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In-Space Operations: Developing a Path to Affordable, Evolutionary Space Exploration

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This paper explores alternative options for future human spaceflight programs which meet the goals of the Vision for Space Exploration and the Augustine Commission, within achievable cost and time limits. Based on four decades of NASA experience in on-orbit operations, an architecture is developed that allows access throughout cislunar space, including the surface of the Moon, through the use of multiple docking propulsion modules to achieve the necessary mission ΔV. The basic axiom of this study is that all Earth launch *must* be accomplished solely with existing launch vehicles, to limit early budget requirements and minimize time between the start of the program and initial flight missions. Analysis of prior spacecraft is used to estimate the feasibility of a human spacecraft with a marginally smaller mass than the Apollo command module, which is capable of launch via a direct translunar injection on a human-rated Delta IV Heavy. An optimized standard propulsion module, the Orbital Maneuvering Stage, is designed for lunar descent and ascent, as well as multiple applications to in-space maneuvering. A modified version of this vehicle, the Terminal Landing Stage, includes landing gear and required avionics for the actual lunar landing. Along with dedicated costs to human-rate the Delta IV Heavy, overall cost analysis was performed to find total program costs, and allow a year-by-year budget plan which keeps the peak-year expense on this program below $2.5B. The final program developed provides semiannual rotation of six crew on the International Space Station, two annual human lunar exploration missions, and "Flexible Path" missions as suggested by the Augustine Commission every 24 months. This architecture will support 6-7 human spaceflight missions per year, with a requirement for 16-17 launches of the Delta IV Heavy or its human-rated variant per year.

Acronyms

CM	Crew Module
CONOPS	Concept of Operations
DIVH	Delta IV Heavy (launch vehicle)
DIVH(H)	Delta IV Heavy (Human Rated)
ESAS	Exploration Systems Architecture Study
LEO	Low Earth Orbit
LLO	Low Lunar Orbit
LOI	Lunar Orbit Insertion
LTO	Lunar Transfer Orbit
NASA	National Aeronautics and Space Administration
OMS	Orbital Maneuvering Stage
TDS	Terminal Descent Stage

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I. Introduction

With the announcement of the Vision for Space Exploration in 2004, and the establishment of an architecture for the Constellation program throughout the latter half of that decade, the shape of future lunar exploration seemed to come into focus. Like the Apollo program, the basic paradigm was the creation of new launch vehicles to allow "all-up" missions to the moon via a single heavy-lift launch. This was modified from the Apollo mold only though the development of yet another new vehicle for crew launch, instantiating a philosophy of separating human and cargo launches to the extent possible. New human-carrying spacecraft for Earth launch and entry and lunar landing and ascent were also to be developed. Given the extensive amount of infrastructure which had to be developed, the gap in human spaceflight following the termination of the shuttle program grew as Constellation cost estimates increased and available funding remained flat or decreased. Faced with continual increases in cost estimates and a "human spaceflight gap" approaching a decade in length, events led to the Augustine commission, the administration's decision to cancel Constellation, and the current (as of the time of writing) political battle over the future of the nation's human spaceflight program.

Current and recent events dictate a simple question: *is there an alternative to the Apollo/Constellation paradigm of creating a totally new architecture prior to pursuing human exploration beyond low Earth orbit?* In the same manner in which "minimum functionality" is used as an initial design approach to aerospace systems, this paper seeks to explore a "minimum cost" approach to human space exploration. The origin of Constellation was billed as a "go as you pay" program, but a number of decisions led to an extremely high initial cost for the program, thereby extending the time between the retirement of the shuttle program and start of Constellation flights. Instead, this paper seeks to examine "bootstrap" solutions wherein smaller, more affordable systems will allow early and continual flight opportunities, while an on-going cost-constrained development program adds functionality into the system to evolve to greater and greater capabilities with time. The goal is to create a system which returns humans to the Moon in a supportable and sustainable long-term program, while enabling the "Flexible Path" of the Augustine commission to expand our horizons on what constitutes human space exploration.[1]

II. Developing a Minimal Space Exploration Architecture

The goal of this paper is to explore the space exploration architecture trade space, and to develop at least a single feasible alternative design that meets the objectives of this study. These objectives include

- Create a notional architecture which will...
  - safely return humans to the moon
  - at the earliest feasible date
  - in a program which is economically viable for extended and extensive exploration
  - achievable within current NASA budget projections
  - without jeopardizing ongoing NASA initiatives in space and Earth science, aeronautics, and technology development
- Ensure that the architecture will also support the in-space goals of the Augustine commission's "Flexible Path" for exploration in cislunar and nearby solar space
- Develop a program schedule which will feature early and continual flight opportunities throughout the development and operational phases

Clearly, it is ambitious to undertake the same endeavor as the NASA Exploration Systems Architecture Study (ESAS) with an even more restrictive set of constraints, especially considering the difference in the number of personnel available to the respective studies. No claim is made that this paper represents a rigorously optimized system, or even that this is the best possible approach within the cited objectives. Rather, the goal for this paper is to create an "existence proof" of a feasible system architecture which does not violate the constraints, and which would provide a economically acceptable, technically achievable approach to human space exploration in the next decade and beyond.

A. ΔV Requirements

Vehicle design calculations are predicated on the ΔV requirements between system nodes in space. The values used in this study for transportation to and from the moon are summarized in Table 1. Values were calculated using patched conics, and landing/ascent ΔVs were modified to reflect actual values from Apollo lunar missions. Notice that the values for lunar descent and ascent differ due to the requirement for hovering and aimpoint variation to allow terminal selection of the landing location.

