

Revisited Virtues of Lunar Surface Rendezvous (LSR)

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Current "1.5-launch" lunar exploration mission architecture is reviewed and compared with a proposed "2.0-launch" lunar surface rendezvous technique enabled by recently augmented Ares V launch capability. Lunar surface rendezvous architecture offers improved reliability, robustness, contingency options, and life cycle economies, along with scalable flight hardware components leading to exploration destinations beyond the Moon.

I. Nomenclature

| | | |
|--------------|---|---|
| g | = | Earth surface gravity acceleration = 9.80665 m/s^2 |
| I_{SP} | = | stage-unique specific impulse |
| m_D | = | stage-unique inert mass (includes m_{PS} ; excludes m_L) |
| m_L | = | stage-unique initial usable propellant mass = $m_T - m_Y - m_D$ |
| m_{PS} | = | stage-unique propellant subsystem mass (excludes m_L) |
| m_S | = | stage-unique initial stage mass = $m_T - m_Y$ |
| m_Y | = | stage-unique payload mass |
| m_T | = | stage-unique initial total mass |
| R | = | stage-unique effective propellant subsystem to initial propellant mass ratio = $m_{PS} / (m_S - m_D)$ |
| v_{EX} | = | stage-unique exhaust velocity = $g I_{SP}$ |
| Δm_L | = | stage-unique m_L increase |
| Δv | = | stage-unique velocity change magnitude |

II. Introduction

Current US space exploration policy envisions a "1.5-launch" piloted lunar mission architecture. It entails a heavy-lift cargo-only Ares V launch placing an Earth Departure Stage (EDS), together with an attached Lunar Surface Access Module (LSAM, also named *Altair*), into low Earth orbit (LEO). An Ares I launch delivers a crew of 4 aboard a Crew Exploration Vehicle (CEV, also named *Orion*) to LEO. Within 4 days of the Ares V launch, CEV performs a rendezvous and docking with the LSAM/EDS, followed by EDS trans-lunar injection (TLI) of the docked CEV/LSAM spacecraft ([1], chart 10). Over the next 3 to 4 days, the spent EDS is jettisoned before LSAM places the docked spacecraft into a properly oriented low lunar orbit (LLO) plane. After the crew transfers to LSAM, CEV is left to loiter in LLO while the LSAM descends to the lunar surface and exploration operations are conducted. Upon conclusion of lunar surface operations, the crew is launched aboard LSAM's ascent stage and rendezvous/docking with the CEV in LLO is achieved. Following crew transfer to the CEV, LSAM's ascent stage is undocked and remotely piloted to a lunar impact. The CEV's LLO plane is then properly oriented before Earth return is initiated, leading to atmospheric entry over the next 3 to 4 days. Immediately prior to entry, the CEV's Service Module (SM) is jettisoned, leaving the CEV Crew Module (CM) to undergo a controlled entry and landing at mission completion.

During 1.5-launch development thus far, Ares V performance requirements have been updated to compensate for Ares I performance deficiencies. This paper documents augmented Ares V capabilities now enabling the "2.0-launch" lunar surface rendezvous (LSR) architecture[†] proposed herein. With LSR, multiple vulnerabilities associated with the 1.5-launch mission profile are remedied. These vulnerabilities are enumerated in Section III, together with their mitigation by LSR. In Section IV, LSR development challenges and payoffs are identified.

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[†] Although the LSR technique is not new ([2], chart 5), its advocates have been few in number since an Exploration Systems Architecture Study (ESAS) report [3] was published by NASA in 2005.

Finally, vehicle mass estimates supporting a notional LSR mission are provided in Section V for reference and future refinement.

The LSR sequence of events outlined below is notable in at least two respects. First, crewmembers only reside in the CEV's CM habitat during outbound and return transits between Earth and the lunar surface. Second, crew and mission completion cargo are not brought together until the lunar surface destination is reached, eliminating orbit rendezvous and docking operations throughout the mission.

