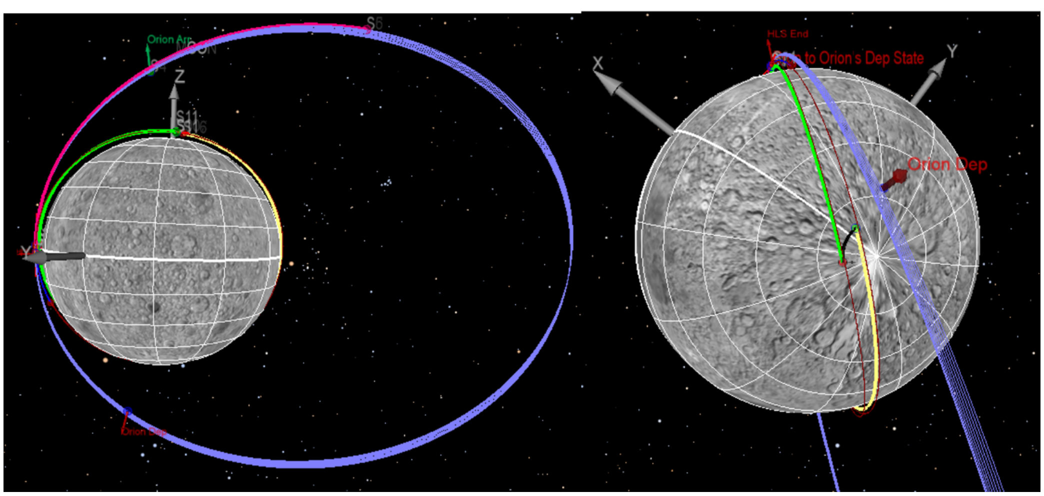

Figure 6. The near-polar lunar landing mission design with Orion in polar CoLA orbit (shown in the Moon-Centered inertial reference frame)

Abort\Ascent ΔV (m/s)	Min	Max	Mean
Min-Time (Stay: 4 days)	449	1304	775
Min-Energy (Stay: 4 days)	447	898	565
Min-Time (Stay: 3 days)	450	1278	815
Min-Energy (Stay: 3 days)	448	887	574
Min-Time (Stay: 2 days)	399	1281	815
Min-Energy (Stay: 2 days)	448	893	583
Min-Time (Stay: 1 day)	450	1269	781
Min-Energy (Stay: 1 day)	447	897	586

Table 5. Comparison of minimum, mean, and maximum ascent abort delta-V for a range of lunar surface stays from 1-4 days

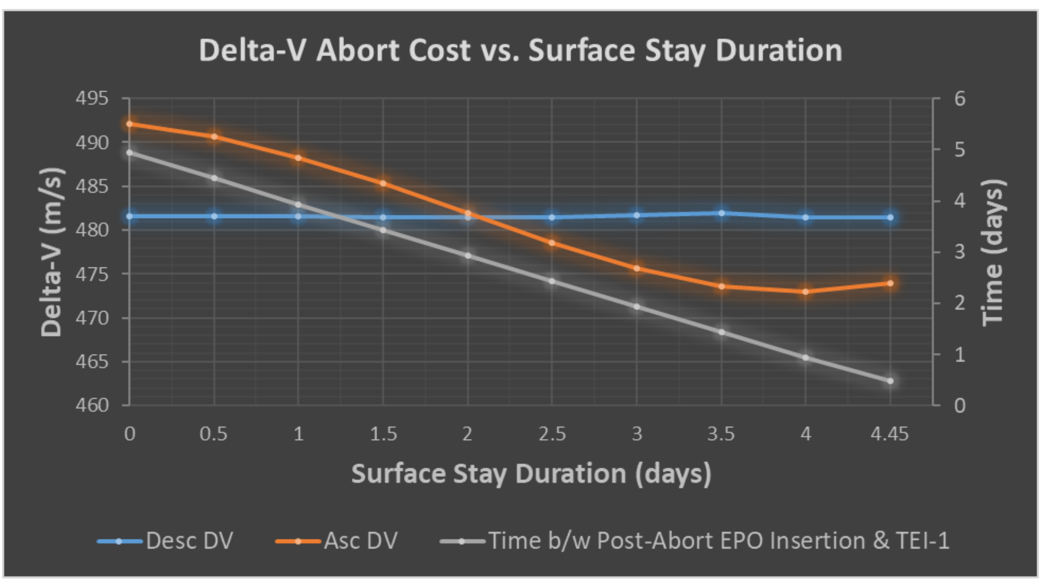

Figure 7. Surface abort results for a selected landing site in case of the near-polar lunar mission. The descent delta-V and time duration between post-abort EPO insertion and Orion's departure from the EPO is shown as well.