Table 1. ΔV Requirements for Translunar Missions (all values in m/sec)

To→ ↓From	Low Earth Orbit	Lunar Transfer Orbit	Low Lunar Orbit	Lunar Descent Orbit	Lunar Landing
Low Earth orbit		3107			
Lunar Transfer Orbit	3107		837		3140
Low Lunar Orbit		837		22	
Lunar Descent Orbit			22		2684
Lunar Landing		2890		2312	

B. Launch Vehicles

A review of the history of Apollo funding shows that launch vehicles are the single largest cost element, representing 43% of the entire Apollo budget. During the formative years of the Apollo program (1964-1968), more than half of all Apollo funding went to launch vehicle development.[2] Apollo represented a historical anomaly, in that for much of this time budget was of secondary importance as compared to achieving the program goal of reaching the moon before the end of the decade. In the presence of real and significant cost constraints, the single most effective place to cut back on costs are by minimizing the development of new launch vehicles. This also reduces time to flight, as the launch vehicle development frequently represents the pacing item in completing the flight system.

The most fundamental assumption of this study is that *only existing launch vehicles may be considered.* This will constrain overall program options and require more extensive use of alternative technologies, such as on-orbit operations (discussed below).

The largest existing U.S. launch vehicle is the Delta IV Heavy (DIVH), which can transport 22,977 kg of payload (including the mass of the payload attach fixture) to low Earth orbit (LEO), or 10,403 kg into a lunar transfer orbit (LTO, assumed C_3=0). Subtracting 419 kg for a 1666-5 payload attach fixture produces the actual payload capacities of 22,558 kg to LEO and 9984 kg into LTO.[3] Other competing vehicles have been proposed for this payload class, including the Atlas V Heavy and the Falcon 9 Heavy, but neither of these vehicles have entered focused development status as of the time of writing. For that reason, the Delta IV Heavy will be assumed to be the standard launch vehicle for this study. While costs are proprietary and vary significantly in various publications, the assumed baseline cost of the DIVH will be $250M/flight.[4]

A number of sources are available addressing the feasibility of human-rating the Delta IV Heavy, or any other EELV. Issues are the cost and schedule impact of the human-rating process, and the additional cost per vehicle due to human-rating. This is complicated by the fact that a large portion of current EELV launch costs are associated with standing costs of supporting infrastructure and workforce required, which are (to first order) independent of flight rate. Also, costs for human-rating EELV are frequently conflated with cost impacts to the baseline Constellation architecture, such as maintaining the facilities and work force for producing solid rocket motors for Ares V if DIVH were to supplant Ares I.[5] For the purposes of this exercise, the assumption will be made that human-rating DIVH will cost $2B in nonrecurring engineering over five years. Using the NASA Advanced Mission Costing Model, this corresponds to a complete redesign of the DIVH as a Block 2 system of“high” complexity. In terms of recurring costs, the human-rated vehicles will carry a 50% surcharge over the baseline costs for the same generation of DIVH vehicles.

C. In-Space Operations

The underlying assumption of the Constellation approach (over and above "Apollo on steroids") was that space operations must be minimized to the greatest extent possible, while returning to the proven Apollo paradigm of lunar orbit rendezvous. This is frequently referred to as "the lesson of International Space Station", which has required more than a decade of dedicated shuttle flights to near its final configuration.

However, the avowed goal of minimizing space operations is inherently questionable. Over seven years in the 1960's and early 1970's, the United States devoted approximately 10,000 crew hours in space to testing, training, and executing a lunar exploration program. In the forty years since then, more than 500,000 crew flight hours of experience have been accumulated in the development and routine use of techniques for in-space operations. Yet, the Constellation paradigm disparaged in-space operations, to the point of only allowing extravehicular activities (EVAs) from the Orion crew vehicle in the case of safety-of-flight emergencies. Given the magnitude of the task under consideration, *all* existing capabilities should be on the table for consideration.

Given the lack of a "heavy-lift vehicle" in the considered transportation architecture, two approaches will have to be used in unison to reach a feasible system: smaller vehicles (as compared to Constellation), and in-space assembly. It is probably a misnomer to refer to the planned operations as "assembly", at least in the context of ISS construction, as the on-orbit activity will be limited to rendezvous and docking. This has been accomplished hundreds of times in the history of space travel, and is a relatively simple and well-understood operation.

One trade study to be performed is on the ideal utilization of the launch vehicle, as it affects the exploration concept of operations. Given the size of the Orion spacecraft and the payload limitation of the Ares I launch vehicle, Constellation operations began with a docking between human and cargo vehicles in LEO. A major component of the Ares V payload was the Earth departure stage (EDS), which performed the translunar injection maneuver with a ΔV of 3.1 km/sec.

Figure 1 shows the performance of the Delta IV Heavy with regards to the orbital staging location. The graph represents the break-even boundary in terms of delivered payload between staging in LEO or directly inserting the payload into the transfer orbit to the Moon (or other destination) and performing no orbital operations in low Earth orbit. Due to the performance of the DIVH second stage, only a highly mass-optimized LOX/LH2 upper stage could equal the payload performance for direct insertion. To get the maximum advantage with the restricted DIVH payload, this results leads to the decision that all orbital operations in this program will be done at the destination (or an alternate high-orbital staging point) rather than LEO.