- 1) Ares V launches a cargo-only payload into LEO. This payload consists of a return consumables module (RCM) and descent module (DM) atop an EDS. Functionally, the DM may be regarded as a scaled-up LSAM descent stage.
- 2) Following a checkout period in LEO lasting at most 6 hours, EDS achieves TLI for the RCM/DM payload and is jettisoned.
- 3) With its RCM payload attached, the DM inserts into LLO and lands on the Moon 3 or 4 days after launch.
- 4) Following landing, an RCM functional integrity check, including lunar landing navigation aids (navaids), is performed as a prerequisite to crew launch. These navaids serve as a precision approach and touchdown radio and/or optical beacon for subsequent landings nearby.
- 5) Up to 6 months after RCM landing and checkout, Ares V launches a crewed CEV and DM into LEO atop an EDS. This CEV's SM has a scaled-up main engine thrust and propellant storage capacity with respect to the CEV currently being designed under Ares I performance constraints. The SM propellant load at launch is partial, intended only for contingency use prior to lunar landing.
- 6) Following a checkout period in LEO lasting at most 6 hours, EDS achieves TLI for the CEV/DM payload and is jettisoned.
- 7) With its CEV payload attached, the DM inserts into LLO and lands on the Moon adjacent to the RCM 3 or 4 days after launch.
- 8) In parallel with crew lunar surface operations, SM propellant and other CEV consumables are topped off from the RCM.
- 9) Using its DM as a launch pad, the CEV employs SM propulsion to insert into LLO and initiate return to Earth, targeting atmospheric entry within 3 to 4 days after lunar liftoff.
- 10) Shortly before atmospheric entry, the CEV's SM is jettisoned, leaving the CM to undergo controlled entry and landing prior to mission completion.

The author has conducted research documented by this paper in the interest of promoting safe, robust, and cost-effective human space exploration beyond LEO. This research was performed entirely with the author's personal time and facilities. Consequently, the broadest possible distribution of this paper without modification is encouraged free of charge to any recipient.

III. Current 1.5-Launch Vulnerabilities And Their LSR Mitigation

The following subsections are each dedicated to identifying a specific performance, mission success, or safety vulnerability associated with the current 1.5-launch lunar exploration architecture. Each subsection also documents attendant vulnerability reduction offered by an LSR mission profile.

A. Vulnerability 1: Two Dissimilar Launch Vehicles

Logistics, procedures, infrastructure, and costs associated with parallel Ares I and Ares V production, assembly, and launch will be a formidable and ongoing burden to lunar exploration. Although 2 Ares V launches in support of LSR appear more expensive than a 1.5-launch architecture, economies of scale and uniformity/consistency are likely to more than offset this disadvantage over only a few mission cycles.

Consider a time-critical crew rescue contingency under the 1.5-launch architecture. From current planning, it appears only one segment of Launch Complex 39 (LC-39), namely the pad having just launched the disabled crew, will be capable of launching the rescue CEV. Moreover, if the disabled crew has departed LEO, a second Ares V launch will be required to achieve rescue. With LSR architecture in place, a single Ares V launch, likely from either LC-39 pad, can effectively respond to the rescue call.

B. Vulnerability 2: CEV Launch/Rendezvous/Docking And The TLI Countdown Clock

Once Ares V launches EDS and its LSAM payload into the LEO plane targeted by 1.5-launch mission design, a "countdown clock" to TLI is running. Ares I must then launch the crew into a LEO permitting propellant-efficient CEV rendezvous and docking with LSAM at least a few hours before the countdown expires. If this deadline is not met, the lunar landing mission is aborted because another TLI opportunity is at least 9 days in the future[‡]. Although the TLI countdown only directly threatens lunar mission success, it may lead to timeline pressures for the crew and mission support personnel. These pressures could in turn trigger human errors or timeline delays threatening crew safety.

A TLI countdown clock is also running after any Ares V launch supporting LSR, but no other launch, rendezvous, or docking events are tied to this clock. The only prerequisite to an LSR TLI following LEO insertion is a successful systems checkout.

Because of the many critical events leading to a 1.5-launch TLI, mission planners will be inclined to stack the mission success "deck" by targeting the Ares V LEO insertion plane such that the interval between Ares V launch and TLI is about 4 days ([1], chart 10). This strategy imposes two performance-robbing consequences. First, LOX/LH2 boil off over 4 days reduces EDS Δv capability with respect to that available for TLI only a few hours after launch under an LSR timeline. Second, atmospheric drag over a planned 4-day interval ending at TLI leads to the following gravity losses[§].

