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Orbital Propellant Depots Enabling Lunar Architectures Without Heavy-Lift Launch Vehicles

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Many human lunar exploration architectures, both flown and conceptual, use at least one heavy-lift launch vehicle to deliver flight hardware to low Earth orbit. There exists a technology, however, that allows these large exploration missions to be performed without the use of a heavy-lift launch vehicle: propellant transfer. This study presents a methodology to incorporate propellant transfer into conceptual design of lunar architectures through the use of a low-Earth-orbit propellant depot. This technology is then applied to a two-launch human lunar architecture without a heavy-lift launch vehicle. The results show that not only is a lunar architecture without a heavy-lift launch vehicle feasible with a propellant depot, but it can also improve the capability to deliver payload to the surface over an architecture that includes a heavy-lift launch vehicle without a propellant depot. The optimal lunar lander for this architecture is a hypergolic lander with a deck height of only 1.53 meters that performs only a portion of the descent burn, while the trans-lunar injection stage performs the other portion. The hypergolic propellant allows for a simpler, less expensive propulsion system, a volumetrically smaller lander, and enables the use of the existing hypergolic propellant transfer technology.

Nomenclature

f_{inert}	=	inert mass fraction, $m_{\text{inert}}/(m_{\text{inert}} + m_{\text{propellant}})$
Isp	=	specific impulse, s
MR	=	mass ratio, initial mass/final mass
m	=	mass, kg
ΔV	=	change in velocity, m/s

Introduction

FOR human exploration of the moon and Mars, both flown and conceptual, at least one heavy-lift launch vehicle, defined as capable of delivering >100 mt payload to low Earth orbit (LEO), is included in a typical architecture for delivery of in-space transportation elements and mission payload [1–3]. These heavy-lift launch vehicles are a major cost driver for these architectures and thus a major inhibitor of a human lunar or Mars space exploration program. Therefore, it would be desirable to develop a smaller launch vehicle that is still capable of delivering a large payload to the lunar surface. The impetus of this study is to determine if manned lunar missions are possible without a heavy-lift launch vehicle by using a propellant depot in LEO.

For typical exploration missions using a heavy-lift launch vehicle, the propellant mass can account for up to 75% of the total launched mass [4]. A near-term strategy using relatively small launch vehicles is on-orbit propellant transfer, where separate vehicles are used to deliver the propellant and flight hardware. Although other depot locations are possible, this study examines a LEO propellant depot. Through the use of an orbiting propellant depot, propellant is stored in LEO and then transferred to the flight hardware just before trans-lunar injection (TLI). Propellant can be delivered to the depot in separate launches using the most efficient launch system available. By only delivering the largest discrete payload mass with the launch vehicle, the flight hardware which is only 25% of the total LEO mass, the required size and cost of the launch vehicle is greatly reduced.

Other benefits of this strategy are that it allows scalability in the architecture, spreads the cost of operations, and allows for growth as the demand for space access increases. This demand could arise from robotic missions that require larger payloads, Earth-bound satellites that need more propellant to extend their lives, and further human missions beyond LEO. The commercial launch vehicle industry could deliver the propellant to LEO, thus stimulating growth in the commercial launch sector. Ensuring frequent commercial launches will decrease individual launch costs by amortizing the fixed cost over more launches. This increase in demand will create more competition and further drive down the cost to access LEO. As a result, this decreased cost to access LEO will create a sustainable space transportation architecture. Finally, multiple launches per year, rather than one or two, would also lead to improved launch vehicle reliability.

Background

For the Apollo program, a mission mode called Earth orbit rendezvous (EOR) was considered, which consisted of two vehicles launched from Earth that separately delivered the lunar module from the oxidizer [2]. The two would then rendezvous in LEO for the TLI burn. The benefit of this mission mode was to reduce the payload mass per vehicle as the oxidizer represented 60% of the necessary LEO mass. When compared with the lunar orbit rendezvous (LOR) mode that was eventually selected, this EOR mode had the lowest launch vehicle mass, but it also had the highest risk because it depended on back-to-back vehicle launches and the development of a propellant transfer capability [2]. During these architecture studies, NASA even considered propellant transfer on the lunar surface to drive down launch vehicle mass [4], as shown in Fig. 1. The LOR mode was eventually selected as a compromise between mass, cost, reliability, and schedule risk.

Even after propellant transfer was eliminated from the Apollo architecture, it was still being considered for future human exploration missions beyond the moon [5] due to its potential for drastic increases in the capability of a given architecture to deliver payload to the surface. As the focus of NASA shifted from exploration to LEO operations, work on propellant transfer diminished. With NASA shifting its focus back toward exploration, propellant transfer returns to the discussion, adding new possibilities for sending humans beyond LEO. It is an essential strategy that can

