A System Architecture for Human Lunar Return

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Abstract

The history of and the present approach to U.S. human lunar landing missions are discussed, with attention to the importance of maintaining leadership in space from the perspective of U.S. global stature and influence. The NASA Artemis lunar mission architecture and plans are outlined and programmatic and technical issues contributing to the complexity and fragility of the present mission design are discussed. An alternative dual-launch architecture is offered and a point-design within that architecture is assessed. The proposed design assumes a crewed lander based on existing technology, incorporates flight proven systems for other elements, and is shown to meet NASA's basic Artemis requirement – four crew to the lunar South Polar region for a week – with significantly reduced complexity and crew and mission risk.

Introduction

The United States currently faces many significant threats and challenges, some of which feel eerily familiar for those who lived through the Cold War era, a time when a daunting adversary affected almost every aspect of our country's policies and actions. The impact of that time on the psyche of our country cannot be underestimated; it has profoundly shaped the lens through which we view the world today, particularly with respect to China and its global ambitions.

Perhaps one of the most striking examples of this is in our civil space program, where a Cold War style competition in space was recently acknowledged by NASA Administrator Bill Nelson. Among other things, he explains why it is important that the United States accomplish a return to the Moon before China can do so: "I don't want them to get there and say, 'This is ours. You stay out.'"[1]

Interestingly, China's former General Director of lunar exploration missions, Ye Peijian, stated the following at around the same time:

> "The cosmos is an ocean, the Moon is the Diaoyu Islands, Mars is Huangyan Island [Scarborough Shoal]. If we can go, but don't go, future generations would condemn us. Once others are there, once others have occupied, no matter how much you wanted to go you couldn't."[2]

Clearly, both countries understand the importance of "being first"; history teaches us that the rules, standards and values of a new frontier are established by those who show up, not by those who stay home. This point was eloquently articulated by President John F. Kennedy in his September 1962 speech at Rice University:

> “The exploration of space will go ahead, whether we join in it or not, and it is one of the great adventures of all time, and no nation which expects to be the leader of other nations can expect to stay behind in this race for space.
>
> … this generation does not intend to founder in the backwash of the coming age of space. We mean to be a part of it – we mean to lead it. For the eyes of the world now look into space … and we have vowed that we shall not see it governed by a hostile flag of conquest, but by a banner of freedom and peace.
>
> … Yet the vows of this Nation can only be fulfilled if we in this Nation are first, and, therefore, we intend to be first.” [3]

With that in mind and recognizing that we are again in competition with a self-declared adversarial power, it is important to examine our nation’s plan for achieving and maintaining preeminence in human space exploration and development. Recognizing that the last human lunar landing in the Apollo Program occurred more than a half-century ago, a brief recap of the achievements of that era might be helpful to set context for the discussion to follow.

How it was Done Before: Apollo Overview

Following President Kennedy’s May 1961 speech calling for the nation to “commit itself to achieving the goal, before this decade is out, of landing a man on the moon and returning him safely to the earth”[4], NASA conceived and led the Apollo Program, culminating in six landings (of seven attempts) on Apollo 11-17 from 1969-72. This was accomplished at a cost of $25.8 B in 1973 dollars ($257 B in 2020 dollars) in expenditures from 1960-73. Of that total, $4.0 B (1973 dollars) was expended for flight operations for Apollo 7-17, at a cost of $355-447 M per mission.[5] Figure 1 depicts the Apollo mission profile.

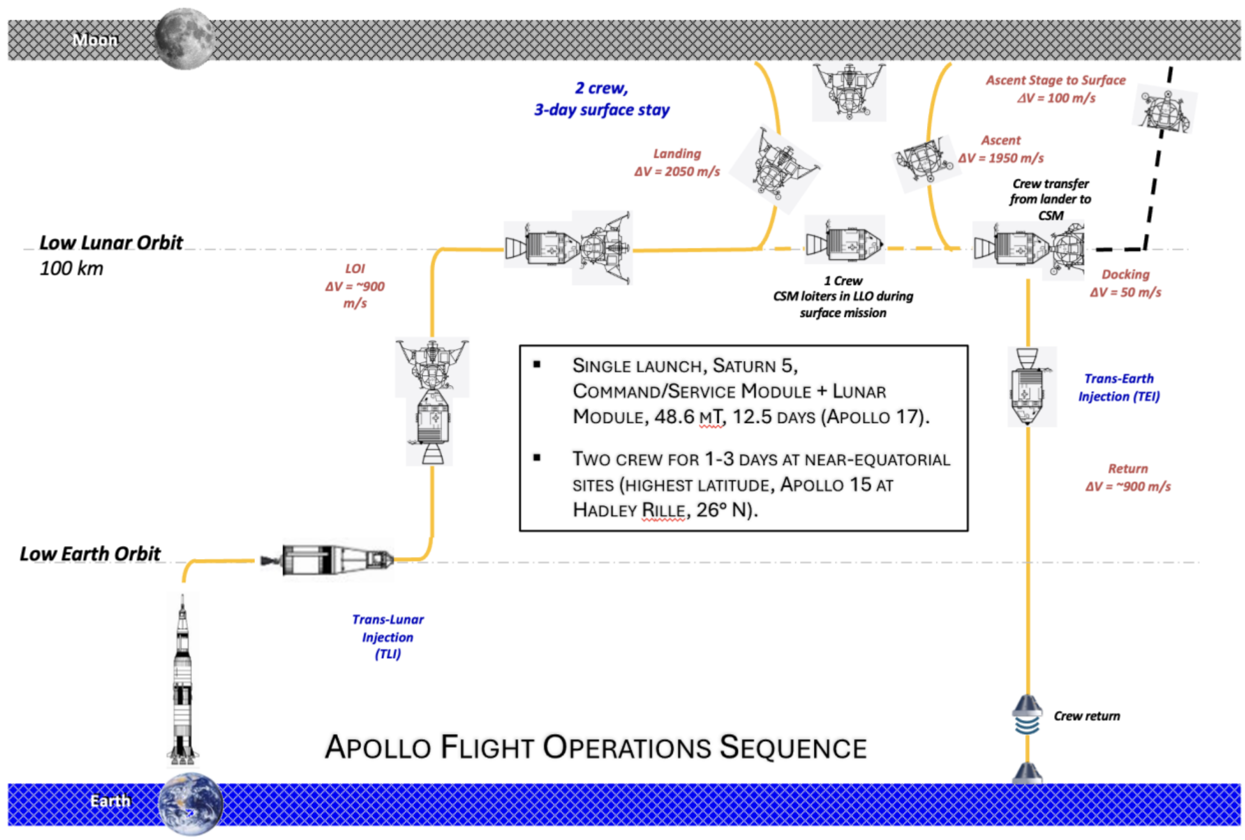

Fig. 1. Apollo Flight Operations Sequence

After Apollo

The Apollo Program was cancelled in a series of decisions by the Nixon Administration in 1969-70, with residual hardware used in four Skylab missions in 1973-74, the Apollo-Soyuz mission in 1975, for museum displays at NASA's Johnson, Kennedy and Marshall Space Centers, or scrapped. There was no strong or consistent support at the national level for human lunar return following Apollo's termination. Presidents George H. W. Bush and George W. Bush each proposed that NASA should undertake to return to the Moon; in both cases their successors canceled those efforts without significant Congressional objection.

Historians, policy makers, NASA leaders and engineers have debated this lack of sustained interest in human lunar exploration and development for two generations, with many reasons having been advanced, all of them likely carrying some element of truth. Probably the most common refrain expressed has been that "Apollo was not sustainable", meaning that it was not affordable. That Apollo was not sustained is an observed fact, but the view that NASA and the nation could not afford continued lunar exploration is difficult to credit.

By 1971-72, with the successful initial missions of Apollo 11-12 accomplished, the immensely valuable experience of the Apollo 13 rescue behind them, and with an increased level of operational experience, hardware maturity and overall system capability, NASA executed two lunar landing missions per year at a cost of less than $900 M/year in 1973 dollars, about $7.4 B in 2020 dollars. At NASA's post-Apollo budgetary low point of about $3 B in 1974, sustaining this two-mission-per-year tempo would have consumed only about a third of the agency's budget in an era when over 50% of that budget was allocated to human spaceflight. Apollo was not fiscally "unsustainable"; it simply was not sustained in favor of other priorities.

Why not? We submit that a major reason, sometimes cited by others, is that with the "race" to the Moon having been "won", there was no clear reason to continue. The lunar landings were risky and there were always competing claims on limited budgets.

But winning a race is a tactical, not a strategic, goal. Two and a half millennia ago, Sun Tzu observed that "Strategy without tactics is the slowest route to victory. Tactics without strategy is the noise before defeat."[6] Apollo was brilliantly executed from a tactical perspective, but in the absence of a long-term strategy for the exploration and development of the space domain, something never prioritized by policy makers in the Apollo era after Kennedy, why continue?

We offer that the proper strategic goal for our nation and its partners is to lead the world into a new human domain. The imperative to expand the human range is as old as the human race; it is a goal to be pursued *whether we have an adversary or not*. It is not a race and should not be framed as such; there is no finish line. While having a formidable adversary during the Apollo era against which we felt the existential need to "win the space race" pushed us to a remarkable accomplishment, that motivation in itself should not have defined our national strategy. And we must not allow ourselves to fall into that trap again.

We believe that former Presidential Science Advisor John Marburger came closest to articulating the true essence of the imperative of human space exploration in a 2006 speech, observing that "The question about the Vision (for Space Exploration) boils down to whether we want to incorporate the Solar System in our economic sphere, or not."[7] We think that is exactly right. In the long run, human exploration, the expansion of our range of action, has always paid dividends, usually not to the individual pioneers (for whom such ventures tend to be quite hazardous) but to the generations that follow. If that were not true, if the motivation to "see over the next hill" did not pay off in the long run, it would have been weeded from the gene pool long ago.