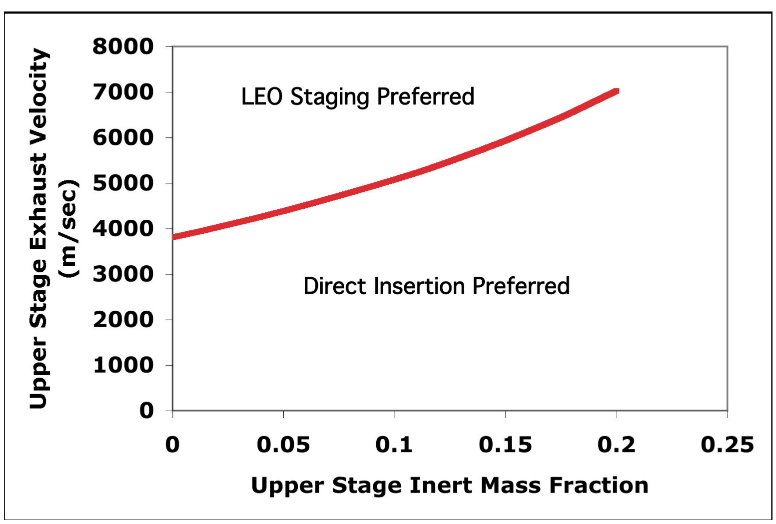

Figure 1. Delivered Payload Break-Even between LEO Staging and Direct Insertion to Destination

D. Vehicle Technologies

A number of technologies have been proposed to enhance and extend the utility of architectures featuring in-space operations. One which has received a fair amount of attention recently has been orbital propellant depots, with particular emphasis on transfer and storage of high-performance cryogenic propellants, typically liquid oxygen (LOX) and liquid hydrogen (LH_2). While on-orbit propellant transfer and cryogenic storage have great potential, the present emphasis on minimum cost/earliest function architecture argues against new technologies in the early phases of the program. This analysis assumes that extended ΔV maneuvers will be performed by multiple propulsive stages, operating serially.

A related issue is the choice of propellants. A parametric analysis of prior vehicles, based on propellant types, is shown in Table 2. Although LOX/LH_2 has clear advantages in terms of exhaust velocity/specific impulse, the boil-off rate of cryogenic propellents limits the on-site loiter time available. Due to the multiple launch nature of any extended mission in this architecture, it is essential to allow extending orbital loiter time to accommodate a launch or enroute failure which necessitates replacements to the modular elements (such as propulsive stages) which were lost. Also, the nature of this analysis argues in favor of well-developed and understood systems, such as storable hypergolic propellant systems. For these reasons, all propulsion systems in this architecture will be based on storable propellents, such as nitrogen tetroxide (N_2O_4) and some form of hydrazine (N_2H_4), such as unsymmetrical dimethyhydrazine (UDMH). In one accommodation to modern technology, it will be assumed that storable rocket engines in this time frame will be available with specific impulses of 320 seconds, which is less than the best available today, but somewhat above the average (from Table 2) of 312 seconds.

Table 2. Heuristic Propulsion System Parameters

Propellant Type	Exhaust Velocity (m/sec)	Inert Mass Fraction
LOX/LH2	4273	0.075
LOX/RP-1	3136	0.063
Storables	3058	0.061
Solids	2773	0.087

III. Developing Vehicle Designs

The essential building block of this architecture is an orbital maneuvering stage (OMS) capable of performing lunar orbit insertion, trans-Earth insertion, and portions of the ascent and descent to the lunar surface. Given the use of a DIVH for direct insertion into the lunar transfer orbit, and the use of a storable propulsion system with a specific impulse of 320 seconds, the mass ratio for a braking maneuver into a 100 km circular low lunar orbit (LLO) is 0.7657. Based on prior systems, the stage inert mass fraction (inert mass over total stage mass) for the orbital maneuvering stage (OMS) is chosen to be 0.1. Given this information, it can be shown that the payload delivered to LLO would be 7385 kg, with an OMS total mass of 2598 kg. This system would use the entire propellant supply of the OMS to brake the payload into lunar orbit.

However, it should be remembered that the basic concept is to use the OMS as a modular system for the lunar descent and ascent phases as well. A cargo flight could not bring three fully loaded orbital maneuvering stages of this size, due to DIVH injection mass limits. Two fully loaded OMS could be delivered, but the system would have more than 2000 kg of payload margin. Clearly, the choice of OMS size is critical to the feasibility of the architecture.

A larger OMS would be capable of bringing performing the lunar orbit insertion (LOI) burn with a correspondingly larger payload, but this would exceed the insertion limits for the DIVH. Alternatively, sizable payloads could be delivered with larger stages by offloading propellants; that is, launching the system with the OMS for LOI partially loaded with propellants. The relationship between total payload, delivered payload, and propellant offload is shown in Figure 2. In this figure, the propellant tanks (with a nominal capacity of 6255 kg) must be offloaded by the specified amount to achieve maximum payload mass delivered to LEO. Since a larger OMS has a higher inert mass, there is some loss of maximum payload with increasing OMS size, but the marginal payload loss for a larger OMS is acceptable.

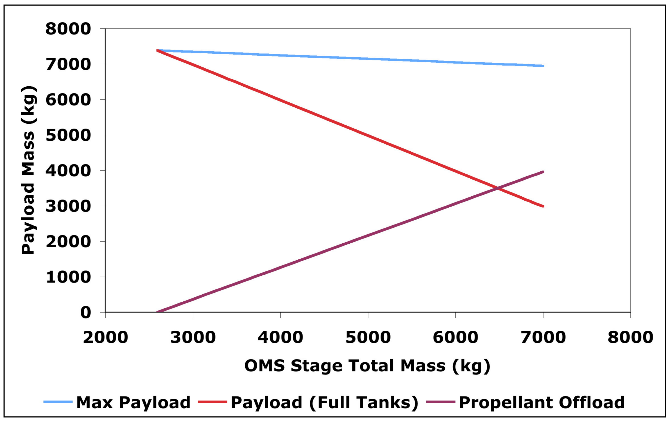

Figure 2. Effect of Orbital Maneuvering Stage Mass on Delivered Payload

The challenge is now to decide on an OMS size which will support lunar mission logistics with maximum load factors and minimum wasted payload. One approach which proved to be advantageous was to observe that an OMS with a 6950 kg design mass, loaded with 2339 kg of propellant (or 37.4% of the total propellant capacity of 6255 kg) can deliver a payload of 6950 kg to LLO. Thus, a partially loaded OMS can deliver a second, fully loaded OMS into LLO while fully utilizing the direct injection capacity of the DIVH.