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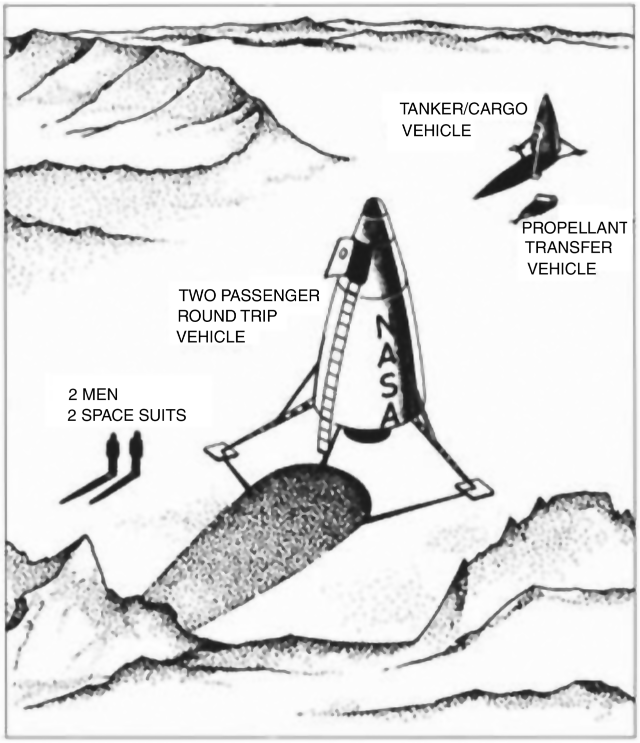

Fig. 1 Propellant transfer concept on the lunar surface [25].

improve missions as grand as sending humans to the moon or Mars [4] to missions as routine as maintaining a Global Positioning System constellation [6]. As recently as 2004, NASA engineers studied the use of a propellant depot in LEO that could provide propellant to support both near-Earth operations and exploration missions [7]. The commercial sector is also developing this strategy by exploring propellant depot and delivery system concepts [8,9].

Implementing this strategy will require some technology development in the near future; however, hypergolic propellant transfer has been performed by the Russian Space Agency to refuel every space station since Salyut-6 in 1978 using the Progress vehicle [10]. Progress transfers N_2O_4/UDMH propellant using gaseous nitrogen pressurant and a bladder that separates the propellant from the pressurant gas to push propellant from one tank to the other. NASA demonstrated its ability to transfer propellant with the Orbital Refueling System (ORS), flown on STS-41G [11], and with Orbital Express, launched in March 2007 [12]. Work has also been done, primarily by Moog, Inc., working with NASA, to develop the interface between the two vehicles with the Universal Refueling Interface System (URIS). The URIS contains the docking mechanism, fluid transfer coupling, and electrical connectors [13]. The technology to transfer propellant on orbit is mature for hypergolic propellant, but this propellant is not used extensively in many exploration architectures mainly because of its relatively low Isp.

Alternatively, cryogenic propellants have a much higher Isp than hypergolic propellants typically at the expense of propellant bulk density. Though, the transfer of cryogenic propellants poses more problems than that of hypergolic propellant. The simple pressure-fed transfer system with bladders to separate pressurant gas and propellant used for hypergolic propellants appears impractical, since no bladder material exists that can operate practically at cryogenic temperatures [14,15]. Pump-fed cryogenic propellant transfer is performed frequently on Earth in a 1-g environment. Boiloff gasses are easily vented out at the top of the tank during the transfer. However, transferring propellants in a zero-g environment makes venting of the boiloff propellant difficult because the propellant location in the tank is not predictable [16]. One solution to this problem is to use linear or angular acceleration to settle the propellant to one end of the tank or to the outer diameter of the tank, respectively [14]. Another simple method for transferring propellant is tank replacement using quick disconnect couplings. However, this method is only practical for transferring small to moderate amounts of fluids, such as life support fluids or small quantities of propellants [14]. A system that must transfer large amounts of cryogenic propellant quickly in a microgravity environment will likely require a combination of acceleration, a pressurization system, and possibly pumps to effectively settle and then transfer the propellant.

Assuming the development of long-term cryogenic propellant storage and transfer technologies, this paper investigates the utilization of an on-orbit propellant depot in a lunar architecture that can eliminate the requirements for a heavy-lift launch vehicle.

Lunar Architecture Concept with LEO Propellant Transfer

There are several options for a lunar architecture with propellant transfer, all of which follow the same general format. There are two modes within a lunar architecture: crewed missions, and cargo missions. The flight hardware and/or crew are launched on a given launch vehicle(s), while the propellant is launched on separate launch vehicles. In LEO, the propellant is transferred from the propellant delivery vehicle, or tanker, to a propellant depot already in orbit. Once the depot has a sufficient amount of propellant, it is transferred to the propulsive stage that performs the TLI burn and possibly to the lunar descent module (LDM) that performs the descent burn. This propulsive stage is typically an upper stage (US) for the launch vehicle delivering the flight hardware. The US performs a lunar orbit insertion (LOI) burn to place the vehicles in low lunar orbit (LLO) around the moon. This stage could also perform a portion of the descent burn if enough propellant remains after the first two burns. The lander must then complete the descent to the lunar surface using the LDM. In the crewed mission mode, the lunar ascent module (LAM) delivers the crew to the return vehicle in LLO, in which they return to the Earth. In the cargo mission mode the mission ends here, where the cargo is waiting on the surface for later use.

Methodology

Architecture-Level Optimization Problem

Determining the best architecture can be thought of as a classical optimization problem with the objective function to maximize the payload delivered to the surface in both the crewed and cargo missions combined. For this study, the design variables considered are the propellant type used in the LDM and the percentage of the descent burn performed by the US in addition to the TLI and LOI burns.