- 1) To offset altitude decay over 4 days, Ares V and Ares I must insert their respective payloads to higher LEO altitudes than would be required of any Ares V launch supporting LSR TLI only a few hours later.
- 2) Altitude decay over 4 days is difficult to predict with precision, leading to 1.5-launch mission planning biases in Ares V insertion altitudes to protect minimum LEO altitude safety constraints. These biases will be toward still higher LEO insertion altitude targeting. Altitude decay is virtually nonexistent over the few hours between launch and TLI in an LSR timeline.

C. Vulnerability 3: Rendezvous And Proximity Operations Propellant Consumption

Any CEV propellant consumption required to achieve rendezvous and docking with the LSAM/EDS in LEO is unnecessary in an LSR mission. Likewise, any CEV propellant consumption required to achieve LSAM undocking or docking in LLO is unnecessary for LSR. Other than propellant needed to launch the ascent stage from the Moon's surface into LLO, LSAM propellant consumption required to achieve rendezvous and docking with the CEV in LLO is unnecessary to LSR.

Rendezvous and proximity operations propellant consumption is subject to dispersions requiring extra propellant allocation covering worse cases than mean consumption, particularly during periods of manual control. The LSR architecture is free from these uncertainties, with the possible exception of manual piloting during CEV/DM final descent to the lunar surface. However, given that an unpowered DM lunar landing with an RCM payload must be reliably automated, manual piloting during CEV/DM descent (particularly with functioning RCM nav aids) will probably be an unlikely contingency procedure.

D. Vulnerability 4: Docked CEV/LSAM During TLI

The pressurized CEV/LSAM docking interface must undergo compressive loads during a 1.5-launch TLI equivalent to 7 times the maximum encountered during Apollo Program missions [4]. This interface must therefore be designed with extra mass to bear TLI loads without loss of pressure or structural integrity. The CM and LSAM ascent stage carry this extra mass throughout their respective missions at the expense of propulsive efficiency.

[‡] During this interval, cryogenic liquid oxygen/hydrogen (LOX/LH2) propellant will boil off to the extent EDS change in velocity (Δv) capability falls below TLI requirements. Depending on atmospheric drag, LEO altitude decay may also lead to EDS destructive atmospheric entry before another TLI opportunity arises.

[§] In the context of performing TLI from a circular LEO, a gravity loss is incurred by launch to a LEO height any higher than necessary under crew/vehicle safety constraints. Because TLI only needs to raise one "side" of the LEO beyond the Moon's orbit, raising the other TLI side above the minimum height constraint conceptually embodies the gravity loss.

Because EDS payloads associated with LSR do not undergo any reconfiguration between launch and TLI, the same structural elements bearing relatively larger launch loads remain in place during TLI. These elements are jettisoned with the EDS following TLI, so they do not affect subsequent propulsive efficiency.

Since LSR eliminates all docking operations under normal circumstances, docking hardware function and mass appear superfluous in connection with this mission architecture. Assuming contingency crew rescue would not rely on docking to the disabled CM, eliminating this hardware in connection with LSR missions may be an advisable performance enhancement.

During a 1.5-launch TLI, the crew faces in a direction opposing EDS thrust and experiences accelerations exceeding 1 g in the "eyeballs-out" direction. Again, because LSR payloads are not reconfigured between launch and TLI, associated accelerations are consistently "eyeballs-in". Early EDS designs employed two J-2X engines but were subsequently scaled back to a less efficient single J-2X configuration, presumably because eyeballs-out accelerations exceeding 2 g were deemed unacceptable for the crew. With LSR imposing only eyeballs-in TLI accelerations, the more efficient dual-engine EDS configuration could be reinstated.

E. Vulnerability 5: CEV Planar Constraints In LLO

During a 1.5-launch mission, the "land anywhere; leave anytime" lunar exploration mantra is nowhere conflicted more than by CEV's sojourn in LLO. No less than three independent planar constraints apply to this LLO.

