Executive Summary: Revisited Virtues Of Lunar Surface Rendezvous (LSR)

Current lunar exploration architecture envisioned by the National Aeronautics and Space Administration's Constellation Program (CxP) entails a crew launch by the Ares I booster and a cargo-only launch by the larger Ares V booster. In developing this "1.5-launch" architecture, CxP has found it necessary to compensate for unanticipated Ares I performance deficiencies by augmenting Ares V performance.

Recently, Ares V performance specifications have increased to the extent that an alternative Lunar Surface Rendezvous (LSR) mission profile can be flown with 2 Ares V launches. Assuming Ares V can be crew-rated, an option CxP continues to retain, this LSR architecture immediately decouples all Moon-bound payload from Ares I performance limitations. Vulnerabilities to 1.5-launch mission success and safety are also addressed by LSR as follows.

1) Only one type of launch vehicle is fabricated, processed, and flown.

2) A second payload need not be launched to rendezvous, dock, and depart low Earth orbit (LEO) for the Moon with a first payload launched less than 4 days earlier.

3) Propulsive consumables uncertainties associated with rendezvous and docking in LEO and in low lunar orbit are eliminated.

4) Propulsive loads on a pressurized docking interface between two crewed spacecraft during LEO departure for the Moon are eliminated.

5) Variations in propulsive consumables budgets associated with landing anywhere on the Moon and departing from that location at any time are dramatically reduced.

6) The spacecraft required for crew return to Earth is not abandoned in low lunar orbit during lunar surface operations lasting a week or more.

7) A spacecraft with expensive life support/avionics systems and toxic residual propellant does not suffer planned lunar impact after crewed mission completion.

Some development challenges, such as consumables resupply on the lunar surface, appear unique to LSR. However, these challenges must eventually be met if a lunar base is to be maintained or lunar in-situ resources are to be utilized. Meeting other LSR development challenges leads to scalable consumables modules supporting crewed missions to destinations beyond the Moon.

Comparisons between 1.5-launch and LSR missions indicate greater reliability, stability, applicability, extensibility, life cycle cost efficiencies, and mission contingency options lie with LSR. The author proposes LSR replace the 1.5-launch lunar exploration architecture in the interest of promoting safe, robust, and cost-effective human space exploration beyond LEO. He may be contacted via adamod@earthlink.net with feedback on this proposal or with requests to further demonstrate and assess specific LSR mission design capabilities and applications.

Daniel R. Adamo December 12, 2008

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. Data from demonstrative lunar surface rendezvous mission designs are presented, further substantiating these claims.

I. Nomenclature

H	=	circular orbit height above reference body's equatorial radius
I_{SP}	=	stage-unique specific impulse
g	=	Earth surface gravity acceleration = 9.80665 m/s^2
i	=	osculating orbit inclination with respect to reference body's true equatorial plane
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 or maneuver-specific 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 uprated to compensate for Ares I performance deficiencies. This paper documents augmented Ares V capabilities now enabling the "2.0-

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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. 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 a posigrade or retrograde 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 and documentation were 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. Subsections are sequenced according to the approximate chronological order in which vulnerabilities are encountered during a nominal 1.5-launch mission.

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 two Ares V launches in support

[†] 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.

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. Current NASA ground infrastructure planning through the year 2019 [4] dictates 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. Ground infrastructure planning currently provides only one Mobile Launch Platform (MLP) compatible with Ares I and one MLP compatible with Ares V. Lack of MLP redundancy threatens even nominal launch processing, let alone contingencies. With LSR architecture in place, a single Ares V launch, ideally departing from either LC-39 pad[‡] atop a redundant MLP, can effectively respond to the rescue call.

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

Once any payload is launched into the LEO plane targeted by 1.5-launch mission design, a "countdown clock" to TLI is running. The second payload must then be launched 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 initial launch's LEO insertion plane such that the interval between this launch and TLI is about 4 days ([1], chart 10). This strategy imposes two performance-robbing consequences. First, LOX/LH2 boil off during any extended LEO loiter period 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 of biased insertion altitudes protecting 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 with the CEV in LLO is unnecessary to LSR.

[‡] Two LC-39 pads compatible with Ares V launches would entail modifying the currently planned Ares I-compatible pad to be Ares V-compatible. Alternatively, a third pad could be added to LC-39.

[§] 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.