The interesting aspect of this problem is the constraints imposed on each vehicle. First, the launch vehicle that delivers the flight hardware to LEO will have a maximum payload capability, and the dry mass of the LDM, the LAM for the crewed mission, and cargo must fit within this limit. Second, the propellant capacity of the US is fixed for a given launch vehicle. This will determine how much of the descent the US can perform after it performs TLI and LOI. Finally, the launch vehicle will also have a volumetric constraint due to the payload shroud. Because it is desirable to minimize the lander deck height for easy egress and off load, this constraint tends to be an equality constraint so that the deck height is minimized for a fixed volume LDM, which is determined by the required amount of descent propellant.

Scaling the Descent Stage

As the design space is explored, the only vehicle in the architecture that changes is the LDM. As the US performs more or less of the descent burn, the ΔV requirement (i.e., the required MR) for the LDM decreases or increases, respectively. Also, by changing the propellant type in the LDM, the bulk density of the propellant changes, resulting in a volumetrically larger or smaller lander.

A common LDM is sized for both the cargo and crewed missions. The surface payload used to size the LDM in the rocket equation is the greater of these two missions' requirements. The payload capability of the launch vehicle is equal in both missions, typically making the crewed mission the sizing case because it contains the

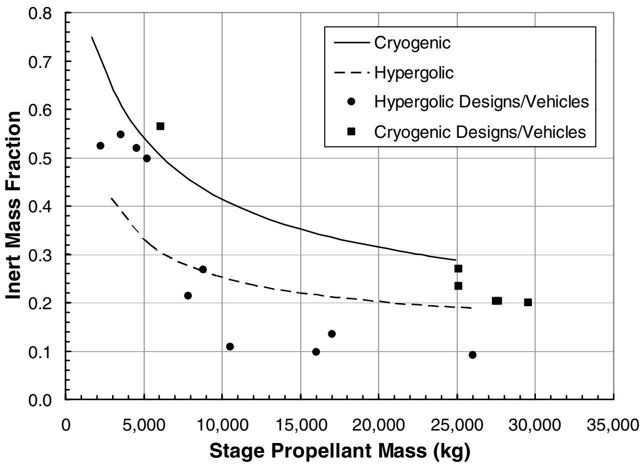

Fig. 2 Inert mass fraction vs propellant mass trends for cryogenic and hypergolic descent stages compared with existing vehicles [19].

crew in addition to the payload delivered to LEO on the launch vehicle.

The produce these scaled landers, the LDM is scaled photographically from the baseline landers (requires both a baseline hypergolic and cryogenic lander) by changing the propellant mass that the vehicle carries. Photographic scaling involves stretching a baseline vehicle while maintaining all characteristics, such as layout, tank pressure, and engine performance, of the baseline vehicle. This scales the vehicle up or down based on the ratio of propellant mass in the scaled vehicle over the propellant mass in the baseline vehicle. The inert mass of the vehicle then changes based on this required propellant mass ratio, which is equivalent to the volume ratio. Tank mass, for instance, scales with the ratio of volumes between the baseline and scaled lander because, for a given tank pressure and material, the mass of the tank is proportional to the volume [17]. Primary structure, secondary structure, and thermal protection, however, scale with the surface area ratio [18] between the two vehicles. Other parameters that control the scaling of the vehicle are landed mass ratio, which scales the landing gear mass, and gross mass ratio, which scales the engine mass, where constant engine and system thrust-to-weight ratios are assumed.

Changing the ΔV requirement affects the propellant mass, which is inversely proportional to f_{inert}, as shown in Fig. 2 for both existing designs and the scaling model used in this analysis. This trend occurs because area ratio is less than the volume ratio (which is based on propellant mass). Because several systems in the vehicle scale with area ratio, this causes diminishing returns as propellant mass increases. Results from the model used in this analysis are shown in Fig. 2 for both cryogenic and hypergolic landers. Assuming a constant inert mass fraction is less accurate than the model presented above, and the trends from this model correspond to trends that are present in existing vehicles [19].

Application to a Lunar Architecture Without a Heavy-Lift Launch Vehicle

Concept of Operations

There are several potential transportation architecture concepts that could be augmented with propellant transfer. For the purposes of this study, analysis is limited to a two-launch crewed lunar mission using the Ares I launch vehicle, as shown in Fig. 3, and one-launch cargo lunar mission using the Ares I launch vehicle, as shown in Fig. 4. These architectures, which use a single launch vehicle for all flight hardware delivery, should have better reliability than an architecture that uses multiple launch vehicle types (i.e., a heavy-lift vehicle for cargo and a separate crew launch vehicle). This is due to the reduced number of engines.