PULSED ENGINE DESCENT ASSESSMENT

The mission plan for the designs with six XLR-132s and four pulsed OMACs engines calls for shutting down the XLR-132s at 200 meters altitude and switching to the OMACs for the final vertical descent. This dual main propulsion system is needed because the minimum thrust of two XLR-132s at 80% thrust is too much thrust for the vehicle to land (i.e. the thrust-to-weight ratio is >1 and the vehicle will ascend until propellant is depleted). The OMACs are pulsed engines and concern exists that the accelerations the crew would experience from turning the OMACs on and off would present a human health hazard. This section will determine the accelerations experienced by the crew during pulsed operations and compare them with the human health requirements.

At 200 meters altitude when the switchover to the OMACs occurs there is a limit to the descent velocity that is consistent with a soft landing. Therefore, we must first determine the maximum descent velocity at 200 meters altitude that the propulsion system can changeover at (change from the XLR-132s to the OMACs). For several reasons it would be preferable to only pulse 2 of the OMACs and not all 4. The results of simulating the final 200 meters of descent with an initial velocity of 9 m/s are show in Figure 8 and the results at an initial velocity of 7 m/s are shown in Figure 9. The velocity plot in Figure 8 shows that while starting with a descent velocity of 9 m/s results in a soft landing it comes close to not achieving the desired touchdown velocity of 1 m/s. This is because even when all 4 OMACs are continuously on for most of this descent phase it is only until the final 15 meters that the velocity error is brought under control.

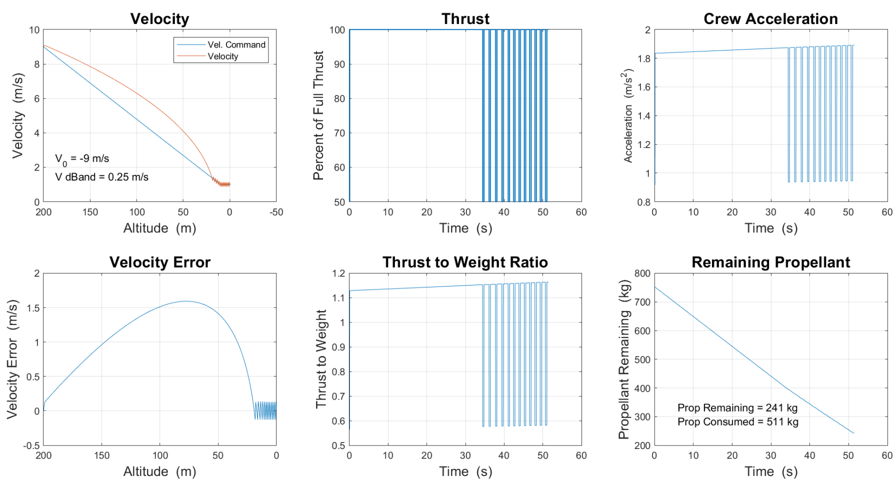

Figure 8. Pulsed Descent with Initial Descent Velocity of 9 m/s

A more conservative approach is to reduce the initial descent velocity to a level where the velocity command is followed for the entire period. Note that the control law includes a deadband centered on the desired velocity. As shown in Figure 9 the desired descent profile can be followed for the entire mission phase with an initial descent velocity of 7 m/s while pulsing of one pair of the OMACs with the second pair firing continuously. The Remaining Propellant plots show that lowering the initial descent velocity results in a higher propulsion consumption during this mission phase. That is because the velocity is lower during the early part of this phase which causes it to take longer. Note that the total propellant consumption would take into account the propellant consumed when the higher Isp XLR-132s operate before the switch to the OMACs occurs.