The challenge is more complex, in that the effort to maximize load fractions and minimize excess payload capacity also pertains to the ascent and descent phases of lunar landing. Starting with ascent and working backwards, one fully loaded 6950 kg OMS on the lunar surface can provide the 2334 m/sec ΔV for ascent to LLO with a payload of 4966 kg. This mass is tentatively acceptable as a crew cabin mass, which will be discussed more fully in a later section. The combination of on OMS and the 4966 kg crew module represents a landed payload mass of 11,916 kg.

The next step is to determine how to use the same modular OMS to effect the landing of the OMS/crew module which makes up the lunar ascent vehicle. While it is highly desirable to utilize the same OMS design everywhere, it will be necessary to design a specific stage for the terminal landing maneuver, incorporating landing gear, crew ingress/egress systems, and landing avionics. Keeping the 6950 kg gross mass for this terminal landing stage (TLS) to optimize the lunar delivery system, the assumed stage inert mass fraction of 0.15 (as compared to 0.10 for the OMS) produces a stage inert mass of 1042 kg and a maximum propellant load of 5908 kg.

An iterative investigation of staging options produces a three-stage descent design, formed by two fully loaded OMS and the TLS components. The first OMS will burn to propellant depletion and produce a ΔV of 664 m/sec. The empty OMS will be jettisoned to crash on the moon, and the second OMS will supply an additional 870 m/sec. This empty stage will also be jettisoned in the landing approach, and the TLS will provide 1178 m/sec, for a total ΔV capacity for the system of 2712 m/sec. This is an excellent match for the design ΔV of 2706 m/sec, and will be adopted as the baseline approach to landing.

The issue of crew module size must now be revisited. This approach assumes that the same crew module is used for launch, transit, orbit, landing, return, and entry. (In this aspect, the current concept of operations, or CONOPS, is more similar to the Apollo-era Earth orbital rendezvous approach than to the final Apollo solution of lunar orbital rendezvous.)

Figure 3 shows the available historical data on crew module sizing across U.S. and (partial) Russian programs. The 4966 kg design point is more than twice as large as Gemini or the lunar module, 70% larger than Soyuz, and only 15% smaller than Apollo. It should also be pointed out that each of these (except for the lunar module) are full Earth entry vehicles incorporating heat shields and recovery hardware. On the basis of these past designs, it is not unrealistic to assume that a three-person spacecraft cabin could be developed with a total mass within the 4966 kg limit.

As an aside, it might be noticed that nowhere was there any discussion of crew size until now. Rather than adopt a crew size requirement and drive the system to meet it, the approach taken was to develop an overall system based on the DIVH vehicle, and then see what a feasible crew size might be. Based on

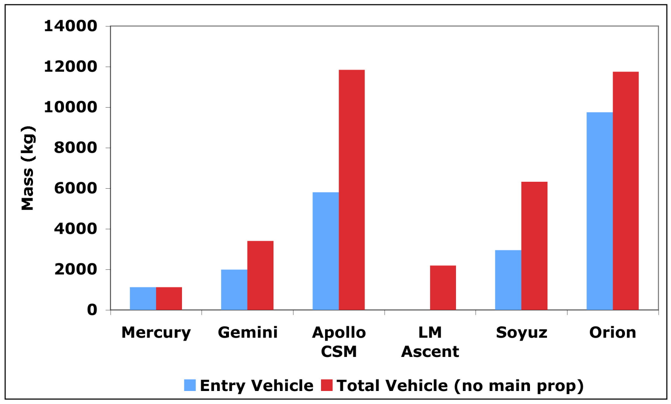

Figure 3. Historical Spacecraft Masses

historical data, there is no doubt that this system would easily accommodate two crew; since the mass limit is 15% below Apollo (which could fit five crew in a contingency), it is not unrealistic to assume the combination of smaller vehicle size and advanced materials options would provide sufficient design margin to allow a three-person crew.

Alternately, it should be pointed out that nearly 19% of the Apollo command module mass consisted of heat shields and recovery devices, such as the parachutes. Without these systems required only for Earth entry, the Apollo CM mass (otherwise fully loaded) was only 4713 kg. While the landing cabin mass is limited to 4966 kg, the delivery system is capable of bringing a 6950 kg payload to LLO. The difference is 1984 kg which can be delivered with the crew cabin to LLO, as long as it is left in orbit when the lunar descent begins. Conceptually, the heat shield and recovery systems might be designed to be modular in nature and detachable on orbit, so that these heavy systems needed only for Earth entry do not have to be landed on the Moon. This would be facilitated by concepts such as the ParaShield concept.[6] While not included in this baseline for the sake of simplicity, future research will investigate the possible benefits of this additional technology.

IV. Building Evolutionary Concepts of Operations

At the start of operations in this architecture, four vehicles have to be developed: a human-carrying spacecraft, capable of atmospheric entry from the moon (11 km/sec); a standard orbital maneuvering stage and a terminal landing stage, both specified above; and the Delta IV Heavy launch vehicle has to be human-rated, although it should be emphasized that the majority of DIVH launches in this architecture are cargo missions which will fly on the "stock" version of the DIVH. Initial efforts will focus on the spacecraft and human-rated DIVH (designated "DIVH(H)" for shorthand), with an aim of demonstrating orbital flight by 2016. In parallel, the OMS will be developed and checked out. This will allow cargo access throughout cislunar space, including low lunar orbit.