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 unpiloted DM lunar landing with an RCM payload must be reliably automated, manual piloting during CEV/DM descent (particularly with functioning RCM navaids) 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 [5]. 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. 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. However, it may be desirable to retain CM docking capability to facilitate LSR rescue contingencies (reference Subsection A). If CM docking is retained under LSR, the associated mass penalty will be but a fraction of that required by 1.5-launch architecture. An LSR docking would not impose any loads on the CM interface beyond those from initial rescue vehicle contact and mated attitude control supporting crew transfer from the disabled CM. These loads would be far less than experienced in the Apollo Program.

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.

[††] 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.

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 [6]. 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 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, 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 left unoccupied in favor of a dedicated habitat pre-emplaced by the RCM/DM, the CEV is in a much less dynamic environment and a much more accessible location to the crew when stationed on the lunar surface, as opposed to LLO.

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

[‡‡] 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.

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.

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) [7]. 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 navaids supporting CEV lunar landing, likelihood of manual piloting intervention is further decreased.

Regarding experience with automated soft lunar landings, the robotic Surveyor Program provides relevant, if somewhat limited, data [8]. These missions were conducted from 1966 into 1968 with relatively primitive technology in today's terms. Of the 7 "blind" landings attempted with only a radar altimeter sensor, 5 were successful. Moreover, no Surveyor mission failure is known to have occurred due to a landing system malfunction. *Surveyor 2* crashed into the Moon after an unbalanced vernier thruster failure caused the spacecraft to tumble during a trans-lunar midcourse correction. Contact with *Surveyor 4* was lost and never recovered only 2.5 min before planned touchdown.

The prospect of landing in a reclining posture aboard a CEV/DM may seem awkward compared to the direct lunar surface view afforded Apollo crews during terminal descent, but this is not a spaceflight challenge unique to LSR. Current LSAM descent stage dimensions have grown to the extent that critical lunar surface viewing by the crew may no longer be possible during descent. Piloting while critically dependent on remote video views has been routinely conducted during the Space Shuttle Program. In addition to *Mir* and International Space Station (ISS) dockings and undockings, many robotic manipulation tasks involving large masses are performed with only partial dependence on direct "out the window" crew visualization.

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 navaids, 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

The criticality of CEV consumables top-off to LSR crew survival and mission success must not be underestimated, particularly in a sortie scenario without access to long-term safe haven on the Moon's surface. However, it should also be understood that consumables top-off is no more critical to an LSR mission than is rendezvous/docking in LLO to a 1.5-launch mission. Both mission milestones entail complex events, and the crew will be marooned if these events are not successfully completed.

Topping-off consumables has become a routine activity aboard 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. Similar performance demands on the LSAM descent stage have driven its use of LOX/LH2 as well. 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.

N. CEV/DM Systems Redundancy

During the Apollo 13 mission, a cryogenic oxygen tank explosion during trans-lunar coast disabled primary Command/Service Module (CSM) propulsion and crew life support systems. After aborting the mission's planned lunar landing, it was therefore necessary to utilize redundant Lunar Module (LM) systems in safely returning the crew to Earth.

When addressing similar abort scenarios with CEV/DM design, it is important to recognize a redundant Apollo 13 LM cockpit and its habitable volume were not essential to safe crew return. Stated in more negative terms, pressure loss in the primary CM cockpit/habitable volume during Apollo 13 trans-lunar coast would have doomed the crew because the only habitable volume capable of intact return to Earth's surface would have been lost. Short of flying two CMs or utilizing an external safe haven in cislunar space, no lunar exploration architecture can support crew survival after its Earth return habitable volume is lost.

Consequently, a well-designed CEV/DM can offer at least the level of systems redundancy required to survive an Apollo 13 trans-lunar abort scenario. During LSR trans-lunar coast, primary propulsion and life support systems would reside in the DM. Any critical failure of these systems would likely entail lunar landing abort, DM jettison, and crew return to Earth using redundant CEV systems nominally designed for this purpose. Adopting CEV/DM performance attributes developed in Section V, robust trans-lunar and other abort capabilities have been demonstrated ([9], Section VIII, Section IX, and Section X).

If two Ares V launch pads are supporting LSR operations (reference Subsection A), abort capabilities documented in [9] can be further supplemented. Consider a disabled CEV/DM whose only remaining functional capability is crew life support. The crew aboard this vehicle can be rescued anywhere in cislunar space with but one Ares V launch performed days after the disabled crew's TLI ([9], Section XI).

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.

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

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.