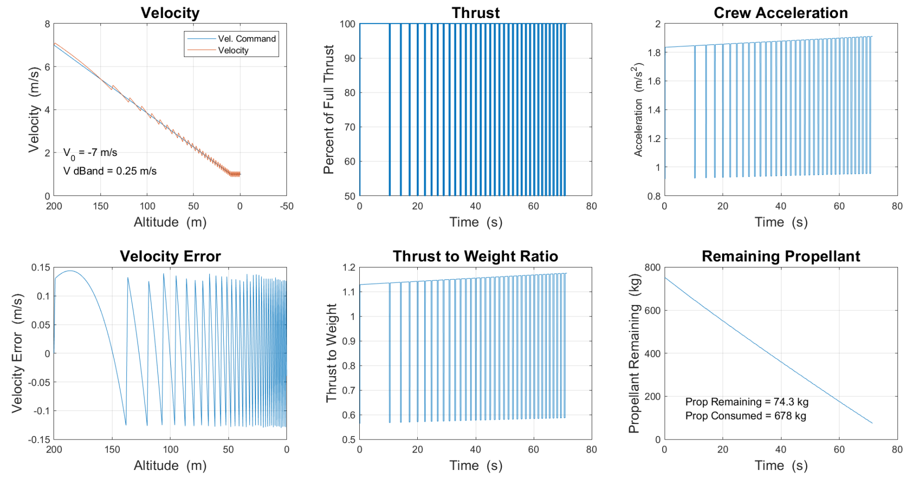

Figure 9. Pulsed Descent with Initial Descent Velocity of 7 m/s

Inspection of these Figures shows that the pulses are approximately 350 msec wide with a frequency of approximately 0.7 Hz. The acceleration that the crew experiences as a result of these pulses is approximately 1.0 m/s2 or 0.1 g's peak-to-peak (from approximately 0.9 to 1.9 m/s2) . HLS-RQMT-001 states the maximum Human Vibration Exposure Limits for frequencies between 0.5 and 80 Hz when exposed for 10 minutes is 3.9 m/s2 and 5.9 m/s2 peak-to-peak when exposed for 1 minute. The 1.0 m/s2 peak-to-peak crew accelerations shown in Figure 8 and Figure 9 are well below the exposure requirement for 10 minutes and even more so for the 1 minute exposure limit. The Vibration Exposure requirement is shown in Figure 10.

Rationale: Internal organs and tissue structures may be damaged if the vibration amplitude goes over these time durations. This duration (under 10 minutes) is expected to bracket the vibration period during lunar descent and ascent. If the dynamic event exceeds this 10-minute duration, requirement [HLS-HMTA-0085] "Long-Duration Vibration Exposure Limits for Health during Non-Sleep Phases of Mission," has to be used from the 10-minute point onward. In accordance with ISO 2631-1, Section 6.3.1, vibration calculations shall be based on a running 1-second time window.

Table 20: Frequency-Weighted Vibration Limits by Exposure Time during Dynamic Phases of Flight

Maximum Vibration Exposure Duration per 24-hr Period	Maximum Frequency-Weighted Acceleration
10 min	3.9 m/s^2 RMS (0.4 g RMS)
1 min	5.9 m/s^2 RMS (0.6 g RMS)

Figure 10. Human Vibration Exposure Limits (HLS-RQMT-001)

In summation the strategy of using pulsed OMACs engines to control the descent velocity during the final phase of landing is feasible when considering both the ability to achieve a soft landing and crew health. Changing main propulsion systems so close to landing, however, is certainly a risky event that needs serious consideration before such a scheme is adopted.

HANDLING QUALITIES

The ability for the crew to take control of the vehicle is an HLS requirement. This manual control requirement calls for more that than just selecting a landing site other than what an automated system has chosen. The crew has to have the ability to manually control the flight path and attitude. For manual control to be meaningful the lander must have adequate Handling Qualities (HQs). Early in the design cycle, when flight or ground test simulators that pilots can rate for handling qualities are not available, a metric based on a dynamic model of the lander provides valuable early insight into the HQs. One metric based on angular acceleration capability is described in Ref. 2. It is a measure of the angular acceleration capability of a lander and is presented as the time it takes to reach an angular velocity of 20 deg/s.

There are three primary methods for controlling attitude: RCS thrusters, differential throttle and TVC. It is possible to combine RCS with differential throttle or TVC to enhance a vehicle's control authority. In order to model attitude control using differential throttle we need a model of the transient when a thrust command to an engine goes from one level to another. For the class of engines analyzed in this study a differential throttle transient of 50% throttle per second is considered reasonable. A change in throttle is modeled as a 2nd order transient with a limit on the change in command of 50 percent throttle per second. This differential throttle transient is shown in Figure 11.