The combination of the spacecraft and OMS fill the payload capacity of the DIVH(H), and immediately allow a number of missions beyond LEO:

- circumnavigation of the Moon in a free return trajectory
- geostationary orbit
- all five Earth-Moon libration points

Effectively, due to the selection of a system based on the translunar insertion payload of the launch vehicle, any point in cislunar space except the lunar surface becomes immediately accessible to humans, within the life support capabilities of the spacecraft.

With the completion of the terminal landing stage (itself a modification of the OMS), the basic architecture for the first phase of this system will be complete. The DIVH can launch a fully fueled landing stage to the moon with an offloaded OMS to perform LEO insertion. A second DIVH can send cargo to rendezvous and dock to the TLS, which will then perform the entire landing maneuver, as shown in Figure 4. Based on the TLS mass and performance parameters, the maximum payload for a lunar landing in this mode is 3270 kg. Since the second DIVH carries nothing except cargo, this is 3680 kg short of a fully loaded condition. As shown in Figure 4, the second DIVH can transport two cargos: one bound for the lunar surface, and one slightly larger cargo package to be left in LLO.

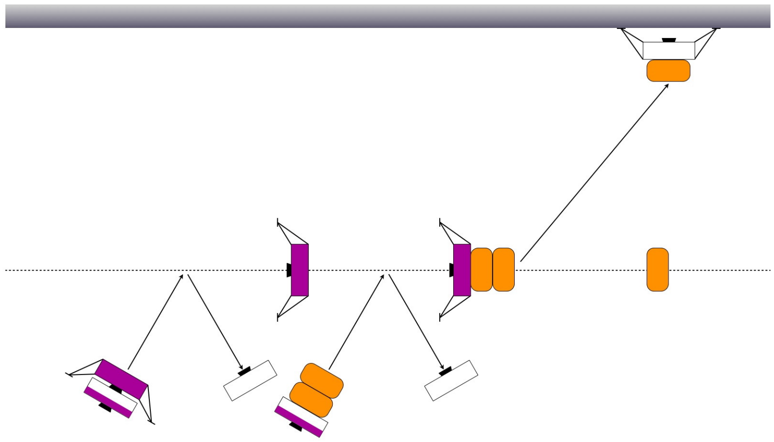

Figure 4. Single-Module Cargo Landing (Landed Cargo Mass 3270 kg, LLO Cargo Mass 3680 kg)

While this approach maximizes the single-module payload to the lunar surface, it does require the use of two DIVH launch vehicles, with the accompanying cost implications. An alternate approach, as shown in Figure 5, recognizes that the sum of the masses of the TLS and cargo must add up to 6950 kg, which is the limiting value for payload delivered to LLO by a single DIVH. By also offloading some of the TLS propellant, a single DIVH can transport the entire stack to LLO directly. While it is assumed the stack goes into LLO for the purposes of orbital phasing to hit the desired landing site, no on-orbit operations are required for this lander mission. This approach is capable of landing 1890 kg of cargo on the lunar surface, using a 68% propellant loading on the TLS.

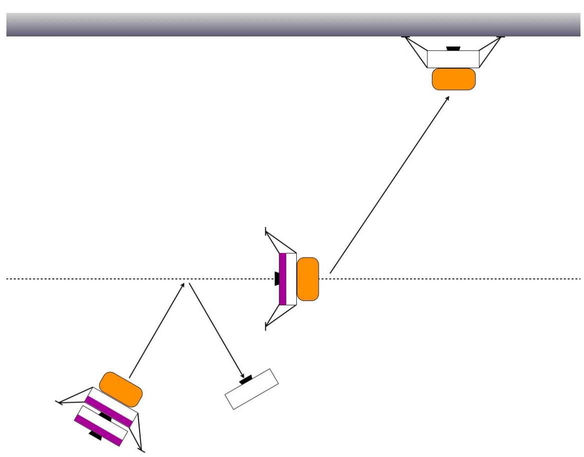

Figure 5. Single-Module Cargo Landing (No Orbital Operations Required) - Landed Cargo Mass 1890 kg)

This type of analysis can be extended to consider other numbers of stages in carrying cargo to the lunar surface. The payload totals for the logical set of cases is presented in Table 3, which includes the two previous cases. It should be noted that the compilation of DIVH flights in this table does not include flight required to launch the cargo, except for the offloaded-TLS cases.

Table 3. Cargo Delivered to the Lunar Surface

Stage Configuration	DIVH Launch Vehicles Required	Cargo to Surface (kg)
TLS (offloaded)	1	1890
TLS	1+ cargo	3270
OMS/TLS (offloaded)	2	4289
OMS/TLS	2 + cargo	7628
OMS/OMS/TLS	3 + cargo	11,916

Although offloading TLS propellant to deliver the landed cargo in the same mission as the TLS is feasible with added OMS stages, it does not in general offer any advantages over the use of a fully fueled TLS. For the case listing in line 3 of Table 3, the TLS is at a 27% propellant load factor, which decreases as the cargo is increased with larger numbers of OMS modules used.

A similar approach to LLO staging is used for human space missions to the Moon. In order to do this, the human crew module size must first be addressed. Two crew module sizes have been considered up to this point: a 6950 kg module, which is constrained by the DIVH(H) injection mass into the translunar orbit, and a 4966 kg module, which represents an optimized mass taking maximum advantage of the limited system infrastructure described so far. Figure 6 shows the sequence used to support a human lunar orbital mission, assuming the maximum size crew module. Since there is insufficient DIVH payload to enable both lunar orbit insertion and departure, a second OMS module has to be predeployed on a separate DIVH flight. The crew cabin has to rendezvous and dock with the second OMS module to allow a safe return to Earth. This is functionally identical to the requirement for the Apollo lunar module to rendezvous with the command/service module for crew transfer before an Earth return can be performed.