O. 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, Δv = 3015 m/s is assumed from summing Apollo 17 impulses found in as-flown mission records ([10], p 519) at lunar orbit insertion (LOI, 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

P. 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 [11]. Finally, Δv = 2800 m/s is assumed from summing Apollo 17 impulses found in as-flown mission records ([10], p 519-520) at LM 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 (TEI, 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 Δv = 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. Key Results From LSR Demonstration Mission Designs

Two LSR demonstration missions have been designed as follow-on research motivated by this paper's original draft. As documented by [9], "Mission A" targets lunar landings at Aristarchus Plateau, a region of high scientific interest. An as yet unpublished "Mission C" targets Clavius Crater landings in 2009. Clavius Crater lies within a lunar region virtually inaccessible to 1.5-launch architecture. The two LSR missions and landing sites are summarized in Table 4.

Table 4: LSR Demonstration Missions And Their Lunar Landing Site Locations

Mission	Site Name	Selenocentric Latitude	Selenocentric Longitude
A	Aristarchus Plateau	26° N	49° W
C	Clavius Crater	59° S	15° W

Adapted from Figure 18 in [12], Figure 1 contains two selenocentric maps known as temporal availability contour plots (TACPs). In contrast to a conventional topographic map, in which contours denote geodetic altitude, TACP contours denote the percentage of time 1.5-launch architecture can access a particular lunar surface location. To clarify this information, regions enclosed by TACP contours are shaded with progressively deeper hues of blue as 1.5-launch accessibility increases. Thus, locations with 100% accessibility have the deepest blue shading, while locations with < 10% accessibility are shaded white. A network of potential landing sites is plotted on each TACP as red dots, with the Mission A and Mission C sites annotated in green.

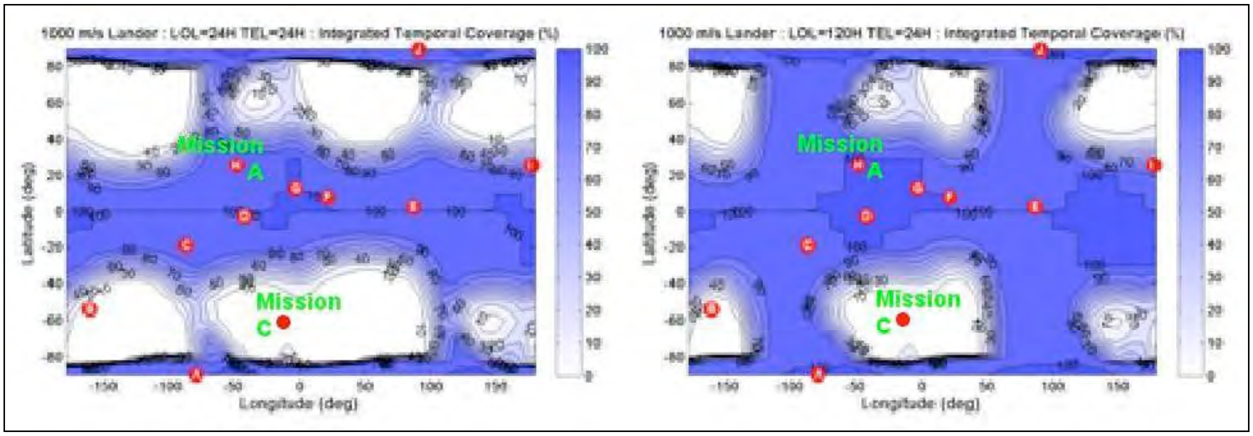

Figure 1: 1.5-Launch Temporal Availability Contour Plots (TACPs)

The two Figure 1 TACPs adopt differing lunar orbit loiter intervals to manage performance-robbing planar misalignments documented by Subsection E. In the left-hand TACP, lunar orbit loiter between initial LOI and landing is constrained to be less than 1 day, while lunar orbit loiter less than 1 day is also assumed between launch and final TEI[§§]. The right-hand TACP permits pre-landing loiter intervals up to 5 days but is otherwise equivalent to the left-hand TACP. As expected, extended loiter time shrinks regions of inaccessibility, most notably at mid-latitudes.

Regardless of loiter time assumptions applicable to Figure 1 TACPs, significant variations in landing site accessibility are evident with respect to 1.5-launch architecture. Both TACPs also indicate Mission A lands in a very accessible location, while Mission C lands in a very inaccessible location.

Table 5 quantifies consistent LSR propulsive requirements between Mission A, Mission C, and Apollo 17-derived LOI/TEI budgets adopted in Section V. These data indicate LSR missions can indeed land anywhere and leave anytime with minimal variations in required propulsive performance. If TACPs were to be produced in an LSR context, Table 5 indicates they would be of nearly constant hue.