- 1) The LLO plane must contain the targeted lunar landing site at the desired site landing time.
- 2) The LLO plane must contain the targeted lunar landing site at the desired launch time from the site.
- 3) The LLO plane must contain the hyperbolic lunar escape asymptote targeting Earth return when return is initiated.

Although all three constraints apply to an LSR mission, the CEV's selenocentric angular momentum is effectively reduced to zero by landing on the Moon, decoupling Constraint 1 from the others. Since an LSR CEV is at the landing site when it launches from this site, Constraint 2 is rendered trivial and satisfied by definition. With CEV enjoying zero selenocentric angular momentum at the landing site, LSR architecture can target virtually any LLO plane at launch from the lunar surface without any performance penalty^{**}, readily satisfying Constraint 3.

To manage Constraint 2, 1.5-launch mission planning will generally need to allocate CEV propellant for the purpose of changing the LLO plane. This allocation will be minimal if the lunar landing site is at low latitude or near either rotational pole, but the "land anywhere" requirement can result in Constraint 2 plane changes approaching 90° for mid-latitude sites. Large Constraint 2 plane changes associated with mid-latitude sites are likely when a "leave anytime" contingency arises and LSAM ascent stage launch must occur days earlier than planned.

Successfully managing Constraint 2 during a 1.5-launch mission only clears the way for docking the LSAM ascent stage to the CEV. Regardless of lunar landing site location and depending on launch time from it, Constraint 3 may be far from satisfied at docking. Large LLO plane changes required to satisfy Constraint 3 generally entail multiple propulsive impulses, often delaying departure for Earth by a day [5]. In time-critical contingencies, Earth return delays imposed by Constraint 3 could have severely undesirable consequences.

No LSR advantage vice a 1.5-launch architecture is offered in managing Constraint 1. Once LLO supporting lunar landing is established, however, LSR total Δv associated with subsequent CEV or RCM operations is essentially fixed regardless of landing site location or timeline variations. This LSR consistency confers powerful flexibility on nominal mission plans and unanticipated contingency responses alike. Land anywhere; leave anytime lunar exploration appears realizable with LSR, but marginal performance cases abound in the 1.5-launch architecture.

F. Vulnerability 6: Unoccupied CEV Loitering In LLO

In a 1.5-launch mission, the CEV is crewless in LLO throughout the interval in which lunar surface operations are conducted. During this loiter period, potentially extending for up to six months, CEV systems must be

^{**} The Moon's slow inertial rotation rate equates to a maximum eastward selenocentric lunar surface velocity of only 4.626 m/s at the equator. Consequently, variations in lunar launch efficiency with azimuth are virtually nonexistent compared to those associated with Earth launch.

maintained for crewed Earth return while direct communications with Earth and power from the Sun are unavailable nearly half of each two-hour orbit^{††}. Even with communications relay infrastructure eventually supporting remote telemetry and commanding with CEV over the lunar far side, response to CEV contingencies during the loiter period may be subject to delayed response and complex commanding procedures. At the very least, autonomous operation requirements arising from loiter in LLO will add cost and complexity to CEV design.

Depending on how the crew deploys during LSR lunar surface operations, the CEV in this mission profile may never be unoccupied. Many LSR missions can be planned such that continuous direct communications with Earth or up to two weeks of continuous sunlight are experienced by a CEV on the lunar surface. Even if unoccupied, the CEV is in a much less dynamic environment and a much more accessible location to the crew if stationed on the lunar surface, as opposed to LLO, during lunar surface operations.

In the event of a solar flare or other similar radiation hazard, crew shielding in the CM is expected to be far more protective than that in the LSAM ascent stage cabin. If lunar surface operations extend more than a week, most of the mission duration will be spent in this phase. Consider a solar flare radiation contingency during lunar surface sortie operations without a base habitat to serve as "storm cellar". With the CM nearby, as opposed to loitering in LLO, LSR offers a significantly reduced crew radiation risk with respect to that incurred during a 1.5-launch mission. In addition, by providing a relatively safe haven on the lunar surface, LSR lunar surface sortie operations are not vulnerable to an unplanned and time-critical LLO insertion/rendezvous/docking abort to CEV as are equivalent 1.5-launch operations.