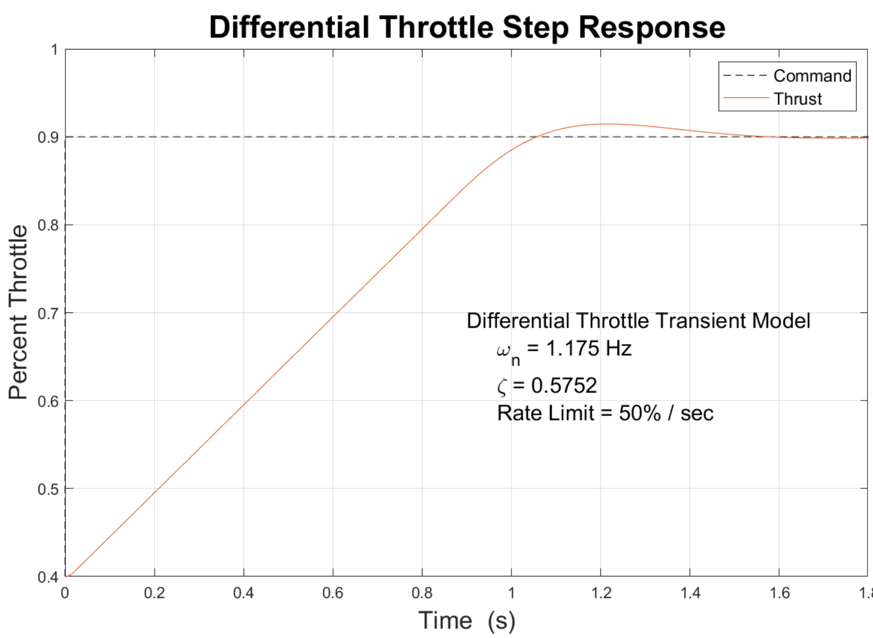

Figure 11. Differential Throttle Transient Model

When TVC is used for attitude control it also has a transient. For this study the TVC gimbal dynamics were modeled as a 2nd order system with a natural frequency of 3 Hz and a damping ratio of 0.8. The TVC transient has a small effect when computing the time it takes for a vehicle to go from zero to 20 deg/s angular rate. Therefore it is not necessary to plot the TVC gimbal transient.

Table 6 illustrates the time to reach 20 deg/s for the 3 vs. 6 engine variants and the 2 vs. 3

element configurations. Since manual control may be necessary towards the final landing phase, this analysis was performed assuming the DE propellant is nearly exhausted hence the inertia of the AE/DE stack is close to the minimum. Columns 3 to 5 show the time to reach 20 deg/s using DE RCS, differential throttling (20% to 100% for the 3 engine variant, 80% to 100% for the 6 engine variant), or TVC only (assuming 6 deg gimbal limit). Columns 6 and 7 show the performance using combinations of DE RCS with differential throttling, and DE RCS with TVC.

Table 6. Time to reach 20 deg/s (seconds)

Configuration	Description	RCS Only	Diff Throttle Only	TVC Only	RCS + Diff Throttle	RCS + TVC
Apollo (4 Thr) Apollo (2 Thr)	Lunar Module Lunar Module	2.2 4.4	N/A N/A	N/A N/A	N/A N/A	N/A N/A
2 Element Design 2 Element Design	3 Engine Variant 6 Engine Variant	39.2 39.2	3.25 6.3	11.87 12.7	3.06 5.15	9.1 9.7
3 Element Design 3 Element Design	3 Engine Variant 6 Engine Variant	21.5 3.7	2.02 3.7	6.55 7.1	1.95 3.4	5.6 6.02

Table 6 lists the time it takes for each design to achieve an angular rate of 20 deg/s. This value is considered a meaningful measure of the handling qualities of a lander during the landing phase. There is a lot more to handling qualities than this metric but it does have value and it can be calculated very early in the design cycle. Figure 12 plots the HQs for the two element lander designs and Figure 13 the three element designs. The figures do not include any of the designs that take more than 8 seconds to reach the 20 deg/s threshold. Designs above the "grey zone" are considered to have satisfactory HQs, those below unsatisfactory and the gray zone something in-between. Note that the Apollo and Altair (Constellation) landers are included for comparison. The Apollo and Altair landers could operate in either 2 or 4 thruster model. That is the pilot had the option of using either 2 or 4 RCS thrusters for attitude control about each axis.