In the two-launch crewed mission, shown in Fig. 3, the propellant depot is initially filled with propellant by commercial launch vehicles. Then, the LDM and LAM are launched on a single Ares I into LEO. The LDM does not have propellant in its tanks during launch, but the LAM is full of propellant. After the lunar lander and US dock in LEO, the propellant depot transfers its propellant to the Ares I US and the LDM. The crew is then launched in the crew exploration vehicle (CEV) on a second Ares I. The two vehicles

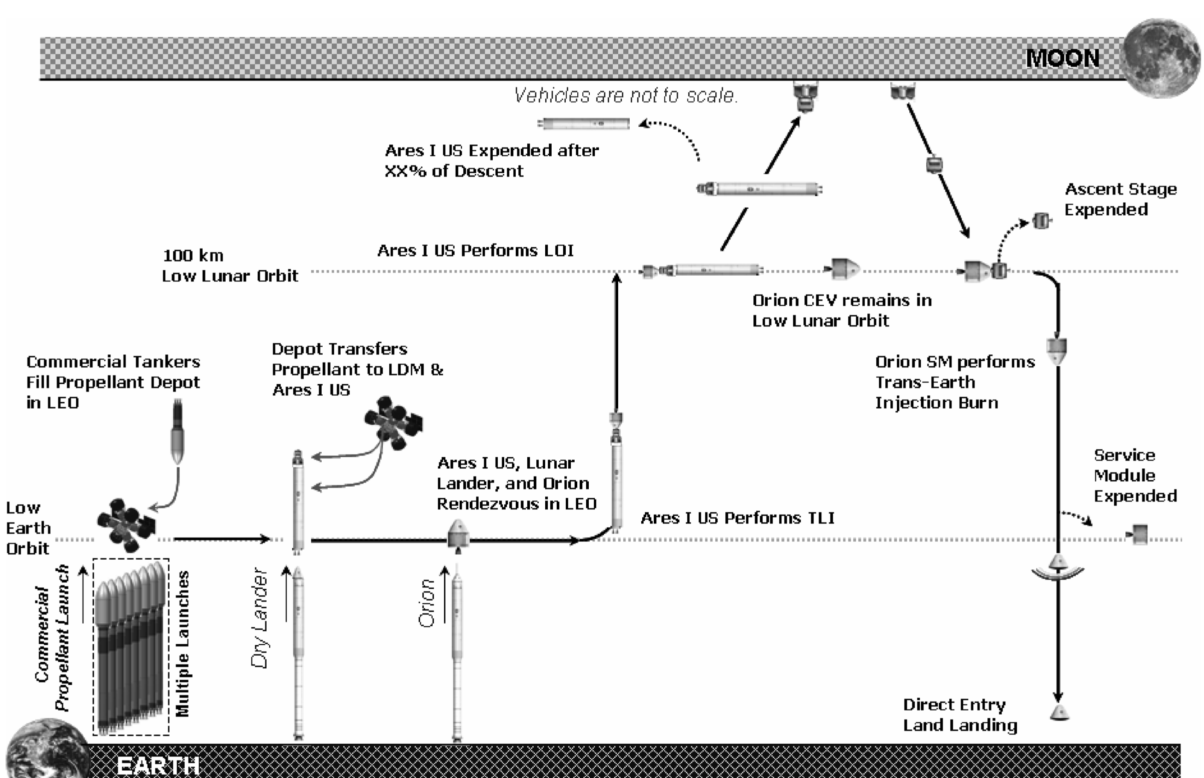

Fig. 3 Crewed lunar concept of operations.

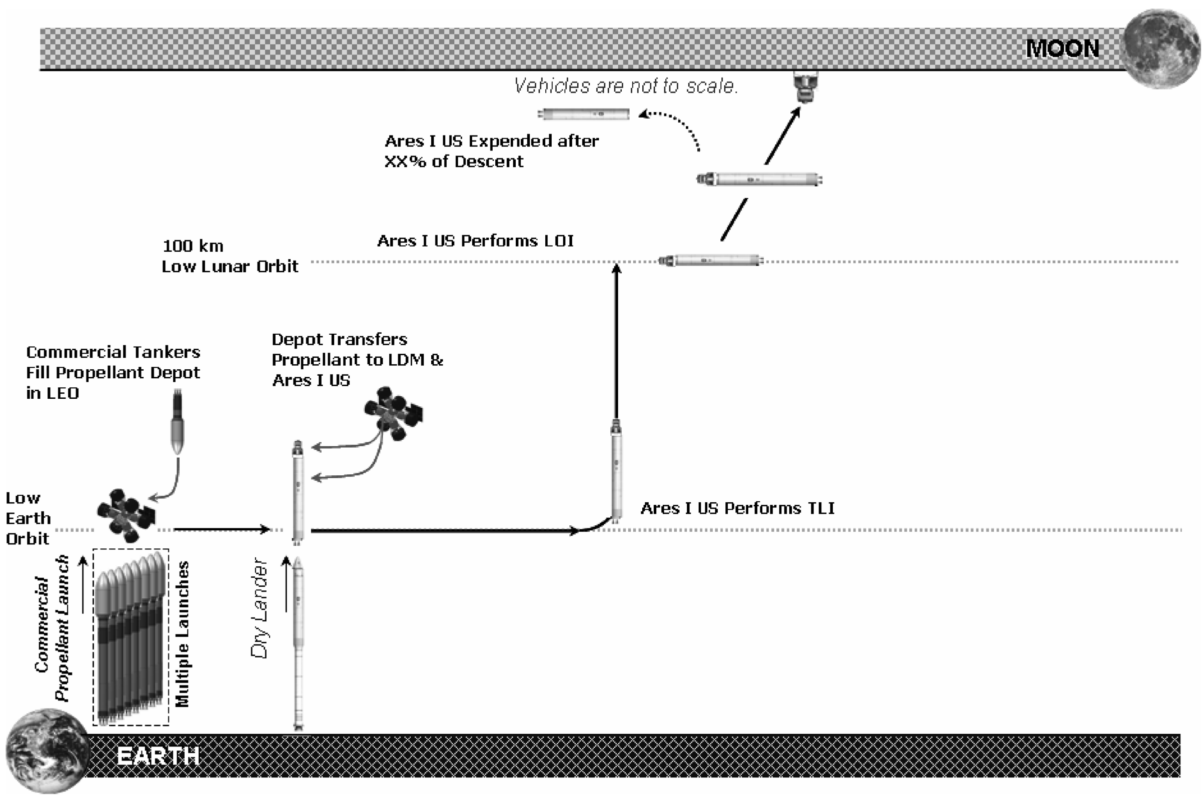

Fig. 4 Cargo lunar concept of operations.