G. Vulnerability 7: LSAM Ascent Stage Disposal Via Lunar Impact

In a 1.5-launch mission, LSAM ascent stage propellant must be allocated to a de-orbit impulse targeting lunar impact disposal. If this impulse cannot be performed after ascent stage jettison, lunar gravity perturbations will eventually bring about an uncontrolled impact. Although targeted ascent stage impacts may be of limited scientific value to future seismic sensors emplaced on the lunar surface, any impact event is not responsible stewardship of a pristine and nearly primordial lunar environment over the long term. If uncontrolled, an ascent stage impact could also threaten inadvertent destruction of future lunar surface infrastructure. Threatened infrastructure might be owned by another space-faring nation or agency, posing serious liability consequences. An LSR mission does not entail any lunar impact. If desired for seismology purposes, up to two EDS targeted lunar impacts are possible during an LSR mission. However, this practice can readily be abandoned in favor of disposal via targeted EDS lunar flyby into solar orbit.

With the possible exception of the CM, an LSAM ascent stage is the most complex and expensive 1.5-launch Ares payload component. Like the CM, it contains power, propulsion, communications, life support, and avionics systems. Unlike the CM, an ascent stage is not reused. It therefore represents a considerable hardware replacement cost associated with every 1.5-launch lunar mission. This cost is almost completely avoided by LSR architecture, where the only systems equivalent to those on an ascent stage are likely to be found aboard the RCM. In contrast to LSAM ascent stage systems destroyed by lunar impact, RCM systems would be intact and potentially available for planned or contingency reuse on the lunar surface for many years. If standardized and designed for interchangeability, RCM components could easily be used to repair failed hardware associated with a variety of lunar surface exploration systems.

IV. LSR Development Challenges And Operational Payoffs

The following subsections document notable aspects of LSR development and operations not relating to 1.5-launch architecture vulnerabilities. Although mature solutions to LSR development challenges may not be in hand at this early stage, practical design approaches are nevertheless evident. In many cases, these LSR-motivated solutions lead to capabilities necessary for robust exploration of the Moon and other solar system destinations.

^{††} In the special case of a near-polar LLO, periods of continuous direct communications with Earth for a significant fraction of a day can be experienced once every two weeks. Similar periods of continuous sunlight generally occur at other times once every two weeks in a near-polar LLO.

H. Standardized Ares V Payload Packaging

As a design goal, deviations in pre-launch processing procedures, Ares V payload shroud dimensions, and ground support equipment are to be minimized between RCM/DM and CEV/DM payloads. Through this standardization, economies of scale and rapid response to contingencies cited in Subsection A are further realized.

Working with a common DM component in both Ares V payloads is an asset to packaging standardization. To minimize Ares V shroud length and payload height on the lunar surface, it would be desirable to partially imbed the RCM and CEV within the DM. This configuration would offer the added benefit of protecting embedded structure from debris and radiation flux. Imbedded packaging may be achievable by evolving the LSAM descent stage's single axial main engine into a ring of smaller engines near the DM's periphery. If this evolution adds some propulsion plumbing weight and complexity, it also offers increased redundancy to a critical system.

I. Crew-Rated Ares V

Although 1.5-launch architecture only requires crew-rating Ares I, doing so for Ares V has not been precluded by US space exploration policy. One possible Ares V crew certification test might be to fly a completely automated LSR mission with the CEV unoccupied. Variants of this strategy could land sufficient cargo at a single lunar surface location, establishing "critical mass" for a lunar base there. Other less ambitious certification missions requiring but one Ares V launch might involve unoccupied CEV round trips ranging from nearby LEO to remote circumlunar space.

By achieving the crew-rated Ares V milestone, important capabilities in addition to lunar exploration are enabled. Even with a single Ares V launch, a properly equipped CEV with three crewmembers is capable of reaching and returning from the most accessible of the scientifically interesting near-Earth asteroids (NEAs) [6]. Other round-trips to service envisioned infrastructure stationed near collinear Sun/Earth libration points would also be enabled.