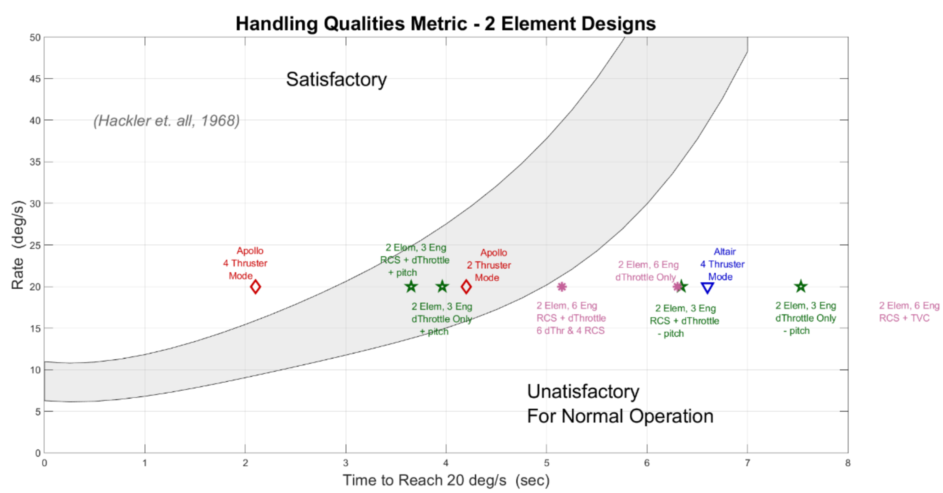

Figure 12. HQs for the 2 Element Designs

The 2 element configuration is more sluggish compare to the 3 element configuration due to a larger moment of inertia. None of the effector combinations for the 2 element configuration falls

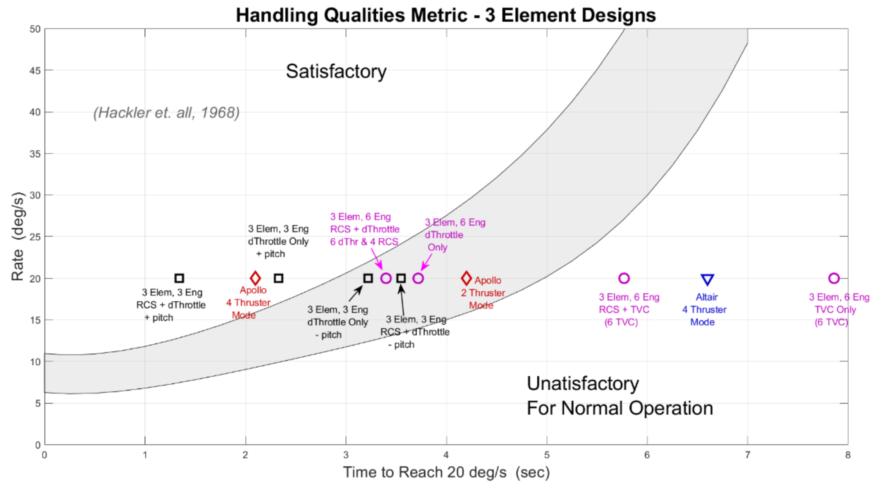

Figure 13. HQs for the 3 Element Designs

within the satisfactory region. With differential throttling (3 engine variant) or pulsing of the 4 OMACs (6 engine variant), the 3 element configuration is able to fall within the satisfactory region. While using the OMACs for attitude control is theoretically possible it is unlikely to be a viable approach. It is worth noting that differential throttling for the 3 engine variant produces asymmetric angular acceleration capability in pitch due to the geometric layout of the engines.

Unlike Apollo, it is evident that none of these designs are able to achieve satisfactory HQ characteristics with RCS only. Furthermore, none of the designs are able to achieve satisfactory HQ with TVC alone or TVC plus RCS. Differential throttling (3 engine variant) improves the HQ characteristics to satisfactory for the 3 element configuration and to marginal for the 2 element configuration. Concerns about the ability to develop new human-rated throttled engines for 2024 certainly exist.