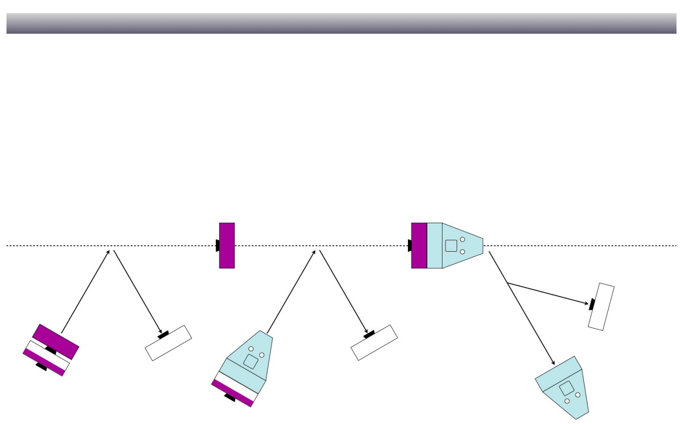

Figure 6. Sequence of Low Lunar Orbit Operations For Human Orbital Missions (6950 kg crew module)

Alternatively, Figure 7 demonstrates the simplified operations assuming that the lunar landing weight module is used throughout. At a cabin mass of 4966 kg, a standard OMS can be offloaded to a 69% propellant load, for a total DIVH payload mass of 9983 kg. This configuration produces 1779 m/sec of total ΔV, which is adequate to perform both the LOI insertion and departure maneuvers with a small (105 m/sec) reserve. More importantly, it saves the expense of a separate DIVH launch. Since the smaller crew module is required for lunar surface operations, the baseline assumption is that it will be used throughout the mission.

At this point, the systems and technologies described can be brought together into a concept for human

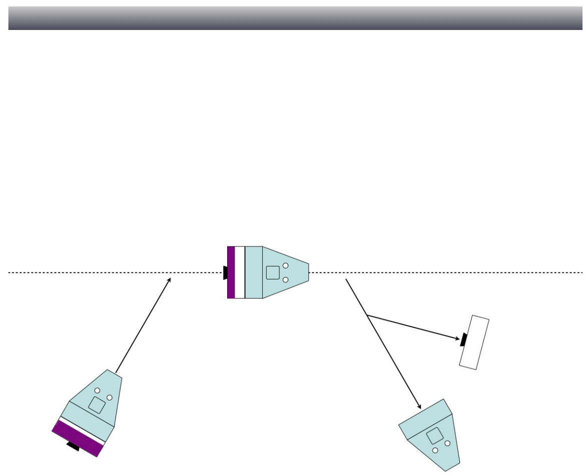

Figure 7. Sequence of Low Lunar Orbit Operations For Human Orbital Missions (4966 kg crew module)

lunar surface access. As outlined in Figure 8, four propulsion modules have to be brought together with a crew module for access to and return from the lunar surface. Dedicated DIVH flights will launch three OMS modules and one TLS module, which get assembled via automated rendezvous and docking. Using the same technique of tailored OMS propellant loading from the lunar orbital scenario of Figure 7, the crew module is launched on a DIVH(H) into the translunar orbit, where it enters LLO and rendezvous with the landing stack. The crew module's partially depleted OMS is left in orbit, and the crew module docks to the propulsion stack for descent.

Two OMS modules are fired serially to burnout, then jettisoned into a lunar impact trajectory. The TLS performs the final 1178 m/sec of ΔV to lunar touchdown, which includes equivalent hover and landing point adjustment capability as the Apollo lunar module descent stage. The Earth return OMS stays in LLO during the lunar surface mission.

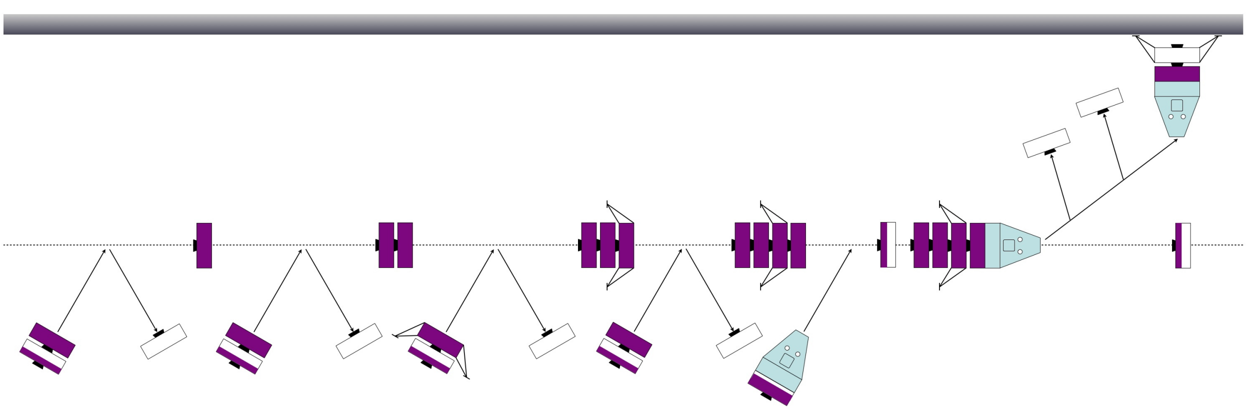

Figure 8. Sequence of Low Lunar Orbit Operations Leading to Crew Landing

The configuration of the vehicle while on the lunar surface is shown in Figure 9, shown to scale with an EVA crew. The crew cabin sits on top of the single fueled OMS module for ascent, with the empty TLS acting in the same launch pad role as the Apollo LM descent stage. The deck of the crew cabin is less than three meters from the lunar surface, providing much easier surface access than the 7 m height of the Constellation Altair lander. Since the cargo version of this vehicle does not incorporate an ascent OMS, the payload is only 2 m off the surface, allowing simpler offloading procedures for delivered cargo.