But we must acknowledge that human space exploration is not for the faint of heart nor, from Oscar Wilde's famous definition of a cynic, is it for people who "know the price of everything and the value of nothing". And we must recognize that the decisions we make today will determine whether we lead in the "great adventure" of space, as Kennedy described it, or whether we "founder in the backwash" of other pioneers whose values do not reflect our own.

Artemis Overview

Beginning in 2017, successive administrations under Presidents Trump and Biden have supported a renewed human lunar exploration initiative, the Artemis Program. The core Artemis mission requirement is to place a crew of four on the lunar surface for about a week, with longer stays leading to the establishment of a lunar base to follow.

Artemis I, conducted in November-December 2022, was a successful uncrewed circumlunar swingby, a development flight test for the Space Launch System (SLS) and the Orion crew vehicle[8]. Post-flight inspection has surfaced concerns about the integrity of the Orion heat shield[9]; these will be addressed prior to Artemis II, again a circumlunar swingby mission but with a crew of four[10].

Mission architectures for Artemis III and beyond invoke a critical new feature, the use of a near-rectilinear halo orbit (NRHO) as a staging orbit for landing missions. Halo orbits are solutions to the circular restricted 3-body problem (CR3BP) of orbital mechanics[12], which describes the motion of a small body moving under the influence of two larger bodies circling their common center of mass, e.g., the Orion spacecraft in the Earth-Moon system. NASA has chosen an approximately 3,000 km x 70,000 km, 9:2 halo orbit, for Artemis, so named because a spacecraft in that orbit makes nine orbital revolutions every two lunar months.

Artemis III is scheduled to conduct the first crewed landing mission in the program; Figure 2 depicts the orbital architecture and shows how the NRHO staging area is used in the flight operations sequence.[11] Two crew members will make the landing, with two remaining with the Orion crew module in NRHO.

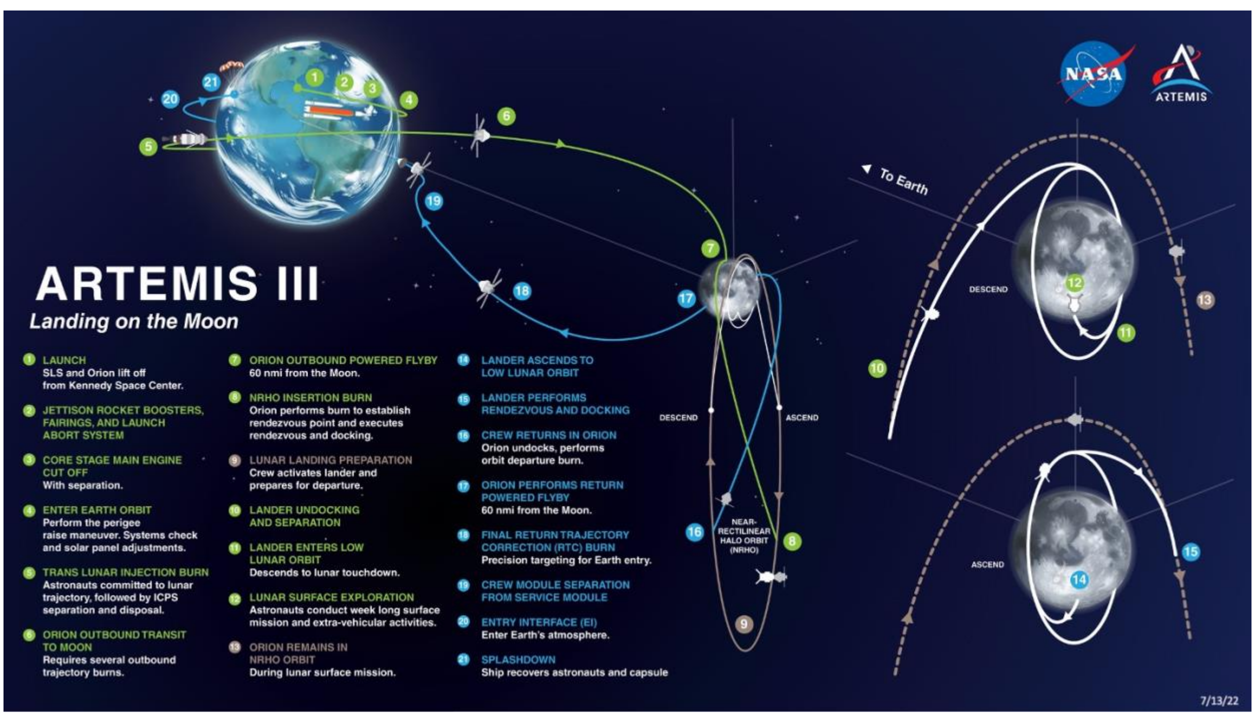

Fig. 2. The Artemis III Flight Plan (Credit: NASA)

Landing missions after Artemis III will stage at the Gateway, shown in Figure 3, a small space station consisting of a habitation module and a power and propulsion element.[13] Gateway is to be deployed in NRHO on the uncrewed Artemis IV mission.

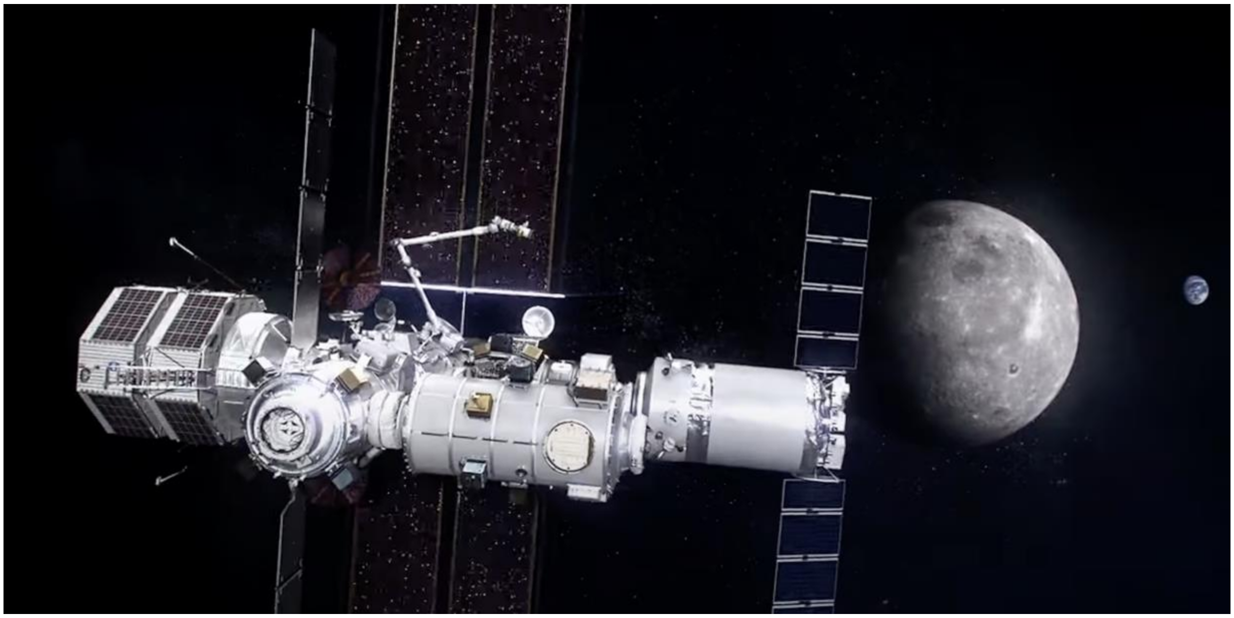

Fig. 3. Artist's Concept of Gateway in NRHO (Credit: NASA)

The Artemis V mission and those to follow will use Gateway as a transfer node and supporting element for the lander, which is pre-deployed to NRHO prior to crew launch, and Orion, which will remain docked at Gateway while the crew descends to the lunar surface in the Human Landing System (HLS).

NASA is using a "commercial service" acquisition model for many elements of the Artemis program, wherein the agency rents or purchases services rather than taking delivery of hardware built by prime contractors. In the current plan, NASA is responsible for:

- Definition of the mission architecture.
- Crew delivery (by means of SLS and Orion) to NRHO for rendezvous with the pre-deployed HLS or, when available, Gateway.
- Crew return from NRHO to Earth surface.

The HLS contractors are responsible for:

- Design and production of their HLS elements.
- Delivery of the uncrewed HLS to NRHO and, when available, rendezvous with Gateway.
- Uncrewed HLS loiter in NRHO for up to 60-90 days (NASA HLS-R-0322 threshold/objective) while awaiting crew arrival.[14,15]

- Crew delivery from NRHO to the lunar surface and vice versa.

SpaceX was selected in 2021 for the Artemis III mission and received a second award in 2022 for a later landing mission.[16] A team led by Blue Origin was selected in 2023 to execute the Artemis V mission.[17]

Given the constraint of staging operations through NRHO, the planned sequence of flight operations for the two contractors is somewhat similar. A fuel depot/transfer vehicle is placed in low Earth orbit (LEO) and subsequently filled by multiple flights of either the SpaceX Starship or the Blue Origin New Glenn. The required number of tanker flights has not yet been definitively announced in either case but is variously estimated to be 10-20 for SpaceX.[18] For SpaceX, the uncrewed lander is launched, fueled in LEO and deployed to NRHO. For Blue Origin, a tanker is filled in LEO and sent to NRHO where it refuels the separately deployed Lander.