rendezvous in LEO and the Ares I US performs the TLI burn. In this study, the required ΔV for this burn is $3,175$ m/s [20]. After this burn the propellant in the Ares I US is not fully exhausted so it is used again to perform the LOI burn with a required ΔV of $1,000$ m/s [20]. The crew then transfers from the CEV to the lunar lander, and the two vehicles separate. The CEV remains in LLO throughout the surface operations. Since there is still propellant remaining in the Ares I US after both TLI and LOI, it will exhaust this remaining propellant to perform a portion of the descent burn, which has a total required ΔV of $2,030$ m/s [20]. The percent of the descent burn performed by the Ares I US is a variable optimized in this study. The remainder of the descent burn is performed by the LDM. After the surface operations are complete, the LAM ascends and rendezvous with the CEV in LLO. Once the crew has transferred back into the CEV, the LAM is discarded in LLO or it deorbits to the lunar surface. The CEV service module (SM) performs the trans-Earth injection (TEI) burn and the crew enters the Earth's atmosphere in the CEV crew module (CM).

In the one-launch cargo mission shown in Fig. 4, the propellant depot is again filled with propellant using the tanker vehicles. Then, the dry LDM and cargo are launched on a single Ares I. While in LEO, the propellant depot transfers its propellant to the empty Ares I US and LDM. Just as with the crewed mission, the Ares I US performs the TLI, LOI, and a portion of the descent burn. The LDM performs the remainder of the descent burn, and the cargo is delivered to the surface without the need to return.

Baseline Vehicles

The vehicles used in these two concepts of operations are shown in Fig. 5. A common LDM is sized to deliver the payload to the surface required by both missions. This common LDM is a scaled version of the baseline LDM shown in Fig. 5. The Ares I US, CEV, and LAM do not change in this analysis from their baseline configurations.

Table 1 contains the mass properties for each of the vehicles used in this study, which have been modified to account for the propellant transfer hardware mass. The inert and gross masses are used to determine the propellant capacity and mass ratio of each vehicle. Although the propellant used by the Ares I US and baseline LDM is LOX/LH_2, the use of N_2O_4/UDMH in the LDM is also explored in this study.

Architecture-Level Optimization Problem

The optimization problem for this architecture maximizes the objective function which is payload delivered to the surface in both the crewed and cargo missions combined. The design variables available to change are the propellant type used in the LDM and the percentage of the descent burn that is performed by Ares I US.

Fig. 5 Ares I US & CEV exploded view (left), and Altair LDM & LAM (right). Images: NASA.

Table 1 Properties of selected lunar exploration vehicles

	Ares I US	CEV crew module	CEV service module	Altair descent stage	Altair ascent stage
Inert mass, mt	18.11	7.43	4.53	8.83	3.17
Gross mass, mt	157.41	7.62	13.83	35.01	6.24
On-orbit refueling?	Yes	No	No	Yes	No
Fixed or scaled	Fixed	Fixed	Fixed	Scaled	Fixed

The constraints imposed on this problem are derived directly from the vehicle characteristics. The Ares I is capable of placing 20.2 mt into LEO [1]. Therefore, the dry mass of the LDM, the LAM (for the crewed mission), and cargo must fit within this limit. This constraint is active through much of the design space. The propellant capacity of the Ares I US is fixed at 139.3 mt, as calculated from Table 1. As the Ares I US has a large propellant capacity, it typically has propellant left over after the LOI burn. Therefore, this constraint is inactive throughout much of the design space until the Ares I US is used to perform a large portion of the descent burn. The crossover point where this constraint becomes active is discussed in a later section and is a function of the LDM propellant selection. The assumed maximum payload/core diameter ratio is 1.55, which is derived from the core/payload diameter ratio of the existing Titan IV-B [21]. For a lander launched on the Ares I, which has a core diameter of five meters, the stowed diameter must be less than or equal to 7.75 m. Because it is desirable to minimize the lander deck height for easy egress and off load, this constraint tends to be an equality constraint so that the deck height is minimized for a fixed volume LDM.

Propellant-to-Orbit Options

Tanker Vehicles

The Space Propulsion Sizing Program (SPSP) [22] developed at NASA Langley Research Center was used to determine the amount of propellant that could be delivered to LEO using modified evolved expendable launch vehicles (EELVs). The two vehicles examined were the Delta IV-Heavy and the Atlas V-552 [21].