With Ares I currently performance-challenged to deliver crewed CEVs with curtailed capabilities to LEO, it is only a question of time before a crew-rated Ares V capability becomes imperative. Following or concurrent with this milestone, a "system of systems" effort bestowing uprated and scalable consumables capacity to a "Block II" CEV uncoupled from Ares I performance constraints will open the door toward NEA "stepping-stones" on the path to Mars and beyond.

J. Crew Lunar Landing Aboard CEV

To achieve a Moon landing with a crew aboard, the CEV will undoubtedly undergo some LSR-imposed modifications. Lunar landing piloting techniques fundamentally different from those developed during the Apollo Program will be required. However, as previously noted in Subsection C, both crewed and cargo-only lunar landings are likely to be under control of the same automated avionics systems. Unless a contingency requires manual intervention, lunar landing piloting procedures will nominally be confined almost entirely to monitoring tasks. With RCM nav aids supporting CEV lunar landing, likelihood of manual piloting intervention is further decreased.

Unavoidable performance penalties are also associated with landing and launching the CM's Earth entry/landing systems on the Moon. But these penalties are offset by performance gains and consumables budgeting stabilities already documented in Subsections B, C, D, E, and G. The expenses of developing and fabricating LSAM ascent stage systems are also avoided. By confining crew operations to a single vehicle, significant simplifications in onboard and ground-based support procedures are realized. These simplifications are accompanied by reduced time and expense associated with crew and support personnel training. Consequently, the question of whether or not to land CEV on the Moon is more a "Can we?" performance issue than a "Should we?" strategy issue. As indicated by data presented in Section V and driven by current Ares V performance projections, the answer is "Yes, we can indeed."

K. CEV/DM Lunar Landing In Proximity To RCM/DM

If time and effort to top-off CEV consumables at the lunar landing site are to be minimized, the CEV/DM must land as close to the RCM/DM as practical. The RCM/DM must therefore be protected from direct exhaust plume impingement by the CEV/DM. Lunar debris raised by the plume poses an impact hazard to the RCM/DM as well.

Plume impingement and debris impact can be mitigated with sufficient shielding applied to vulnerable RCM components. Another possible solution addressing the debris problem would be RCM/DM deployment of a dust shield on the lunar surface. Assuming sufficient CEV/DM landing precision supported by RCM nav aids, this shield's size and weight would be relatively small.

Proximity lunar landings are not an explicit requirement associated with 1.5-launch missions, but such landings are necessary to logistics supporting any lunar base, regardless of crew/cargo transport architecture. Precision landings close to existing infrastructure are also necessary if exploration missions are ever to enjoy the efficiencies of in-situ resource utilization.

L. CEV Consumables Top-Off From RCM

Topping-off consumables has become a routine activity aboard the International Space Station (ISS). The ISS Russian segment supports sophisticated storable propellant transfers from visiting vehicles much as are envisioned for LSR. Meanwhile, US segment resupply activities typically entail relatively simplistic and manual intra-vehicular consumables transfers. Although manual consumables transfer techniques may serve as a backup on the lunar surface, efficient crew timeline utilization dictates nominal RCM-to-CEV consumables transfers be automated to the greatest extent possible. Flexible and extensible leads or shuttling transports may be applicable transfer solutions.

If top-off solutions can be developed for application in weightless flight as well as on the lunar surface, multiple useful capabilities result. In a consumables-critical contingency, a single RCM/DM launch could potentially perform a CEV rescue anywhere in the Earth/Moon system. Multiple RCM or DM derivatives, placed in series with a CEV, could supply incrementally greater capability to visit NEAs and other destinations progressively more remote from Earth. As noted in Subsection I, scalable capability such as this will blaze a trail to Mars and beyond.

M. DM Cryogenic Propellant

As detailed in Section V, sufficient DM performance delivering adequate LSR payload mass to the lunar surface depends on LOX/LH₂ propellant with $I_{sp} = 450$ s. Less efficient and more storable propellant must be used aboard the LSAM descent stage in a 1.5-launch mission because lunar landing may occur up to 8 days after launch. In an LSR profile, lunar landing is only 4 days after launch. Consequently, the cryogenic propellant storage interval aboard a DM bound for the Moon is no longer than that required of EDS design in the 1.5-launch architecture ([1], chart 10).