Following the SLS/Orion crew launch and rendezvous with the HLS in NRHO, the crew leaves NRHO in a half-day transfer to a polar low lunar orbit (LLO), a parking orbit to set up the descent and landing near the lunar South Pole. The NRHO selected by NASA and shown in Figure 2 has a period of approximately 6.5 days, permitting optimal access to and return from the lunar surface on that schedule. The HLS departs the surface for LLO and the subsequent half-day transfer to NRHO and rendezvous with Orion or (beginning with Artemis V) the Gateway/Orion assembly, 6.5 days after initially leaving NRHO. Figure 4 shows the Artemis III mission design[19] while Figure 5 recasts it in the same format as employed elsewhere in this discussion. Figure 6 shows the Artemis V approach, which differs from Artemis III with the inclusion of the Gateway and of a lander refueling operation in NRHO.[19]

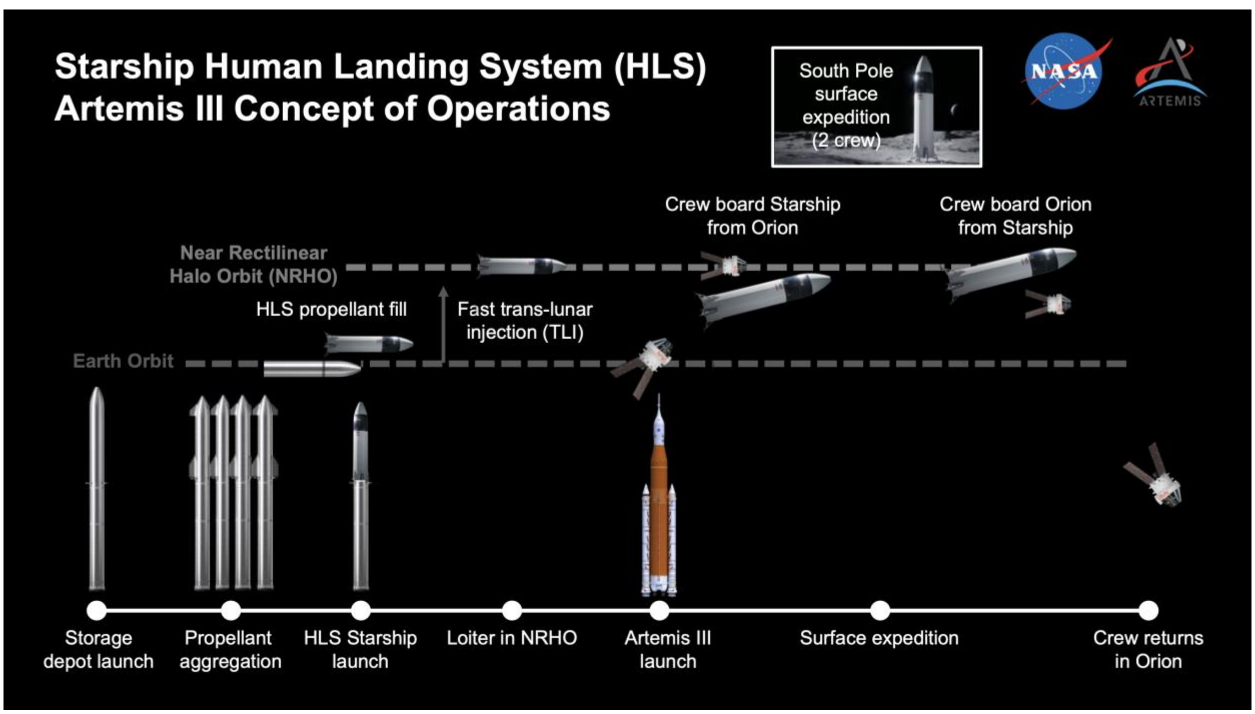

Fig. 4. Artemis III Mission Sequence (Credit: NASA[19])

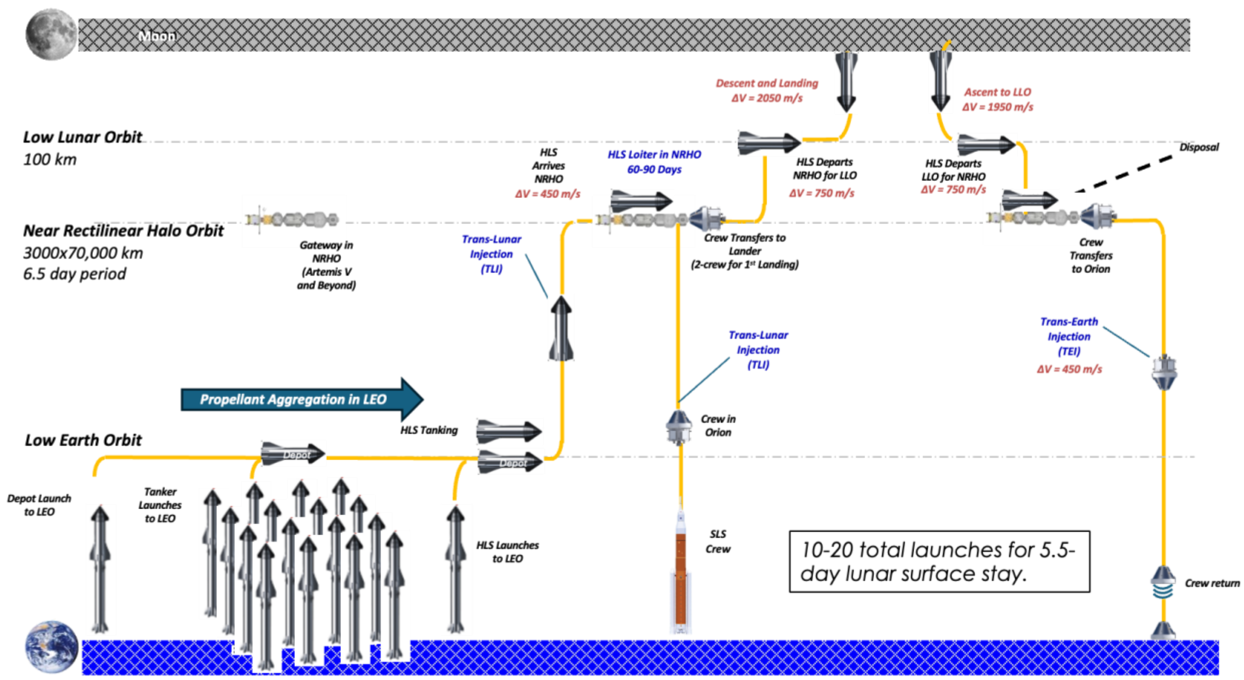

Fig. 5. Artemis III Concept of Operations

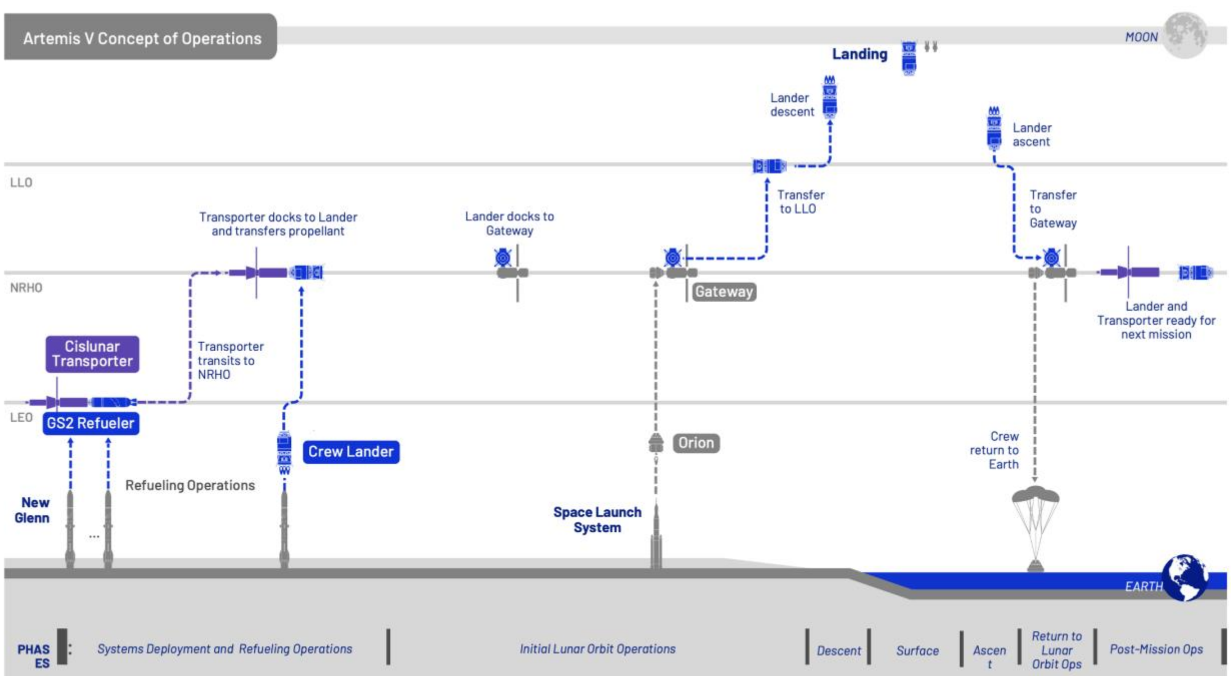

Fig. 6. Artemis V Concept of Operations (Credit: NASA[19])

Preliminary Observations

Compared to the Apollo architecture, the Artemis mission designs are very complex. Complexity equates to increased risk to mission success, or worst yet, to crew risk, and should be accepted only when simpler alternatives will not meet mission requirements.