NAVIGATION STUDIES

Another area of focus for HLS is the assessment of initial errors and uncertainty growth in the final phases of flight prior to landing. Due to the number of vehicles operating at the same time, preflight coordination is needed to asssign ground assets for orbit determination to ensure the vehicle can meet its landing accuracy. This is primarily driven by accurately completing on-orbit burns across the mission. The baseline approach is typically performed by first doing a ground-update (or other navigation measurement) to provide a best estimate of the vehicle's position. From this data, the parameters of the burn can be calculated, such as duration and direction in order to correct any orbit dispersions or insert into a desired orbit. Following the maneuver, typically another navigation measurement is generated to assess how well the vehicle performed and provide an estimate of the trajectory. For long duration space missions to locations such as Mars, this can take place over many hours, due to the long time intervals involved and long tracking passes with the ground.

For HLS though, this problem becomes much more involved due to the time scales involved. The various phases are listed below in Figure 14. For the reference design, the vehicle starts in an NRHO orbit, transfers to a Low Lunar Orbit, and then phases its orbit for the desired landing site prior to beginning final descent. The times between these events is critical and decreases toward final descent. This has ramifications for the vehicle design, such as onboard batteries to maintain operations in the lunar shadow, crew supply requirements, and also active crew shift time. As such, it is important to balance uncertainty in vehicle's trajectory (which will grow over time unless updated through external measurements), frequency of Earth-based observations, and overall mission duration to ennsure landing accuracy.

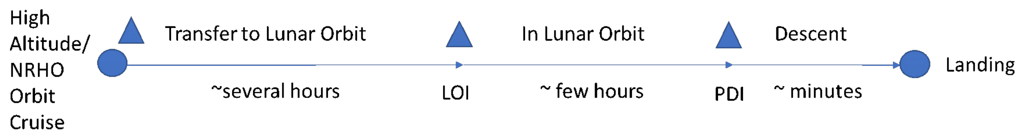

Figure 14. Misson Phases

To address this, a study was performed to understand the relationship between navigation and state errors with the number of cruise orbits prior to beginning descent. The analysis focused on a vehicle in Low Lunar Orbit in order to assess the amount of observation required for ground-based navigation. The first step was to assess the accuracy of orbit determination in Lunar Orbit. Typical DSN Errors (Ref. 3) were used to model mesaurements (two-way range and range=rate) over several Lunar orbits. The FreeFlyer* tool was used to simulate passes to DSN stations on Earth and to perform orbit determination. The results are shown in Figure 15. This analysis varied both the number of orbits as well as time allocated for ground processing and checkout prior to updating the vehicle's onboard state. The baseline requirement is 100 m 1-sigma position accuracy at the beginning of powered descent (when the vehicles transitions to autonomous operations). This analysis showed that, in order to confidently achieve this requirement, 3 orbits are necessary. Fewer orbits would results in reduced accuracy and less time to perform orbit determination and checkout.

Figure 15. Postion Error in Lunar Orbit

With this knowledge in hand, it is also possible to assess the impacts to landing accuracy given this initial uncertainty. Monte Carlo analysis was performed to assess knowledge during descent. To include effects of vehicle state dispersions with guidance in the loop, Linear Covariance Analysis (LINCOV) (Ref. 4) tools were also used with a representative lunar lander trajectory to provide

*https://ai-solutions.com/freeflyer/

additional insight. For the Monte Carlo analysis, the initial error was assessed across a variety of initial error conditions and covariance analysis was performed to assess landing uncertainty. The results of the plot are given below in Figure 16. The left y-axis (solid lines) shows the landing accuracy and the right y-axis (dashed lines) show the uncertainty at the onset of Terrain Ralative Navigatoin (TRN) (consistent with the baseline government design). The x-asis is the initial error. The solid black line shows the uncertainty at TRN turn-on using the baseline design with 100 m uncertainty prior to powered descent. This uncertainty is a function of inertial state propagation form the last ground-based update at 100 km to the 15 km altitude when TRN engages. This plot shows that with greater initial uncertainty in orbit, the vehicle can still land very accurately. The primary impact is to the state uncertainty at the first TRN fix. The sensor system must be designed to work with a larger initial uncertainty on the vehicle's onboard position estimate when polling map images and determining an initial solution. But once it does, the navigation errors are greatly reduced to the level necessary for precision landing.