Transits to the Moon are likely the briefest to be encountered in deep space exploration. Unless cryogenic propellant storage technology can be significantly improved, it will be necessary to develop a DM derivative using storable propellant. This evolved DM would initially provide Δv to cargo-only payloads at remote destinations such as collinear Sun/Earth libration points or NEAs. If SM propulsive redundancy is required on long-duration CEV transits, the evolved DM could fill this role as well. Should evolved DM and SM propellant be compatible and inter-connectable, a broad range of deep space propulsive contingencies would be addressable.

V. Vehicle Mass Estimates Associated With A Notional LSR Mission

Each of the following two subsections provides mass estimates for a propulsive "stage" in an LSR mission. Driven by Δv requirements and appropriate propellant assumptions pertinent to each stage, these estimates are shown to be compatible with current Ares V performance estimates.

The DM Stage applies to LSR trajectories outbound from Earth. Because DM propulsion generates impulses required to achieve LLO and a landing on the lunar surface, this stage pertains to both RCM/DM and CEV/DM Ares V payloads. The SM Stage only applies to CEV's Earth return from the lunar surface.

Neither DM nor SM mass buildups have been documented in accord with LSR architecture. To produce LSR vehicle mass estimates of interest, it is therefore necessary to scale existing baseline designs. For the DM Stage, baseline masses are obtained from the ESAS report's LSAM descent stage buildup ([3], Table 4-23, PDF p 174). Corresponding SM Stage baseline masses are provided by the ESAS report's SM buildup ([3], Table 5-2, PDF p 247). Baseline values are summarized in Table 1.

Table 1: Stage-Unique Mass Buildup Baselines

| Mass Component | DM Stage | SM Stage |
|----------------|----------|----------|
| m_{PS} | 2362 kg | 1423 kg |
| m_D | 9464 kg | 4576 kg |
| m_S | 35055 kg | 13647 kg |
| R | 0.092298 | 0.15687 |

The iterative process of scaling a stage baseline into an LSR-compatible mass buildup entails preserving R, the stage's effective ratio of propellant subsystem mass m_{PS} to initial usable propellant mass m_L . An iteration commences with application of the rocket equation to an initial guess at stage inert mass m_D , estimating initial total stage mass m_T inclusive of payload m_Y and initial usable propellant m_L mass components as follows.

$$m_T = (m_D + m_Y) e^{\Delta v/v_{EX}}$$

With $m_L = m_T - m_Y - m_D$, the R-preservation criterion in turn drives an improved m_D value for use in the next iteration if m_D has yet to converge sufficiently.

Scaling ESAS baselines into the larger stages necessary to support LSR using the R-preservation criterion is viewed to be a conservative process. For a specified Δv , rocket equation iterations will progressively increase m_T and inferred m_L with respect to their baseline values. The R-preservation criterion requires m_{PS} to increase with $R \Delta m_L$. Such is not the case in real world propellant systems. In reality, volumetric propellant mass increases are generally matched with larger hollow tanks and longer hollow feed lines, while pumps and combustion chambers may contribute little to no mass increase. Consequently, real world m_{PS} increases would be expected in proportion to $\Delta m_L^{2/3}$ or less.

N. Estimated DM Stage Mass Buildup

Constraints applicable to the DM Stage dictate payload mass m_Y be computed assuming $m_T = 62800$ kg. The m_T constraint is imposed by estimated maximum Ares V payload mass deliverable to TLI without a supporting Ares I launch ([1], chart 4). The DM's propulsive events are expected to occur over the same 4-day interval following launch to which the 1.5-launch architecture's EDS is being designed. Consequently, equivalent LOX/LH2 propulsion is assumed for the DM stage with $I_{SP} = 450$ s. Finally, $\Delta v = 3015$ m/s is assumed from summing Apollo 17 impulses found in as-flown mission records ([7], p 519) at lunar orbit insertion (910.7 m/s), primary descent orbit insertion (60.0 m/s), secondary descent orbit insertion (2.3 m/s), and lunar landing (2041.6 m/s). Table 2 contains the resulting DM Stage mass buildup.