This widely accepted principle of engineering design does not appear to apply to Artemis. From first principles, to put four people on the Moon should not require significantly more total mass and complexity than to execute two Apollo-style missions with two people each. There are differences, of course. To remain on the surface for 5.5 days is more difficult than to do so for three days. The Saturn V translunar insertion (TLI) mass for Apollo 17 was approximately 48 mT[20,21]; the presently advertised limit for the SLS Block 2 is 46 mT.[22] Finally, the lunar poles present a more difficult landing challenge than the Apollo equatorial region landing sites. But at the system level, these differences would not appear to pose as much difficulty or to require as much complexity as the Artemis architectures would imply.

Crew Risk

The Artemis mission architecture imposes significant crew risk. An optimal half-day return to the "mother ship" in NRHO is possible only on 6.5-day centers. An urgent abort from the lunar surface can require up to 3.6 days to return to NRHO[23], and Orion does not have the capability to leave NRHO to rescue a crew from a lander stranded in a lower orbit.

Beyond crew risk, the Artemis architecture poses significant mission risk in two very fundamental ways. First, both contractor mission designs require multiple launch, docking and automated on-orbit cryogenic propellant transfer operations. From the laws of simple statistics, any mission requiring the execution of multiple independent events for its accomplishment will be challenging, even if each individual operation is judged to have a high probability of success. Table 1 illustrates this point; as a simple example, if it is assumed that each operation in a required sequence of events has a 98% chance of being successful, an attempt to execute fifteen such events in a row will fail about 25% of the time. Each of the proposed contractor mission designs requires many more individual operations than offered in this example. Second, both designs will require cryogenic propellant boiloff rates at levels that so far are only theoretically possible; the technology to achieve such levels has never been demonstrated. (See below.) In traditional program management parlance, this is equivalent to putting a TRL-1 technology on the critical path, something that should never be done.

Cumulative Reliability of Multiple Independent Events				
Single Operation Reliability	Number of Operations			
	5	10	15	20
99%	0.95	0.90	0.86	0.82
98%	0.90	0.82	0.74	0.67
97%	0.86	0.74	0.63	0.54
96%	0.82	0.67	0.54	0.44
95%	0.77	0.60	0.46	0.36

Table 1. Cumulative Reliability of Multiple Independent Events

Regarding the implications of Table 1, it is sometimes suggested that multiple repetitive operations such as planned for Artemis III and Artemis V and beyond do not create a significant mission risk because if a single tanking flight, docking, or propellant transfer operation fails, another can quickly be launched. This view ignores the long history of recovery from spaceflight anomalies, especially if a failure might involve a component or system that is common to a crew system. It also ignores the much more extensive history of responses to aircraft accidents, where, in recognition of the fact that failures are often due to a systemic design flaw and are therefore not "independent events", a significant failure due to an unknown cause often results in the grounding of an aircraft model until the cause is understood and a remedy is implemented. Thus, because Table 1 assumes independence of events, it offers an upper bound to the mission success probabilities provided.

Cryogenic Propellant Boiloff

Of even greater concern is the unsolved problem of controlling cryogenic propellant boiloff for the lander and the various tankers and fuel depots required to implement the Artemis mission architecture. Solar heating of the propellant tanks in space causes the propellent to boil, resulting in an increase in internal pressure that must be relieved by venting the tanks before burst pressure is reached. This poses numerous problems for any deep-space mission design using cryogenic propellants.

The lunar lander must be pre-deployed to NRHO, as it makes no sense to launch a crew unless the lander is known to be available. This imposes a significant loiter requirement for the uncrewed lander, which employs cryogenic propellants in both of the selected mission designs. A similar requirement exists for propellant storage in LEO for the unknown time required for the multiple tanker flights to deliver the required total propellant mass. Finally, the thermal environment to which the lander is exposed on the lunar surface can be worse than that in space.

The present experience with cryogenic propellant maintenance in space is largely with the Centaur III upper stage of the Atlas V launch vehicle. Flight history for the uninsulated Centaur

III shows an average loss rate of about 2.4%/day for liquid oxygen (lox, LO_2) and 15%/day for liquid hydrogen (LH_2). Model results indicate that 3-layer multi-layer insulation (MLI) could reduce this to about 1.5%/day and 5%/day for LO_2 and LH_2, respectively. Theoretical results for active refrigeration indicate the possibility of achieving boiloff rates of as low as 0.1%/day. [24] (Flight history for the lox/methane propellant combination is not yet available.) Table 2 shows the effect of these various assumptions on long-term deep-space mission designs using cryogenic propellants.

%Propellant Remaining vs. Time					
Daily Propellant Loss Rate	Number of Days				
	1	5	30	60	90
15% (LH2, no insulation)	86	47	1.1	0.01	0
5% (LH2, 3-layer MLI)	95	78	22	5.0	1.1
2.5% (LO2, no insulation)	97.5	88	47	22	10.5
1.5% (LO2, 3-layer MLI)	98.5	93	64	41	26
0.1% (Active cooling, theoretical)	99.9	99.5	97	94.2	91.4

Table 2. Propellant Fraction vs. Time for Various Boiloff Rates

It is clear from Table 2 that a 60-90 day loiter requirement for a cryogenic lander is untenable with present technology; even a month would be impractical. By itself, a 5-day stay on the lunar surface (assuming the thermal environment to be no worse than in space) would pose a significant design challenge for a cryogenic propellant lander. The ability to maintain cryogenic propellants in proper condition for long periods of time in space is a critical technology for future space development but to assume that it will be available for near-term lunar missions is unwise.

Getting to the Lunar Surface

This is a complex issue that is important to explore carefully. To that end, it is helpful to examine Fig. 7, NASA's schematic diagram showing the approximate ΔV requirements for various paths to and from the lunar surface following the 3.2 km/s TLI ΔV for departure from LEO.[25] That requirement is common to all subsequent mission phases and will not be further considered here. Also ignored are variations in the TLI requirement due to the 18-year cycle of lunar orbit plane inclination from 18.4°-28.6° relative to the ecliptic plane. The present focus is on options for reaching the surface from an inbound Earth-Moon trajectory and for returning home.

Only "split missions" are discussed here, i.e., missions in which a crew return vehicle is left in a lunar orbit while the lander descends from that orbit to the lunar surface and returns to it for crew vehicle rendezvous and Earth return. The split can occur after both vehicles are injected into the same lunar orbit (as for Apollo) or because they are deployed separately from Earth (as for Artemis), but wherever it occurs, this separation generates critical operational constraints.

While Fig. 7 offers a very useful summary of various options for reaching the lunar surface from translunar coast, it can be misleading unless care is taken to understand certain points. The diagram depicts two essentially symmetrical "ladders", each with several "rungs" representing different options for reaching the lunar surface from the inbound translunar trajectory or returning to Earth on the outbound trans-Earth trajectory.

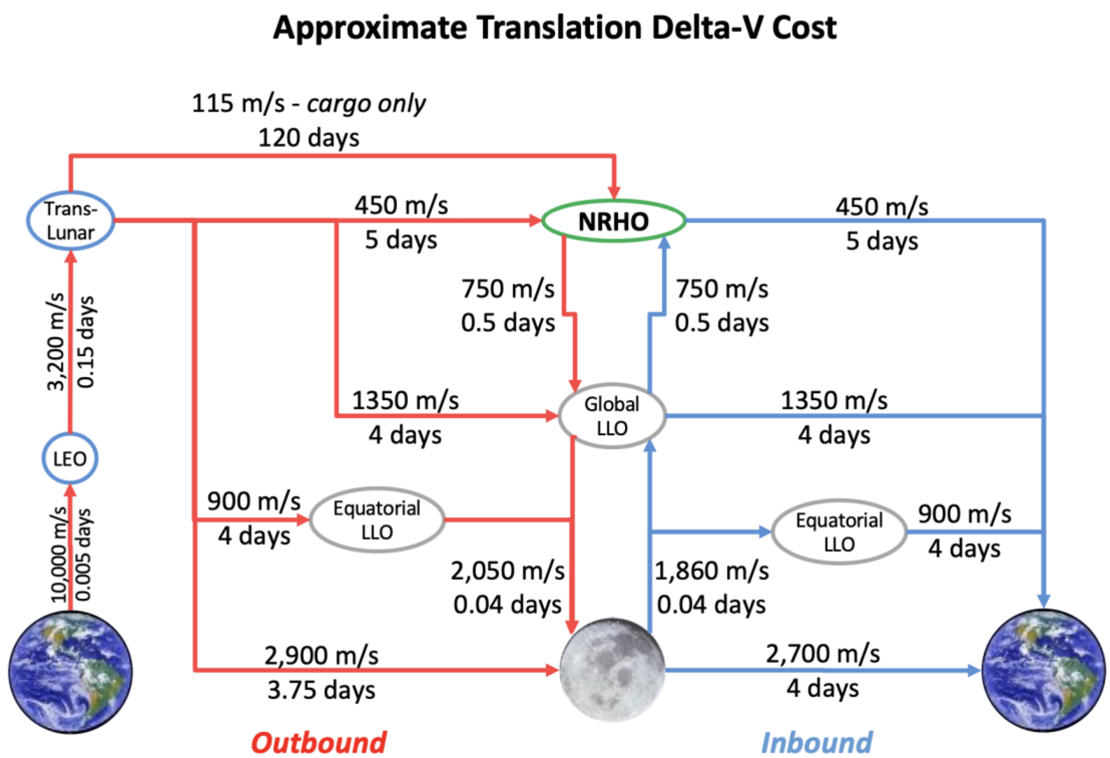

Fig. 7. ΔV Requirements for Alternative Lunar Landing Architectures[25]

The lowest rung of the red ladder represents a descent directly to the lunar surface from translunar coast and the lowest rung of the blue ladder represents a return to Earth directly from the lunar surface. While this technique was used for the landings of the robotic Surveyor missions of the 1960s, it is inapplicable to split missions such as Apollo or Artemis, where an Earth return vehicle, e.g., Orion, remains in a parking orbit while a separate lander descends to the surface.