The upper stages of these EELVs were stretched so that they can deliver propellant to orbit instead of payload. The payload and payload adapter were removed and the propellant tanks were extended to deliver the maximum amount of payload to orbit within the tanks of the upper stage. The stage diameter and height were constrained to the standard upper stage and payload shroud combined. This reduces any integration or dynamic issues this modification may cause.

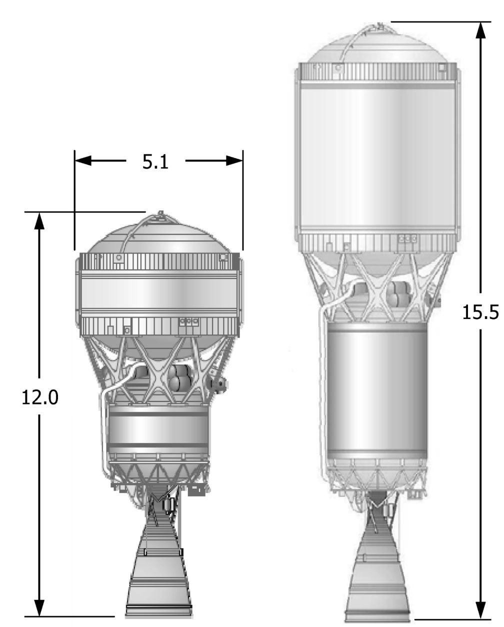

Fig. 6 Delta IV-heavy upper stage (left), and tanker variant (right). All units are in meters [23].

The Delta IV-Heavy tanker, shown in Fig. 6 [23], can deliver 21.0 mt of propellant to LEO. The tanker diameter is equal to that of the standard Delta IV upper stage, but is 3.5 m longer. The Atlas V-552 tanker is capable of delivering 11.2 mt of propellant to LEO.

Ground Operations Capability

With a capability for the Delta IV-Heavy of 21.0 mt of propellant in LEO, eight resupply launches are required to completely refill the Ares I US and the LDM, for which the propellant capacities are presented in Table 1. This includes transfer inefficiency due to boiloff, residuals, and other losses. The capability of this architecture is not sensitive to this inefficiency due to additional propellant left over after the eighth flight. For long duration LEO storage until the Ares I US is ready, zero-boiloff technology may be necessary to mitigate losses due to boiloff in the depot. The Ares I US can be used as is without any extra boiloff mitigation hardware because it will topped off shortly before TLI, and the boiloff from the five day translunar cruise is relatively small. In this sense, a propellant depot is nice at mitigating any risk due to boiloff in an exploration architecture.

The current launch frequency capability of an EELV is approximately one launch per month [23]. This is an unacceptable launch frequency if there are to be two lunar missions per year. Therefore, a launch frequency on the order of 1 per week is required to reduce the LEO depot fill time to eight weeks. This allows for delays in the tanker launches without affecting the lunar architecture schedule.

The current Delta IV launch facility occupies the launch pad during integration. Therefore, only one Delta IV-Heavy could be stacked and launched at a time [23]. Unless more launch facilities were constructed, this operational concept cannot support the volume of launches that a propellant resupply architecture would demand.

The Atlas V launch facility uses a mobile platform that carries the launch vehicle from a vehicle integration facility (VIF) to a stationary launch pad approximately 24 hours before launch [24]. With the construction of more VIFs around a central launch pad, several EELVs could be stacked simultaneously and launched in succession. This operational concept could support weekly launches with the construction of a sufficient number of VIFs.

Results

The most valuable aspect of augmenting a lunar architecture with a propellant depot is the increase in payload to the surface that it realizes. This section discusses the optimal configurations throughout the design space. Again, the design variables used are the percent of the descent burn that is performed by the Ares I US and the propellant type in the LDM (cryogenic or hypergolic).

Cryogenic Lander

Figure 7 contains the performance data (crewed and cargo payload delivered to the surface and the lander deck height) for this architecture using a cryogenic lander, allowing the Ares I US to perform varying amounts of the descent burn. The horizontal axis is the percentage of the descent burn that the Ares I US performs, the left vertical axis is the payload delivered to the surface, and the right vertical axis is the deck height of the lander.

As the Ares I US performs more of the descent burn, the LDM requires less propellant and is, therefore, smaller, so more payload can fit on the Ares I at launch. After the completion of the TLI, LOI, and the given portion of the descent burns, the Ares I US still has propellant capacity until it has performed 37% of the descent burn. The Ares I launch capability is the active constraint, and the smaller LDM is directly proportional to the larger payload capability. Once the Ares I US reaches 37% of the descent, it has exactly exhausted all of the propellant in its tanks for the crewed mission. This is the crossover point where below this percentage there is propellant left in the Ares I US, and above this percentage the Ares I US capacity is the active constraint. To complete a larger portion of the descent burn