Table 2: Scaled DM Stage Mass Buildup Assuming LOX/LH2 Propellant

| Mass Component | Mass (kg) |
|----------------|-----------|
| m_{PS} | 2869 |
| m_D | 9972 |
| m_L | 31087 |
| m_S | 41059 |
| m_Y | 21742 |
| m_T | 62801 |

O. Estimated SM Stage Mass Buildup

Constraints applicable to the SM stage dictate payload mass m_Y be set equal to the CM's total mass. An ESAS-vintage $m_Y = 9506$ kg is adopted ([3], Table 5-1, PDF p 234) to minimize changes imposed by subsequent Ares I constraints. Because LSR architecture calls for SM Stage propellant to be stored over intervals up to six months, Apollo Service Propulsion System performance is assumed. This equates to nitrogen tetroxide (N2O4) and

Aerozine 50 propellant producing $I_{sp} = 314$ s [8]. Finally, $\Delta v = 2800$ m/s is assumed from summing Apollo 17 impulses found in as-flown mission records ([7], p 519-520) at Lunar Module ascent stage orbit insertion (1851.9 m/s), ascent stage vernier adjustment (3.0 m/s), ascent stage Terminal Phase Initiation (16.4 m/s), and Trans-Earth Injection (928.5 m/s). Table 3 contains the resulting SM Stage mass buildup.

Table 3: Scaled SM Stage Mass Buildup Assuming N2O4/Aerozine 50 Propellant

| Mass Component | Mass (kg) |
|----------------|-----------|
| m_{PS} | 3735 |
| m_D | 6689 |
| m_L | 24010 |
| m_S | 30699 |
| m_Y | 9506 |
| m_T | 40205 |

Per Table 2's $m_Y = 21742$ kg, a CEV with full propellant load cannot be delivered to the lunar surface because Table 3 $m_T = 40205$ kg clearly exceeds that capacity. However, Table 3's $m_D + m_Y = 16195$ kg permits the CEV/DM payload to land on the Moon with $21742 - 16195 = 5547$ kg of SM propellant already loaded. In a contingency, this propellant could apply $\Delta v = 907$ m/s to the CEV if it became necessary to perform a DM jettison before lunar landing. This contingency impulse is nearly equal to Apollo 17's nominal Trans-Earth Injection Δv .

In a nominal mission, the CEV/DM lands on the Moon adjacent to the RCM/DM with 5547 kg of propellant already aboard its SM. Per Table 3's $m_L = 24010$ kg, $24010 - 5547 = 18463$ kg of additional top-off propellant is required from the RCM aboard the SM in order to achieve Earth return. From Table 2's $m_Y = 21742$ kg, this leaves $21742 - 18463 = 3279$ kg of RCM mass margin to cover structure, avionics, and reserve CEV consumables.

VI. Conclusion

It was not easy to modify Apollo Program lunar mission architecture from its original "direct ascent" concept to lunar orbit rendezvous, but it was a necessary refinement in meeting programmatic goals. In a similar vein, serious Constellation Program consideration should be given to LSR lunar exploration architecture at this time if the fundamental "land anywhere; leave anytime" lunar exploration requirement is to be retained.

Since the ESAS effort spawned current 1.5-launch lunar exploration architecture, the originally preferred Cargo Launch Vehicle (CaLV) with single-launch TLI payload capacity of 54600 kg ([3], Section 1.5.3.2, PDF p 52) has evolved to Ares V single-launch TLI payload capacity of 62800 kg ([1], chart 4), a 15% increase. This paper has demonstrated two Ares V launches with currently anticipated performance will achieve a notional LSR mission with payload capacity to spare. In addition, comparisons between LSR and 1.5-launch architectures indicate greater reliability, stability, life cycle cost efficiencies, and mission contingency options lie with LSR. Finally, LSR architecture leads to a scalable CEV mission capability supporting destinations beyond the Moon with minimal additional development costs. The author is prepared to further demonstrate and assess specific LSR mission design capabilities and applications on request.

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