A similar observation applies to the highest rung of the inbound ladder, which represents the family of ballistic lunar transfers[23] and is distinguished by the approximately 120-day transfer time and the low ΔV requirement of 115 m/s to inject into NRHO from translunar coast. As noted in Fig. 7, these trajectories may prove useful for future cargo delivery but are not of interest for human lunar missions due to their prohibitively long transfer times.

The other lunar landing options considered in Fig. 7 go through LLO as a “parking orbit”, used here as a general concept that can refer to several kinds of orbits. In this sense, both NRHO and LLO can be parking orbits and there can be more than one, depending on the architecture. LLO is a special case in this discussion, because it is common to all architectures considered here. Unlike Apollo, in the Artemis architecture the Orion crew vehicle remains in NRHO and does not itself transfer to LLO. However, the Artemis landers stage through LLO to set up the descent to the selected near-polar landing site, and similarly on ascent for proper phasing for the return to NRHO.

A key point noted in Fig. 7 is that the ΔV requirements for the descent to the lunar surface from a low lunar parking orbit and the reverse on ascent, shown as 2,050 m/s and 1,860 m/s respectively, are the same for any landing from a 100 km altitude circular parking orbit, whether polar (for global access) or equatorial (as in Apollo). This is the basic lander performance requirement, regardless of how the lander is delivered to its parking orbit. To simplify the subsequent discussion, this requirement is rounded to descent and ascent requirements of 2,050 m/s and 1,950 m/s respectively, for a total of 4.0 km/s as the minimal requirement for the lander element.[23] As with the 3.2 km/s ΔV TLI maneuver, this requirement is common to all options shown in Fig. 7 and is therefore not a discriminating feature among them.

The choice of parking orbit inclination, the angle at which it crosses the lunar equator, constrains the allowable landing locations because it corresponds to the highest latitude over which the lander passes. Further, because once established the orbit plane is fixed in inertial space (neglecting small perturbations that are not relevant here), the selection of a parking orbit also constrains certain arrival, landing and departure conditions. Below are summarized some of the key features and constraints associated with the various options.

The easiest and safest lunar landing missions are to the near-equatorial regions accessible from low-inclination (i.e., near-equatorial) parking orbits. Constraints such as those on landing sun-angle may apply, but from a purely orbital mechanics perspective, the crew can arrive in LLO and land at any time and, more importantly, can depart the surface for the crew return vehicle at any time. In turn, the crew return vehicle can depart for Earth at any time. These features are important when resource constraints or a hardware failure mandate an urgent return.

Insertion into near-equatorial LLO from translunar coast and departure from LLO on a trans-Earth insertion (TEI) trajectory can be accomplished for a ΔV expenditure of 900 m/s or less. It is worth noting that “near equatorial” is a loosely defined term: the latitude limit for a given mission is a function of the intended surface stay time (in a two-week cycle of optimal launch opportunities) and the lander ΔV reserve available for a plane change on ascent. As a guide to what constitutes an “equatorial” mission, the highest latitude Apollo-era landing was for the 3-day Apollo 15 mission to Hadley Rille, at 26° N.

Next easiest and safest are missions to the lunar polar regions. The lander can be deployed to polar LLO at any time with a lunar orbit insertion (LOI) ΔV of about 900 m/s. For the same ΔV expenditure, the crew vehicle can subsequently be inserted into the lander LLO for rendezvous on twice-monthly windows, with margin depending on selection of the translunar coast time at Earth departure and insertion ΔV reserves on arrival. This is not a safety issue and should not pose a significant operational constraint. Importantly, the crew can abort from near-polar sites to the safety of the mother ship at any time. However, the Orion/lander stack may be required to loiter in LLO for many days prior to Earth return, depending on the alignment of the polar LLO orbital plane with respect to the range of trans-Earth injection vectors reachable within the remaining Orion capabilities. If the surface stay is chosen to accommodate the optimal Earth return window and no urgent abort is required, the TEI ΔV from LLO is the same as that for near-equatorial orbits, about 900 m/s. If "anytime return" to Earth is required, the crew vehicle must have a remaining ΔV margin of up to 1,350 m/s to reach trans-Earth coast under the worst-case LLO plane alignment, i.e., 90° from the desired direction.[23]

Much more challenging are mid-latitude landing missions. Earth launch opportunities for crew vehicle rendezvous with a lander pre-deployed to LLO are the same as for polar missions, but once accomplished, the landing crew may need to loiter in LLO for many days, waiting for the intended landing site to rotate underneath the LLO plane. If "anytime launch" for crew vehicle Earth departure is required, a ΔV of up to 1,350 m/s[23] may be required for LLO insertion. Because the landing site must be in the plane of the crew vehicle orbit on ascent, return from the surface is possible only on two-week centers, again with margin on either side depending on the lander maneuvering reserve. Then, after rendezvous in LLO, it may again be necessary for the crew to loiter in LLO for alignment with the desired Earth departure trajectory. If "anytime return" is desired, then a departure ΔV of up to 1,350 m/s may be required. These challenges may make it advisable to postpose crewed mid-latitude missions until sufficient surface infrastructure has been robotically deployed to the intended site to allow a lengthy stay in the event of problems with either the lander or crew vehicle. Figure 8 illustrates the basic geometry.

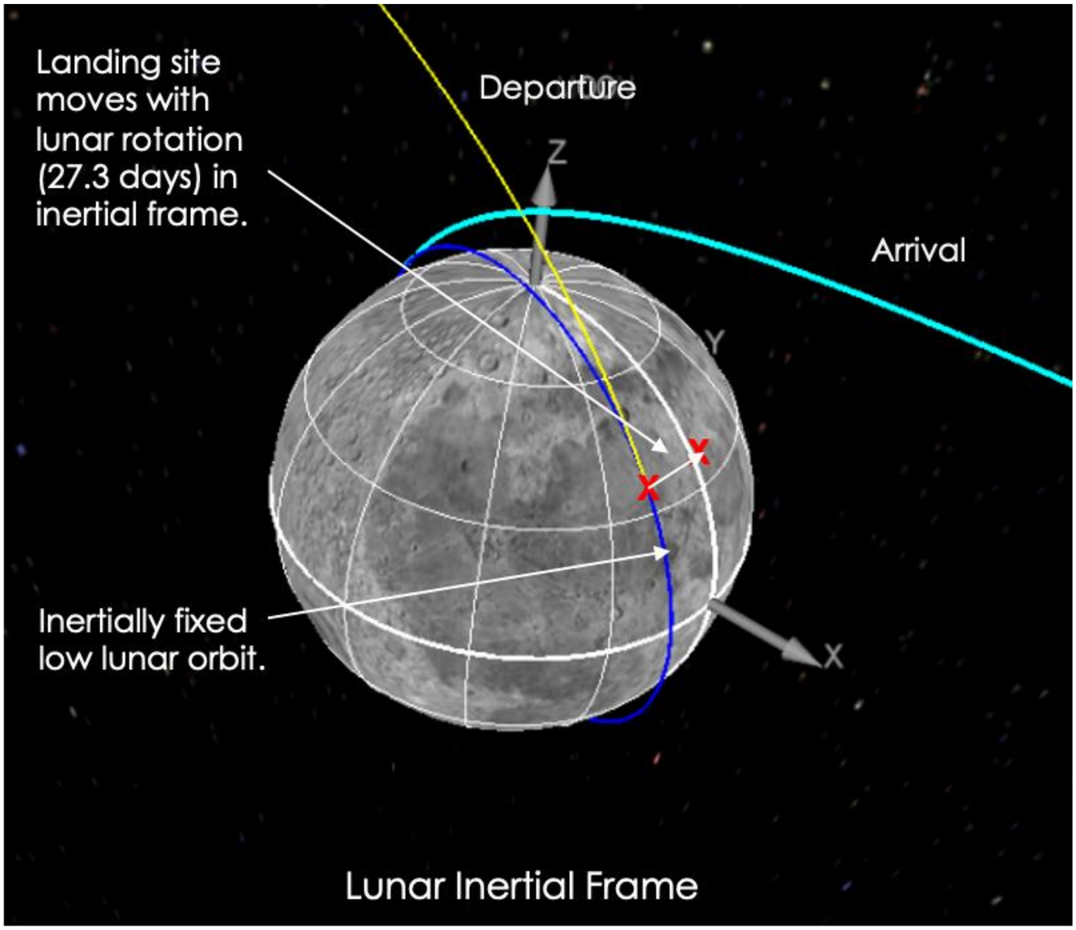

Fig. 8. Landing Site Translation Between Arrival and Departure at the Moon[32]

Staging Through NRHO

The above observations allow a more nuanced interpretation of Fig. 7, because not all "global access" missions pose the same constraints. While polar missions are more difficult than equatorial region missions, they are significantly safer and easier than mid-latitude landings. For non-equatorial split missions, the requirements for lander and crew vehicle should not be conflated. Lander deployment to a polar LLO never requires more than a 900 m/s insertion ΔV, and the lander is not required to carry propellant for Earth return. For the crew vehicle, especially for near-term lunar landing missions, it is almost certainly undesirable from a system engineering perspective to trade a significant amount of LLO insertion ΔV to protect an arbitrary schedule for departure from Earth. Thus, in most cases the 1,350 m/s "Global LLO" ΔV requirement shown in Fig. 7 should be regarded as a bounding requirement and not used as an architectural discriminator.