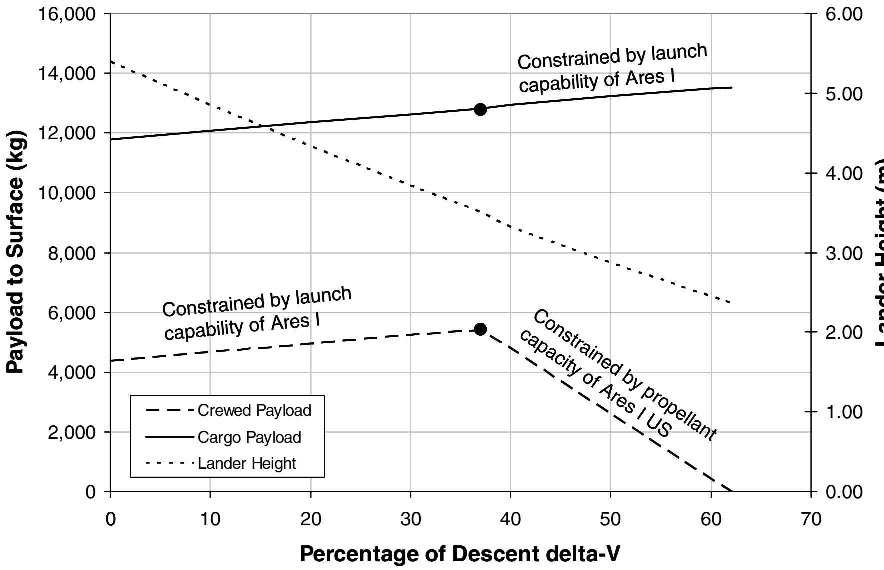

Fig. 7 Architecture performance using a cryogenic lander.

using the Ares I US, the crewed payload capability must decrease after 37% until it eventually reaches zero payload capability at approximately 62%.

Because the cargo mission does not have the CEV attached during TLI and LOI, the Ares I US does not use as much propellant to achieve the same mass ratio as it uses during the crewed mission. Therefore, it still has propellant capacity remaining throughout the design space for the cargo mission, and the Ares I launch capability constraint is always active.

The optimal point for the cryogenic lander with respect to combined crewed and cargo payload capability is the crossover point at 37% of the descent. Although the cargo capability continues to increase beyond this point, the crewed capability decreases more rapidly. At this optimal point the crewed mission can deliver 5.2 mt of payload (along with the LAM, airlock, and crew) to the surface, and the cargo mission can deliver 12.8 mt of payload to the surface.

It is also important to note that as the Ares I US performs more of the descent burn, the LDM does not have to carry as much propellant. Therefore, the lander deck height becomes lower as the volume required decreases. It is valuable to decrease the deck height as much as possible to reduce complications with crew egress and cargo off load. The optimal cryogenic lander has a deck height of 3.51 m.

Hypergolic Lander

Figure 8 contains the performance data for this architecture using a hypergolic lander. The trends for this propellant option are similar to those of the cryogenic lander. The Ares I US has excess propellant capacity until it has performed 26% of the descent burn, meaning that the active constraint is the launch capability of the Ares I. Above 26% the propellant capacity of the Ares I US is the active constraint and the payload capability decreases as previously explained. Again, for the cargo mission, the Ares I US does not use all of its propellant capacity, and the launch capability of the Ares I is the active constraint.

Compared with the cryogenic lander, the hypergolic lander configuration delivers more payload to the surface while also having a smaller deck height due to the high propellant bulk density of the hypergolic propellant as compared with that of the cryogenic propellant. The dry mass of the hypergolic lander is less than the dry mass of the cryogenic lander because the tanks and structure are smaller. Because the lander is launched without propellant, and the 20.2 mt launch constraint is the active constraint until the Ares I US performs 26% of the descent, more payload can be placed into LEO.

Unfortunately, once in orbit the hypergolic lander requires more propellant because its Isp is significantly lower than the cryogenic

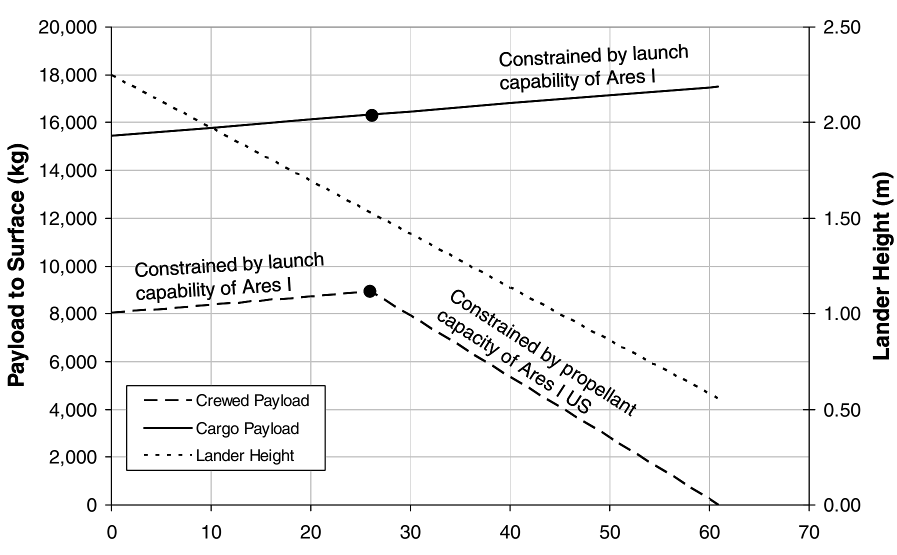

Fig. 8 Architecture performance using a hypergolic lander.