Table 5: Nearly Constant And In-Budget Performance Among LSR Demonstration Missions

Mission	LOI Δv (m/s)[***]	TEI Δv (m/s)
A	874.668 (862.304)	916.729
C	899.913 (897.130)	890.644
Apollo 17 Budget	910.7	928.5

It should be noted the Mission A and Mission C designs do not require highly variable loiter time in lunar orbit. The half-day interval between LOI and landing adopted for these LSR designs is in anticipation of crew sleep and navigation accuracy requirements but otherwise arbitrary. The interval between CEV launch and TEI in these designs is also notional, with values between 2 and 4 hours. This sequencing is evident in Table 6 (Mission A) and Table 7 (Mission C), where major mission events are summarized for reference purposes. Coordinated Universal Time (UTC) in these tables is listed in "day-of-year/hrs:min" format. Note also the pre-landing LLO is retrograde ($i > 90°$) to facilitate Earth circumlunar free return following a trans-lunar abort akin to Apollo 13's. Although LSR permits CEV launch into retrograde LLO for nominal Earth return, a posigrade option ($i < 90°$) with nearly identical TEI Δv also exists and is selected for both Mission A and Mission C. Posigrade lunar launch is generally preferred because TEI is performed with a direct line-of-sight to Earth.

[§§]These 1.5-launch loiter intervals may each entail up to 3 impulses required to establish, reorient, and circularize or escape an eccentric lunar orbit with a 24-hour period. All nominal LSR mission designs utilize single-impulse LOI and TEI.

[***]For LSR missions, two Table 5 LOI Δv values are presented in the format "CEV/DM Δv (RCM/DM Δv)".

Table 6: LSR Demonstration Mission A Profile Summary

2019 Date	UTC	Event
June 11	162/19:39	Ares V launches RCM/DM into LEO: H = +185 km, i = 28.5°.
June 11	162/22:24	EDS achieves RCM/DM TLI: Δv = 3147.214 m/s.
June 15	166/03:21	DM achieves RCM/DM LOI: Δv = 862.304 m/s, H = +100 km, i = 109.5°.
June 15	166/16:09	RCM/DM LLO #7 landing at Aristarchus Plateau: local solar elevation = +13.8°.
July 10	191/18:22	Ares V launches CEV/DM into LEO: H = +185 km, i = 28.5°.
July 10	191/21:10	EDS achieves CEV/DM TLI: Δv = 3151.525 m/s.
July 14	195/00:09	DM achieves CEV/DM LOI: Δv = 874.668 m/s, H = +100 km, i = 102.9°.
July 14	195/14:56	CEV/DM LLO #8 landing at Aristarchus Plateau: local solar elevation = +8.6°.
July 21	202/17:04	CEV launch into LLO: H = +100 km, i = 26.1°, local solar elevation = +63.7°.
July 21	202/20:48	CEV achieves TEI: Δv = 916.729 m/s.
July 24	205/23:47	CEV entry interface: speed = 11.010 km/s.

Table 7: LSR Demonstration Mission C Profile Summary

2009 Date	UTC	Event
April 2	092/06:59	Ares V launches RCM/DM into LEO: H = +185 km, i = 28.5°.
April 2	092/08:57	EDS achieves RCM/DM TLI: Δv = 3139.272 m/s.
April 5	095/12:39	DM achieves RCM/DM LOI: Δv = 897.130 m/s, H = +100 km, i = 116.4°.
April 6	096/01:21	RCM/DM LLO #6 landing at Clavius Crater: local solar elevation = +12.2°.
May 1	121/07:57	Ares V launches CEV/DM into LEO: H = +185 km, i = 28.5°.
May 1	121/09:49	EDS achieves CEV/DM TLI: Δv = 3143.152 m/s.
May 4	124/13:32	DM achieves CEV/DM LOI: Δv = 899.913 m/s, H = +100 km, i = 115.0°.
May 5	125/02:13	CEV/DM LLO #6 landing at Clavius Crater: local solar elevation = +9.4°.
May 12	132/12:54	CEV launch into LLO: H = +100 km, i = 60.6°, local solar elevation = +27.1°.
May 12	132/16:30	CEV achieves TEI: Δv = 890.644 m/s.
May 15	135/22:37	CEV entry interface: speed = 11.007 km/s.

VII. 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 consistently achieve LSR missions to any lunar surface location 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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