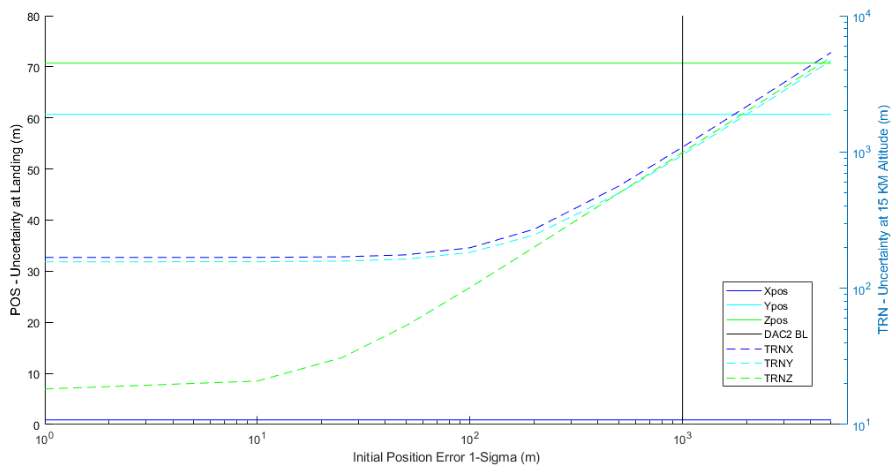

Figure 16. Initial Position Error 1-sigma (m)

The other impact, clear from the LINCOV results, is on the delta-V requirements. Onboard state uncertainty defines how accurately the initial de-orbit burn is performed. Once external measurements are provided, the vehicle can re-target based on the onboard errors and reduce these state dispersions. The impact to the vehicle will be on the amount of delta-V required to clean up these errors. The results of this study are given in Table 7. This analysis was performed using LINCOV analysis and an older reference trajectory, but with the government baseline sensor suite. From this analysis, the impact to delta-V is clearly shown that for two or three orbits, the delta-V dispersions double when compared to the baseline. This impact can be addressed through either including higher altitude operating sensors, such as a feature tracking-based TRN system (Ref. 5) or allowing for extra propellant margin.

An alternate approach to enhancing and improving onboard navigation knowledge is to build out an infrastructure to support vehicle operations. This has been studied for several missions(Ref. 6, 7, and 8), and shows the need for in-situ capabilities. Studies were completed looking at the placement of navigation beacons based on the surface and in various orbits to support navigation both while on approach to and in low lunar orbit. The analysis used covariance analysis to assess navigation capability for various beacon configurations. The results indicated that for approaching the landing

GN&C PERFORMANCE METRICS	TD Rel Navigation Errors (3s)			TD Rel Trajectory Dispersions (3s)			PDI Rel Trajectory Dispersions (3s)			Delta-v Disp (3s)
Sensor Suite	|Pos| (m)	|Vel| (m/s)	|Att| (deg)	|Pos| (m)	|Vel| (m/s)	|Att| (deg)	DnRng (m)	CrTrk (m)	Alt (m)	Disp (m/s)
Case #1: High Altitude Terrain Camera + TRN + NDL	29	0.20	0.042	29	0.24	3.482	2677	190	377	4.84
Case #2: DSN 3-Orbits + TRN + NDL	29	0.21	0.042	29	0.25	3.463	2412	199	631	4.49
Case #3: DSN 2-Orbits + TRN + NDL	29	0.21	0.042	29	0.25	4.183	6429	695	960	10.37
Case #4: DSN 1-Orbits + TRN + NDL	29	0.21	0.043	29	0.25	4.470	7592	817	984	12.15

Table 7. Initial Position Error 1-sigma (m)

site, polar-based ground beacons provided a great benefit for allowing a navigation update prior to performing insertion maneuvers. Similarly, once in LLO, navigation beacons in NRHO provided a great benefit to maintaining state accuracy. This analysis only touched on the surface of the potential implementations, with many aspects under continued assessment, including the navigation measurement method, beacon location, concept of operations, and technology demonstration (Ref. 8). These additional studies are being documented in more detail in a future work.

CONCLUSION

The alternate staging orbits assessed in the mini-DAC showed that HLS could feasibly utilize an orbit other than the NRHO for the first Artemis mission, however, significant analysis is required across multiple programs to assess the full viability. The Controls team showed the viability, but possibly risky option of using storable propellants with pulsed engines, but the handling qualities of those designs is lacking due to the slow response rate of the lander. Finally, the Navigation team assessed what would be required to meet the necessary state accuracy for the mission and provided several alternatives to improve the navigation accuracy.

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