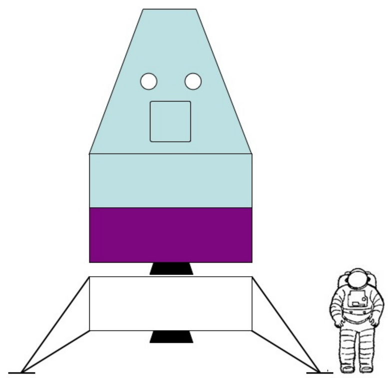

Figure 9. Human Spacecraft Assembly on the Lunar Surface

Lunar ascent is shown in Figure 10. The crew module is sized to reach orbit with the use of a single OMS module for propulsion. Once on orbit, the depleted ascent OMS is jettisoned as the crew rendezvous and docks with the OMS module used for lunar orbit insertion. The remaining propellant is sized for performing the trans-Earth insertion maneuver, and the spacecraft performs a direct entry and landing upon arrival at Earth.

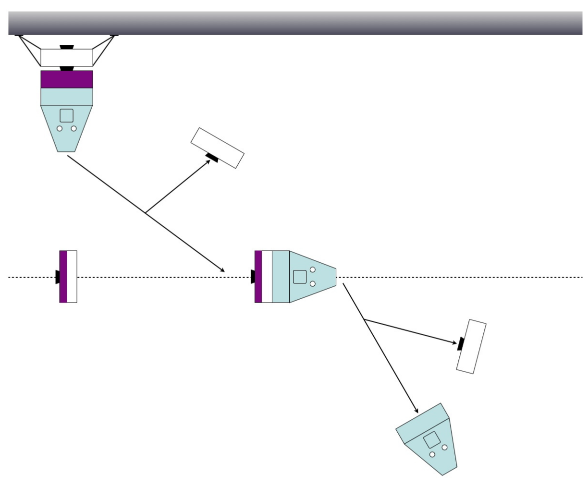

Figure 10. Lunar Ascent and Earth Departure

V. Economic Analysis

Feasibility for an architecture is not limited to mass ratios and payload fractions; in the modern age, it is also a function of economic viability. In order to fit within NASA's current human space flight budget, the overall annual cost of this program should be capped at no more than $3-5B/year. At the same time, to be politically viable, it must provide regular lunar missions, periodic "Flexible Path" missions, and support crew rotation and emergency escape from the International Space Station for the indefinite future.

System cost estimates were obtained with the NASA Spacecraft/Vehicle Level Costing Model (SVLCM), with results corrected to FY10 dollars. Table 4 summarizes this data. As specified above, $2B was allocated for DIVH human rating. The initial production price for a DIVH launch vehicle is $250M, with a 50% surcharge raising the individual cost of a DIVH(H) launch to $375M.

Table 4. System Cost Estimates from NASA SVLCM Model

System	Nonrecurring Costs, $M(FY10)	First Unit Costs, $M(FY10)
OMS	329	16.3
TLS	411	21.3
Crew Module	2460	200.8

A 15-year planning scenario was developed, starting in FY11 and proceeding through FY25. As a "lean, fast" architecture concept, major activities (DIVH human-rating and crew module development) were fast-tracked for a five-year period between FY11 and FY15. All costs were distributed on a year-by-year basis by using standard beta functions to create typical bell curve distributions. OMS development was scheduled for FY13-FY17, and TLS development from FY14-FY18 to minimize peak funding early in the program.

The next step was to devise a nominal flight schedule during the program. This schedule had to incorporate a reasonable progression of flight tests, regular lunar missions, periodic deep-space "Flexible Path" missions, and routine ISS crew rotation and resupply. Since the relatively small crew module in this study can only carry three crew, a quarterly resupply schedule was established which would result in full ISS crew rotation with 6-month stays. Table 5 summarizes the flight schedule developed.

Table 5. System Cost Estimates from NASA SVLCM Model

Mission	FY14	'15	'16	'17	'18	'19	'20	'21	'22	'23	'24	'25
Orbital Tests	1	2	1									
ISS Flights			2	4	4	4	4	4	4	4	4	4
Flexible Path				1		1		1		1		1
Circumlunar			1									
Lunar Orbit				1								
Landing Rehearsal				1								
Cargo Landing							1	2	2	2	2	2
Human Landing						1	2	2	2	2	2	2

All recurring costs are adjusted to reflect an 80% learning curve. Table 4 is modified to incorporate the required production run sizes for the component systems and average unit costs over the program, including launch vehicle production costs; the revised estimates are summarized in Table 6.

Table 6. System Production and Cost Estimates, Revised for 80% Learning Curve [All Costs in $M(FY10)]

System	Nonrecurring Costs	First Unit Costs	Number Produced	Average Unit Cost
OMS	329	16.3	73	5.75
TLS	411	21.3	29	10.6
Crew Module	2460	200.8	60	78.8
DIVH		250	70	93.4
DIVH(H)	2000	375	61	146.4

The total year-by-year budget outlays for this program are shown in Figure 11. This graphic illustrates the basic tenet of this architecture: funding is primarily used for flight, rather than new system development. The three largest cost elements are procurement expenses for the two categories of Delta IV Heavy launch vehicles and the crew modules. (It should be noted that the majority of human launches are for ISS support; since these are LEO missions, they could easily be accomplished with smaller Delta V launch vehicles. There should be almost total commonality in the human rating process between the Delta IV line and the larger

DIVH. However, for conservatism at this level of analysis, it was assumed that all human space launches occur on the DIVH(H) vehicle.)