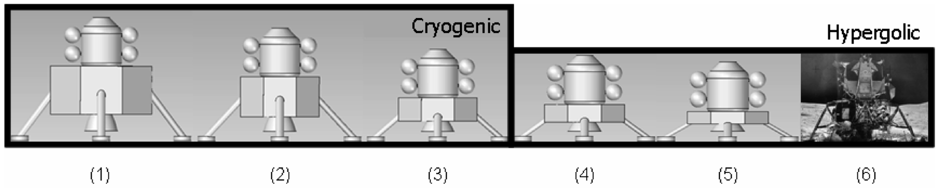

Fig. 9 Possible lander solutions.

lander. The Ares I US must burn more propellant once in LEO because its payload mass is much higher. This phenomenon is the reason why the crossover point for the hypergolic lander is more than 10% less than the cryogenic lander crossover point. The optimal hypergolic point of 26% of the descent can deliver 8.9 mt of payload to the surface in the crewed mission and 16.3 mt of payload to the surface in the cargo mission, both at a height of 1.53 m.

Solutions of Interest

Four solutions from above, along with the baseline Altair lander and the Apollo Lunar Excursion Module (LEM) [18], are selected to compare more closely. These six landers, as shown in Fig. 9 and detailed in Table 2, are, from left to right, 1) current Altair lander (cryogenic) which performs LOI and descent in an architecture that contains a heavy-lift launch vehicle but no propellant depot, 2) cryogenic lander which performs 100% of descent, 3) optimal cryogenic lander which performs 63% of descent, 4) hypergolic lander which performs 100% of descent, 5) optimal hypergolic lander which performs 74% of descent, and 6) Apollo LEM (hypergolic) which performs 100% of descent.

The solution in which the Ares I US does not perform any of the descent is reported for several important reasons. First, these cases show the performance benefit of using the remaining Ares I US propellant for descent. Second, these cases allow a comparison between cryogenic and hypergolic landers without the confounding effect of allowing the Ares I US to perform a portion of the descent. Programmatically, it may be desirable to leave the empty Ares I US in LLO to accommodate the development of a propellant depot in LLO, which will be discussed briefly in a later section. The argument could also be made that performing descent with the Ares I US is too risky if there are predeployed assets on the lunar surface. The impact point of this drop stage could be difficult to keep away from those assets in all cases, especially if an abort is necessary.

The cryogenic landers are volumetrically larger than the hypergolic landers, but cannot deliver more payload to the surface. Although the performance (Isp) is better in the cryogenic landers, the propellant bulk density is much higher in the hypergolic lander, creating volumetrically smaller hypergolic landers that must hold more propellant. Because the Ares I launch capability constraint fixes the dry mass in LEO, the smaller dry landers can be launched with more payload. This effect can be clearly observed using Table 2 by comparing the data on landers (2) and (4) without the confounding of allowing the Ares I US to perform a portion of the descent burn.

The optimal lander configuration (that delivers the most payload to the surface) is lander (5): the hypergolic lander where the Ares I US performs 26% of the descent and the lander performs the remaining 74%. This solution increases payload to the surface by a factor of 3.43 over the baseline Altair lander (1). This payload is also only 1.53 m above the surface, which is the lowest height of all landers considered, including the Apollo LEM. This lander is much simpler than the cryogenic landers, adopting a propulsion system that is more akin to the Apollo LEM propulsion system. Also, hypergolic propellant transfer is a proven technology that will need to be scaled to an architecture of this size.

Table 2 Possible lander solutions' details

Lunar descent module	(1)	(2)	(3)	(4)	(5)	(6)
Inert mass, mt	10.11	8.43	7.38	4.76	3.88	2.86
Gross mass, mt	35.01	20.91	14.64	22.81	16.13	11.64
Surface payload, mt	2.60	4.38	5.42	8.04	8.93	0.50
Deck height, m	6.33	5.40	3.51	2.25	1.53	3.23
Diameter, m	9.96	7.75	7.75	7.75	7.75	4.51
% Descent by LDM	100	100	63	100	74	100

Conclusions

Not only is a lunar architecture without a heavy-lift launch vehicle feasible with a propellant depot, but it can also improve the capability to deliver payload to the surface over an architecture that contains a heavy-lift launch vehicle without propellant transfer. Two smaller launch vehicles, when combined with a propellant depot, can more than double the payload to the lunar surface. The optimal lander that results from allowing the Ares I US to perform the TLI, LOI, and a portion of the descent is a hypergolic lander with a deck height of 1.53 m. The hypergolic propellant allows for a simpler, less expensive propulsion system, a volumetrically smaller lander, and enables the use of the proven hypergolic propellant transfer technology.

Using propellant transfer in a lunar architecture stimulates growth in the commercial launch vehicle industry because that sector would be supplying the propellant to LEO. Propellant delivery to LEO could drastically decrease the cost per flight for the commercial sector by demanding approximately 16 flights per year. This demand enables other companies to enter the market, leading to a more competitive launch industry, reducing launch costs even more, and, therefore, creating a sustainable space transportation system.

Launching large human payloads to LEO en route to Mars would be nearly impossible without propellant transfer. Several relatively small launches of the dry vehicles which would then be refueled in LEO is a much more feasible and sustainable solution. If humans wish to extend their presence beyond the Earth's gravitational influence, propellant transfer is a necessary technology. Robotic missions too would be able to use propellant from a propellant depot to escape Earth instead of using the capability of the upper stage of their launch vehicle.

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