The one aspect of polar landing missions where the bounding "Global LLO" ΔV of 1,350 m/s noted in Fig. 7 may be important is for Earth return. As previously stated, for equatorial or polar region missions the crew can always depart the lunar surface for an urgent abort to LLO and for equatorial region missions can always return home. However, for polar missions where urgent return to Earth is required because of a problem with the crew vehicle rather than the lander, the situation is different. The orbital plane of the crew vehicle must be favorably aligned for the departure trajectory or a higher departure ΔV is required, varying on a two-week cycle between 900-1,350 m/s, discussed later in more detail. The Artemis architecture solves this problem with a rendezvous of crew and lander in NRHO prior to the transfer to polar LLO, but at the price of other constraints which, as discussed here, are of greater concern from a system perspective.

These requirements are summarized in Table 3. It is seen that the use of NRHO as a node in the architecture imposes a net additional ΔV requirement of more than 1,000 m/s on the lander as compared with staging directly through LLO, assuming in all cases that the lander carries its own LOI propellant, again to be discussed later. From a system engineering perspective, this is an undesirable constraint to impose on the lander element of the architecture. From the rocket equation for a single-stage lander (per the Artemis contractor designs) and ΔV = 1,000 m/s and g = 9.81 m/s:

$$\begin{aligned} \frac{M_i}{M_f} &= e^{\Delta V/gI_{sp}} \\ &= 1.2542 \text{ for } \mathrm{LO_2/LH_2}\ \left(\mathrm{I_{sp}} = 450\ \mathrm{s}\right) \\ &= 1.3273 \text{ for } \mathrm{LO_2/CH_4}\ \left(\mathrm{I_{sp}} = 360\ \mathrm{s}\right) \end{aligned} \tag{1}$$

e.g., a gross mass increase of 25% for a hydrogen-fueled design and 33% with methane fuel. The mass penalty would be less for a two-stage lander (not contemplated for Artemis by either of the selected contractors) but is still significant.

The lander is at the far end of the chain of maneuvers (the "ΔV gear train") required to reach the lunar surface and return to Orion. While not as important as the concerns about crew safety, mission reliability and cryogenic fuel storage raised previously, it is good engineering practice to absorb the ΔV penalty for lander LLO insertion elsewhere in the maneuver chain. The justification for failing to do so in the Artemis architecture is not apparent.

An architectural approach that addresses this and the other concerns raised here is outlined below. For purposes of comparison with Artemis III, only the requirements for a polar region mission are considered in detail, with occasional reference to corresponding features of equatorial or mid-latitude missions when informative.

Lander/Crew Vehicle Split-Mission ΔV Requirements Summary									
Mission Design	ΔV (m/s)								
	Lander (w/Integrated ΔV Propellant)					Crew Vehicle			
	LOI	Transfer to LLO	Descent /Ascent	Return from LLO	Lander Total	LOI	TEI	Crew Vehicle Total	Total Mission ΔV
Near-Equatorial	900	0	4,000	0	4,900	900	900	1,800	6,700
Global LLO (min/min)	900	0	4,000	0	4,900	900	900	1,800	6,700
Global LLO (min/max)	900	0	4,000	0	4,900	900	1,350	2,250	7,150
Global via NRHO	450	750	4,000	750	5,950	450	450	900	6,850

Table 3. ΔV Requirements Summary for Split-Mission Lunar Landing Options

Dual-Launch Lunar Landing Architecture

The approach proposed here employs two Space Launch System (SLS) missions launched from Earth to LLO, each augmented with a presently existing (though soon to be out of production) Centaur III upper stage to be used for LOI, as outlined below and depicted in Figure 9.

1. The SLS Block 2 cargo variant (46 mT to TLI[26]) launches a payload consisting of an uncrewed, fully fueled, two-stage storable propellant lander and a partially fueled Centaur III LOI stage.
2. The Centaur III injects the 32 mT lander into polar LLO to await rendezvous with crew in Orion. There is no cryogenic propellant storage limit; lander stationkeeping ΔV is less than 100 m/s/yr.[27]
3. At a later time, the Orion crew vehicle is launched on an SLS Block 2 crew variant (43 mT to TLI[22]), also with a Centaur III LOI stage, for insertion of the fully fueled Orion into LLO, followed by rendezvous with and crew transfer to the lander.
4. The lander descends to the surface and the crew executes its mission. No fixed surface stay is required to permit return to the crew vehicle in LLO for equatorial or polar region missions. For polar region missions, the window for Earth return is constrained by the combination of allowable surface stay and orbital loiter and the Orion maneuvering reserve (up to a maximum of 1,350 m/s). For mid-latitude missions, the same observations about LLO loiter for return to Earth apply, and there are additional constraints on the surface stay to accommodate crew vehicle rendezvous, as previously discussed.

5. The crew returns to LLO in the lander ascent stage for rendezvous with and transfer to Orion, following which the ascent stage executes a controlled surface disposal maneuver if desired.
6. The crew returns to Earth at a time consistent with Orion LLO loiter capability and maneuvering reserve.

This approach is based on the observation that the lander design poses the most challenging requirements and should drive the mission design. Given the approximately 315 second specific impulse attainable with storable propellants, it is likely advantageous for the lander to carry only the propellant required for descent/ascent, which for that reason is assumed here. The use of an LOI stage also decouples the lander design from the method of inserting it into lunar orbit, a potentially attractive feature. Finally, the LOI stage offers potential additional value, as discussed below.

For a liquid oxygen/liquid hydrogen (LO_2/LH_2) lander design, the trade on gross launch mass likely favors an integrated design, with the lander carrying its own LOI ΔV propellant. However, as discussed above, it is unrealistic from a system engineering perspective to impose a significant loiter requirement, almost always a feature of a split mission, on a cryogenic element, even if the gross mass trade favors an integrated lander design.

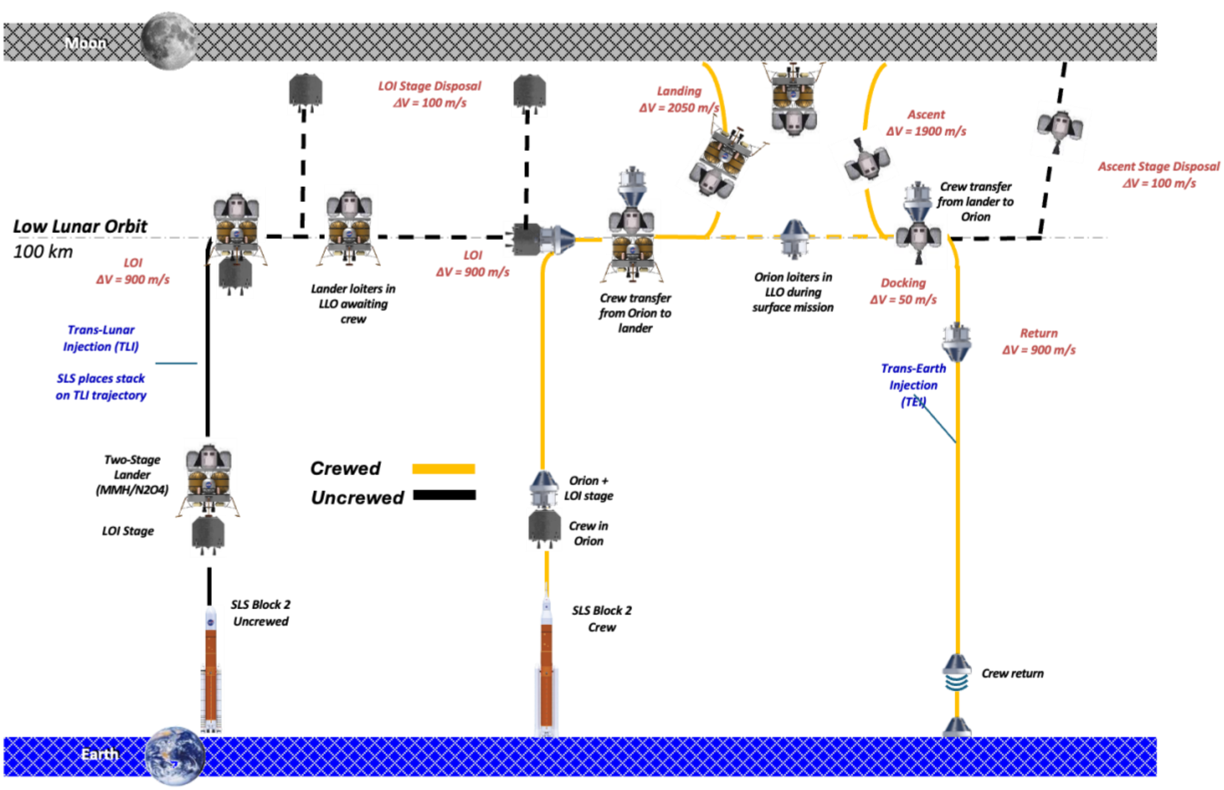

Fig. 9. Dual-Launch Concept of Operations

With this approach, the lander penalty for staging through NRHO in comparison to doing so through LLO is even more severe than shown in Table 3. If the lander is injected into lunar orbit (either NRHO or LLO) via a separate LOI stage (or is refueled in NRHO as in the Artemis V architecture), the lander departing from NRHO (whether from Gateway or not) must carry additional propellant for the 1,500 m/s round trip to LLO. In this case, the lander gross mass penalty for staging through NRHO is

$$\begin{aligned} \frac{M_i}{M_f} &= e^{\Delta V/gI_{sp}} \qquad (2) \\ &= 1.4047 \text{ for } \mathrm{LO_2/LH_2} \; (\mathrm{I_{sp}} = 450 \text{ s}) \\ &= 1.5292 \text{ for } \mathrm{LO_2/CH_4} \; (\mathrm{I_{sp}} = 360 \text{ s}) \end{aligned}$$

or about 40% and 53% for lox/hydrogen and lox/methane, respectively. It is unwise to impose such a requirement on the most challenging element of the architecture.