Costs for cargo missions are dominated by launch costs, although it should be noted that the large production runs on the launch vehicles results in more than 50% savings on launch costs. The initial nonrecurring cost levels are well constrained, peaking at only $1.5B in annual costs. Yet, this human spaceflight program results in 6-7 human missions per year, including ISS logistics, two lunar exploration missions per year. and a Flexible Path mission in alternate years.

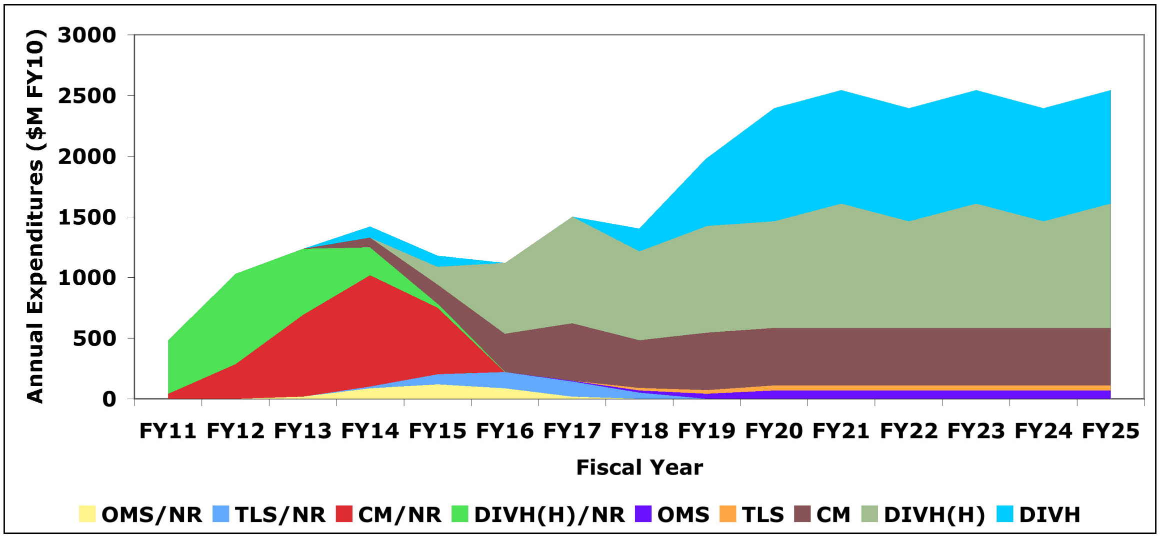

Figure 11. Annual Program Expenditures per System

VI. Future Studies

This paper has documented a limited investigation into a mission architecture which is well suited to evolutionary capabilities while adhering to strict limits on budget outlays. Due to the multilaunch nature of lunar landing missions, there is a higher likelihood of the failure of one or more components, either in the delivery to the LLO staging site or during the mission execution. Past analysis of similar architectures identified logistics strategies in terms of spares availability which mitigates the risk aspects of an assembled mission; this type of analysis needs to be applied to this architecture to budget for appropriate levels of spares. Greater attention needs to be paid to innovative targets for the Flexible Path missions, along with the definition of system requirements for extended duration missions in cislunar space. Current vehicle designs are based on historical estimating parameters; detailed designs of the component systems would allow higher fidelity in mass and performance estimates.

VII. Conclusions

If there is one thing NASA's Vision for Space Exploration has not lacked, it has been alternative suggestions for how it might be done "better". This paper has examined the specific trade space of minimum functional programs which might accomplish the same goals as the Constellation program, as expanded by the Augustine Commission, while performing the "mundane" tasks of ISS crew rotation and staying within an austere budget limit. Principal in accomplishing these goals is the insistence on using existing launch vehicles, to avoid the high costs and long lead times required by developing new launch vehicles.

The system developed here appears to be capable of meeting all of the listed requirements, and supporting all of the announced human spaceflight goals at an annual budget outlay below $3B. Even at twice the estimated cost, a human space flight program at this level of activity could inspire renewed interest in space exploration, while not starving all of NASA's other responsibilities to pay for it.

It is inevitable that the reader will be able to find "holes" in this analysis; several orders of magnitude of difference currently exist between the resources applied to this study and that which into the NASA Exploration Systems Architecture Study, or even the grass-roots "Direct" concept papers. However, it is hoped that this first analysis will illustrate the potential for a highly focused human space flight program making maximum used of existing components, and taking full advantage of in-space operational skills obtained over decades of NASA experience.

References

[1]Norman R. Augustine, et. al., "Seeking a Human Spaceflight Program Worthy of a Great Nation: Review of the U.S. Human Spaceflight Plans Committee" National Aeronautics and Space Administration, October, 2009

[2]Richard W. Orloff, "Apollo by the Numbers: A Statistical Reference" NASA SP-2000-4029, revised September, 2004

[3]–, *Delta IV Payload Planners Guide* United Launch Alliance, September 2007

[4]Michael C. Gass, *Briefing to the Review of U.S. Human Space Flight Plans Committee* United Launch Aliance, June 17, 2009

[5]David A. Beardon, John P. Skratt, and Matthew Hart, "Human-Rated Delta IV Heavy Study Constellation Architecture Impacts" Aerospace Report No. TOR-2009(2187)-9151, June, 2009

[6]David L. Akin, "Applications of Ultra-Low Ballistic Coefficient Entry Vehicles to Existing and Future Space Missions" AIAA 2010-1928, AIAA SpaceOps 2010 Conference, Huntsville, AL, April 2010