Lander Deployment

They key constraint for this mission design is the allowable mass that can be injected into LLO with the Centaur III stage while remaining within the SLS Block 2 specification of 46 mT to TLI.

The relevant specifications for the Centaur III, the single-engine Atlas V upper stage employing an RL-10C-1 lox/hydrogen engine, are[28]:

I_{sp}	–	450.5 s
Dry Mass	–	2.247 mT
Propellant	–	20.83 mT (max)
Gross Mass	–	23.08 mT
Diameter	–	3.05 m
Length	–	12.68 m

Assuming a representative 100 km altitude circular parking orbit as noted earlier and applying the rocket equation for the LOI maneuver

$$\frac{M_f}{M_i} = e^{-\Delta V_{LOI}/gI_{sp}} = \mathcal{R}_{LOI} \qquad (3)$$

with

$$\begin{aligned} M_I &= 46 \text{ mT} = \text{SLS Block 2 TLI maximum payload} \\ \Delta V_{LOI} &= 950\, m/s = 900\, m/s \text{ insertion } \Delta V + 50\, m/s \text{ margin} \\ g &= 9.81\; m/s^2 \end{aligned}$$

Thus,

$$\mathcal{R}_{LOI} = 0.8066 \tag{4}$$

hence

$$M_f = 37.1 \text{ mT} = \text{Net payload to 100 km LLO} \tag{5}$$

and

$$M_p = M_i - M_f = 8.90 \text{ mT} = \text{Propellant required for lander LOI} \tag{6}$$

To account for propellant boiloff from the Centaur III on a nominal 4-day translunar coast, note that the total LO_2/LH_2 propellant load is apportioned according to the RL-10C-1 oxidizer/fuel mixture ratio of 5.5:1.[29]. Thus, the masses of oxidizer and fuel are respectively

Mass of LO_2 = 7.529 mT
Mass of LH_2 = 1.369 mT

As a first approximation, the lox at about 90 K warms the liquid hydrogen fuel to its boiling point of about 20 K, causing it to vent to relieve excess tank pressure. Assuming a 3-layer MLI insulation wrap as discussed earlier, about 5% of the fuel will boil off each day. Including a day of margin, about 25% of the original LH_2 propellant, or 342 kg, will be lost during translunar coast. The total fuel load can then be conservatively sized as:

1.369 mT = LH_2 mass required for LLO insertion
+ 0.342 mT = 5 days fuel loss due to boiloff
1.711 mT = LH_2 mass required at TLI

Rounding up to 2.0 mT of fuel provides additional margin for unusable fuel and protects against the possibility of an oxygen-rich shutdown. It is similarly conservative to assume an 8.0 mT oxidizer load, for a total propellant load of 10.0 mT. The mass budget for the lander deployment to LLO is then:

SLS Block 2 TLI Payload		46.0 mT
Less: Dry Mass, Centaur III	(2.3 mT)	
Less: Propellant Mass	(10.0 mT)	
Less: Airborne Support Equipment	(1.7 mT)	
Subtotal, LOI Requirements		(14.0 mT)
Maximum Allowable Lander Mass		32.0 mT

For reference, the Apollo 17 lunar module mass was 16.2 mT, excluding the lunar rover it carried, and sustained a crew of two for three days on the Moon.[30] Conservative scaling suggests that a 4-crew lander using the same propellant combination can be built for no more than twice the mass of a 2-crew version. Nearly all transportation systems enjoy economies of scale; it is unlikely that lunar landers will prove to be an exception.

As an interesting comparison, a storable propellant lander design with integrated LOI capability at a specific impulse of 315 seconds yields about the same lander mass deployed in LLO, albeit with a higher dry mass and therefore less net useful payload. A separate LOI stage offers other potential benefits as well, yet to be discussed, but as both techniques will work and as the lander element is not yet designed, it would be a useful trade study to determine the best approach for lander deployment to LLO.

Crew Vehicle Deployment

Applying a similar analysis as above to the Orion crew vehicle with NASA's published data[31] for the Artemis II circumlunar mission (in SI units),

Orion Post-TLI Mass – 25,900 kg
Usable Propellant – 8,635 kg

Rounding conservatively for analysis purposes, it is assumed that Orion post-TLI mass is

$$M_{Orion} = M_{fO} = 27 \text{ mT}$$

Using the same LOI ΔV and Centaur III parameters as above and allocating 700 kg margin yields the mass to be inserted into LLO, M_f = 30 mT. From the rocket equation, the mass to TLI is then

$$M_{iO} = M_{fO} e^{\Delta V / g I_{sp}} = 37.2 \text{ mT} \qquad (7)$$

The required propellant load is

$$M_{pO} = M_{iO} - M_{fO} = 7.2 \text{ mT}$$

which is rounded here to 8 mT for additional margin including that for propellant boiloff. The combined Orion/Centaur stack mass of 38 mT is thus well within the SLS Block 2 (crew variant) limit of 43 mT to TLI[22]. This is helpful because the configuration will likely require additional structural support beyond the existing SLS payload adaptor and accommodations for co-manifested payloads.[31,36]

Coming Home

After ascent from the lunar surface, rendezvous with and crew transfer to Orion, the TEI maneuver is performed to depart LLO, escape from the Moon's gravity and return to Earth. Recapping earlier comments, for a polar region mission it is necessary to depart at one of the two opportunities each month when the LLO plane is sufficiently well aligned with the required departure direction that an acceptable departure ΔV can be had.

The required departure ΔV also depends on the desired transfer time, which is difficult to calculate accurately for a Moon-Earth trajectory. A spacecraft departing the lunar environment spends a significant amount of time subject to the gravitational influence of three bodies – Earth, Moon and Sun – none of which is completely dominant. The relatively simple two-body orbital mechanics analyses that work well for a satellite near the Earth are not adequate in this case and it is necessary to resort to more complex methods that are beyond the scope of this discussion.[32] The results are shown in Table 4 and Figure 10, which give the required transit time vs. the incremental ΔV that is required to be added to the 1.633 km/s circular orbit velocity for TEI from a 100 km altitude LLO.

Transfer duration (days)	Departure Δv (km/s)
2	**1.33**
2.5	**1.08**
3	**0.94**
3.5	**0.87**
4	**0.83**
4.5	**0.81**
5	**0.80**

Table 4. ΔV vs. Trip Time for Departure from 100 km Circular Lunar Orbit[32]

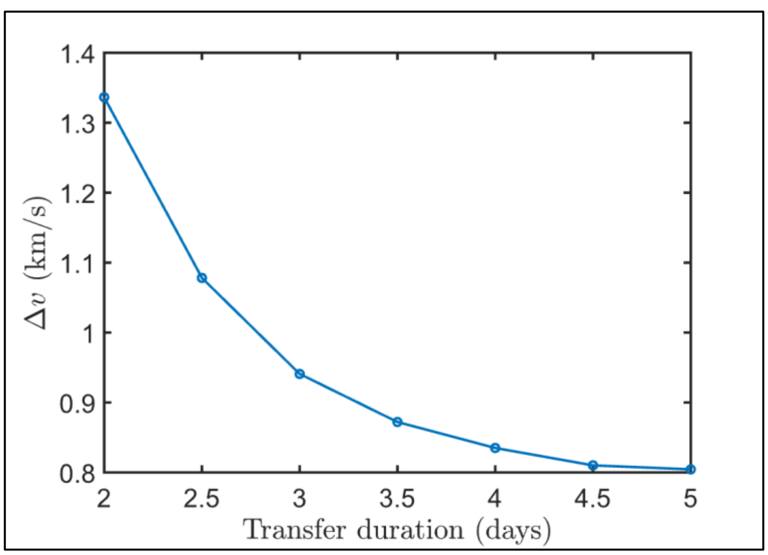

Fig. 10. ΔV vs. Trip Time for Departure from 100 km Circular Lunar Orbit[32]

It is seen that a LLO departure ΔV of 900 m/s or less is suitable for transfer times of about 3.25 days or more. The Orion main engine is an Aerojet AJ10-190 storable propellant engine well characterized from its use on the Space Shuttle.[33] Key Orion propulsion system parameters are:

M_{prop} = 8,635 kg (usable)
I_{sp} = 316 s

which yields a total ΔV capability for the spacecraft of 1,195 m/s. Conservatively allocating 195 m/s for rendezvous, proximity operations, docking, course corrections and reserve leaves 1,000 m/s for the TEI ΔV and subsequent minor course corrections.

With this, the calculation of the allowable departure window around the twice-monthly optimum alignment is straightforward. As noted earlier and neglecting small perturbations, the Orion LLO plane is fixed in inertial space as it revolves with the Moon around the Earth in a 27.3-day period. Thus, the ΔV requirement to compensate for misalignment of the plane varies sinusoidally over that period from the optimum to the worst case, i.e., orthogonal to the desired direction. For transfer times in the range of 3.25-3.75 days, there are two five-day windows in each lunar month when return to Earth is possible within the allocated 1,000 m/s Orion TEI capability, four days each month where return may be possible depending on fuel consumption during the mission, and two seven-day windows where the required ΔV exceeds or is unacceptably close to Orion's maximum performance capability. The worst-case loiter (whether in orbit or by extending the surface stay) does not exceed eight days.

Optimizing LOI Stage Utility

To this point the LOI stage, for which the Centaur III is assumed here, has been treated as a single-purpose element of the architecture, included to perform spacecraft insertion into LLO from translunar coast. The assumed constraints are the required LOI ΔV and the 46 mT maximum TLI payload capacity of the SLS Block 2, with the allowable lander mass a derived quantity. This approach offers the merit of simplicity but does not take advantage of the full capability of the Centaur III.

An alternative approach is to assess the system performance if the Centaur III is fueled to its maximum capacity, resulting in a launch mass beyond what the SLS Block 2 can deliver to TLI, with the Centaur then used to make up the difference. SLS performance would be degraded but that of the Centaur would be enhanced, leading to the question of what payload increase might be possible. An exact answer depends upon launch vehicle performance analysis that is well beyond the scope of this work. However, a good approximation can be obtained subject to the assumption that the SLS ascent to LEO with the heavier payload is substantially the same as in the baseline case, except that the Exploration Upper Stage (EUS)[34,35,36] arrives in LEO with less fuel than in the nominal case.

As noted in Fig. 7, the SLS Exploration Upper Stage (EUS) must provide a ΔV_{TLI} of about 3.2 km/s from LEO for TLI, with the exact value depending upon details of orbital mechanics and launch epoch that are not important here. With additional Centaur propellant and lunar payload mass, the stack will be heavier than 46 mT and cannot reach TLI velocity. However, the EUS provides the same total impulse regardless of the mass of the stack. Thus, applying the rocket equation for the EUS at burnout for the base case, the mass ratio for the EUS and its payload stack is:

$$\frac{M_F}{M_I} = \frac{(M_{EUS} + M_i)}{(M_{EUS} + M_{pEUS} + M_i)} = e^{-\Delta V/gI_{spEUS}} = \mathcal{R}_{EUS} \tag{8}$$

where

$\mathcal{R}_{EUS}$ = Mass ratio for EUS with 46 mT TLI payload.
M_F = Burnout mass of EUS plus payload stack, base case.
M_I = Initial mass of EUS plus payload stack, base case.
M_{EUS} = EUS empty mass = 14.11 mT.[35]
M_{pEUS} = EUS propellant mass = 129 mT.[35]
M_i = Initial mass of 32 mT lander plus partially fueled Centaur, etc.
= Maximum SLS Block 2 paylolad to TLI = 46 mT.
I_{spEUS} = Specific impulse of EUS RL-10C-3 = 460.1 s.[29]

hence $\mathcal{R}_{EUS}$ and therefore the achievable ΔV for the base case are known.

When additional Centaur propellant and lunar payload mass are added, the EUS mass ratio becomes:

$$\mathcal{R}^+_{EUS} = \frac{M^+_F}{M^+_I} = \frac{\left(M_{EUS} + M_i + \Delta M_p + \Delta M_{PL}\right)}{\left(M_{EUS} + M_{pEUS} + M_i + \Delta M_p + \Delta M_{PL}\right)} = \frac{\left(M_F + \Delta M_p + \Delta M_{PL}\right)}{\left(M_I + \Delta M_p + \Delta M_{PL}\right)}$$

$$= e^{-(\Delta V - \Delta V^+)/gI_{spEUS}} = e^{-\Delta V/gI_{spEUS}}\, e^{\Delta V^+/gI_{spEUS}} = \mathcal{R}_{EUS}\, e^{\Delta V^+/gI_{spEUS}} \tag{9}$$

Dividing numerator and denominator on the right by M_I and recognizing that $M_F/M_I = \mathcal{R}_{EUS}$, Eq. (9) becomes

$$e^{\Delta V^+/gI_{spEUS}} = \frac{\mathcal{R}^+_{EUS}}{\mathcal{R}_{EUS}} = \frac{1 + \dfrac{\Delta M_p + \Delta M_{PL}}{\mathcal{R}_{EUS} M_I}}{1 + \dfrac{\Delta M_p + \Delta M_{PL}}{M_I}} \tag{10}$$

where

ΔM_p = Additional propellant mass in fully fueled Centaur = 10.83 mT.

ΔM_{PL} = Additional lunar payload mass.
ΔV^+ = Velocity to be gained by the Centaur III + ΔM_{PL} to reach ΔV_{TLI}.
= Centaur velocity increment to compensate for additional EUS payload.

For the Centaur, with the added payload mass above the SLS Block 2 TLI limit, two maneuvers approximately 3.5 days apart are performed to inject the increased payload mass, $M_f + \Delta M_{PL}$, into LLO. The first of these is ΔV^+, to reach TLI velocity after EUS burnout, and the second is ΔV_{LOI}, for injection into LLO following translunar coast. Neglecting the change in mass due to propellant boiloff during the translunar coast and assuming an LOI burn to depletion, the mass ratio for the Centaur in the new application is

$$\mathcal{R}_C^+ = \frac{M_f^+}{M_i^+} = \frac{(M_f + \Delta M_{PL})}{(M_i + \Delta M_p + \Delta M_{PL})} = e^{-(\Delta V^+ + \Delta V_{LOI})/gI_{sp}} = e^{-\Delta V_{LOI}/gI_{sp}}\, e^{-\Delta V^+/gI_{sp}} \qquad (11)$$

Dividing numerator and denominator by M_i and applying Eq. (3),

$$\mathcal{R}_C^+ = \frac{\mathcal{R}_{LOI} + \dfrac{\Delta M_{PL}}{M_i}}{1 + \dfrac{\Delta M_p + \Delta M_{PL}}{M_i}} = \mathcal{R}_{LOI}\, e^{-\Delta V^+/gI_{sp}} \qquad (12)$$

or

$$e^{\Delta V^+/gI_{sp}} = \frac{\mathcal{R}_{LOI}}{\mathcal{R}_C^+} = \frac{\mathcal{R}_{LOI} + \dfrac{\Delta M_{PL}}{M_i}}{1 + \dfrac{\Delta M_p + \Delta M_{PL}}{M_i}} \qquad (13)$$

The system of Eqs. (10) and (13) can be solved numerically for ΔM_{PL} and ΔV^+. Recognizing that minor differences exist among EUS performance parameters as cited in the literature, for those used here,

ΔM_{PL} = 5.8 mT
ΔV^+ = 722 m/s

Because minor trajectory differences can matter to the overall launch vehicle performance, a detailed simulation is required to determine an exact result. However, it seems likely that a payload increase of at least several tons to LLO would result from the inclusion of an LOI stage in the system architecture.

Summary and Final Thoughts

The dual-launch lunar mission architecture presented here offers a simpler, lower crew and mission risk option to meet NASA's top-level requirements for the Artemis program – four crew to a lunar pole for a week – while remaining within (or very close to) Orion's present 21-day in-space operational limit[37], even with a worst-case loiter in LLO for TEI. The mission design allows return from the surface to Orion at any time. A 3.5-day return to Earth is available at any time for an equatorial region mission, or with no more than an 8-day loiter in LLO for a polar region mission. Many fewer and far less complex operations are required than for either of the current Artemis architectures, and no new technology such as in-space cryogenic propellant transfer or long-term storage is required.

Many design details remain to be defined. Examples include structural support interfaces between Centaur and Lander/Orion, modification of the SLS launch platform to accommodate additional stack height, similar alterations to the Vertical Assembly Building at NASA-KSC, etc. Such details are critical but are beyond the scope of the present discussion.

The present mission design was selected for its simplicity (if that term can reasonably be applied to a lunar landing mission) but offers many possible options for improvement that should be assessed in more detail than can be done here. We begin with further examples of the need to avoid unnecessary risk in achieving mission success:

1) For early lunar return missions, especially the first, would it be wise to minimize mission elapsed time by selecting the surface stay to enable a favorable TEI geometry with minimal LLO loiter?

2) Is it wise to target a lunar pole for the first lunar landing in six decades? The poles are characterized by poor viewing and lighting conditions for landing, more difficult terrain, unknown surface characteristics and restricted Earth return opportunities as compared to the equatorial regions. Staging through NRHO is not well suited to equatorial missions, whereas the present approach accommodates them. Is it wise to begin a program of human lunar return by taking on any challenge not absolutely required? How does doing so advance either the tactical or strategic goals of the U.S. human spaceflight program?

We also offer several more technically oriented suggestions for further consideration:

1) A sensitivity study should be performed to assess the parameter space within which the mission design presented here is applicable and to optimize the design within that space.

2) The approximate analysis presented here for the estimated payload increase obtained with a fully fueled Centaur III LOI stage should be refined through re-optimization of the SLS ascent profile to take advantage of the upper stage for both LOI and TLI.

3) The 100 km altitude circular LLO parking orbit assumed here for convenience is unlikely to be the optimum choice. For example, would it be better engineering practice to

select a slightly higher parking orbit to provide a more generous TEI window for Orion, at the expense of imposing an increase on the lander ΔV requirement? What is the best balance between these competing requirements? Is a circular orbit the best option? The system-level benefit from a careful assessment of parking orbit design could be significant.

4) Improved versions of the legacy RL-10 exist and will be used on the United Launch Alliance Vulcan Centaur and the SLS Block 2 Exploration Upper Stage.[29,34,35,36] Is it worthwhile to re-engine the Centaur III for the specialized LOI stage of this architecture to obtain the additional specific impulse, approximately 10 seconds, that could be available?

5) Storable "green" propellants[38] have been demonstrated in flight[39] and are expected to come into routine use over the course of the next decade. Given the advantage offered in terms of higher specific impulse, lower toxicity and greater ease of handling, is this a technology upgrade that should be considered for the lunar lander and, if so, does that meaningfully affect the trade between the benefits of an integrated lander design vs. those associated with the use of the LOI stage, especially the additional payload enabled by optimal use of that stage?

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