Architecture Options for Propellant Resupply of Lunar Exploration Elements

James J. Young* , Robert W. Thompson*, and Alan W. Wilhite†

Space Systems Design Lab
School of Aerospace Engineering
Georgia Institute of Technology, Atlanta, GA, 30332-0150
james_young@ae.gatech.edu

The NASA Exploration Systems Architecture Study (ESAS) produced a transportation architecture for returning humans to the moon affordably and safely, while using commercial services for tasks such as cargo delivery to low earth orbit (LEO). Another potential utilization of commercial services is the delivery of cryogenic propellants to LEO for use in lunar exploration activities. With in-space propellant resupply available, there is the potential to increase the payload that can be put on the moon, increase lunar mission durations, and enable a wider range of lunar missions.

The availability of on-orbit cryogenic propellants, either at a propellant depot or directly from a propellant transfer stage, would impact the sizing and reusability of many architecture elements, including the Earth Departure Stage (EDS), the Crew Launch Vehicle (CLV) upper stage, the Lunar Surface Access Module (LSAM), and the Service Module (SM). These vehicles are modeled to approximate the baseline established by ESAS. Methods and tools used include launch trajectory optimization with POST[6], vehicle aerodynamic analysis using APAS[5], weights and sizing using historical and physics based estimating relationships, and cost estimation using NAFCOM[7]. Uncertainty in estimates of advanced technology performance is modeled using probabilistic methods, such as Monte Carlo simulation.

The impact of propellant resupply capability on vehicle design and performance is investigated. The availability of propellant at LEO allows for many possible architecture changes, including increasing the payload delivered to the moon, decreasing the size of the heavy lift launch vehicle, or staging extra propellant in lunar orbit. Other elements could also be designed to be reusable with propellant resupply available, such as the LSAM, the CLV upper stage, and the Service Module. Several of these architecture changes are investigated, and compared via metrics such as mission reliability, cost, and the range of lunar and near-Earth mission capabilities (i.e. duration time, mission latitude, missions to Lagrange points) they allow.

Nomenclature

CaLV	=	computer aided design
CLV	=	cost estimating relationship
DSM	=	crew escape system
EDS	=	crew exploration vehicle
ESAS	=	crew launch vehicle
LEO	=	design, development, test, & evaluation
LLO	=	design structure matrix
LSAM	=	Exploration Systems Architecture Study

*Graduate Research Assistant, School of Aerospace Engineering, Student member AIAA.
† NASA Langley Professor in Advanced Aerospace Systems Architecture, School of Aerospace Engineering, Member AIAA.

SM = Earth to orbit
SRB = design structure matrix
SSME = Exploration Systems Architecture Study
TLI = Earth to orbit

I. Introduction

The baseline lunar architecture developed during the Exploration Systems Architecture Study requires that the cargo launch vehicle deliver the propellant required for Trans Lunar Injection (TLI). This is a substantial burden as the TLI burn requires 230,0000 lbs of propellant which is roughly 70% of the total payload delivered to LEO by the cargo launch vehicle. This is a considerable burden and significantly limits the design freedom of the architecture. Developing a reliable and cost effective means of providing propellant to LEO can help both reduce the cost of the overall lunar campaign and increase lunar surface payload capabilities. The work outlined in this paper will discuss the positive and negative effects of several architecture design options utilizing LEO propellant re-supply capabilities. These results will then be used in an ongoing study to determine if such a capability provides for a better solution for returning humans to the moon and beyond.

II. ESAS Baseline Architecture Validation

A. ESAS Baseline Architecture

The baseline reference architecture is taken from the ESAS report published in November 2005[1]. This report outlines the use of a 1.5 launch solution for returning humans to the moon. This solution includes the development of both a heavy lift launch vehicle for delivering cargo elements to LEO and a crew launch vehicle that will replace the role of the Space Shuttle in delivering humans to Earth orbit. These vehicles are designed to take advantage of Shuttle derived components to help lower development cost and improve system reliability. The baseline concept of operations for the lunar architecture is provided in Figure 1 below. This configuration will remain constant throughout the different trades and only the individual elements will be altered

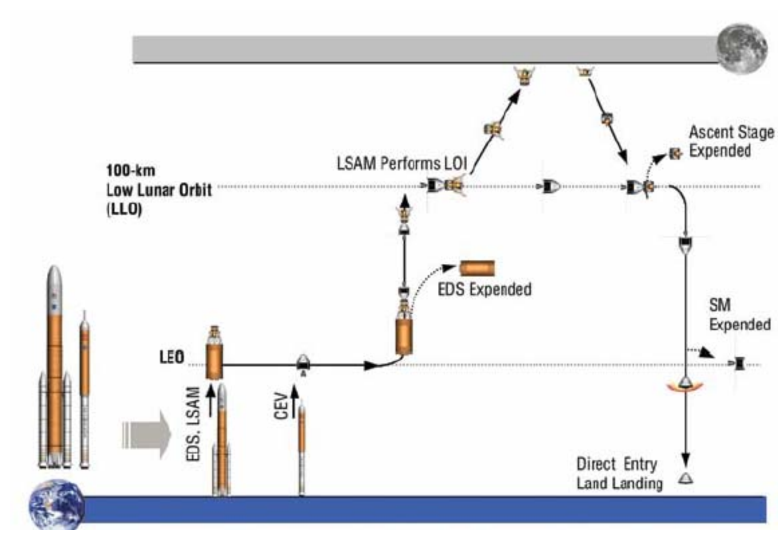

Figure 1: ESAS Baseline Lunar Architecture

The architecture has gone through a series of changes since the baseline configuration was published in late 2005, but these changes have focused mainly on the individual elements, such as an increase in the diameter of the CaLV and a change from SSME's to RS-68's on the core stage, and left the overall architecture intact. These changes should not significantly affect the results of this study as it is a comparison of the overall architecture and not the individual elements. Therefore the baseline design will remain set as of the November 2005 edition of the architecture, the comparison results through out this report will reference that work. The resulting conclusion will be applicable to the final architecture selection.

B. ESAS Baseline Architecture Comparison

The first step in this study was to develop models of the individual architecture elements used in the baseline lunar campaign. These models include a crew and cargo lunch vehicle, Earth Departure Stage, and a Lunar Surface Access Module. These elements can then be assembled together to model the overall lunar architecture. The launch vehicle models consist of three parts, an APAS aerodynamic model, a POST trajectory model, and a weights and sizing model based on historical Mass Estimating Relationships (MER's). The vehicle propulsion is fixed to the existing engine specifications. The EDS and LSAM models are weight-based models that are size based on the delta-V requirements for TLI, LOI and Lunar ascent; these values are taken directly from the ESAS report. A Design Space Matrix (DSM) was created for the architecture to aid in defining how information flows between the various design elements. This DSM is provided in Figure 2. The ascent block contains the launch vehicles, which feed payload and EDS dry weight information into the in-space block. The TLI propellant is then calculated and returned to the launch vehicle block as an input for the payload that must be delivered by the cargo launch vehicle. This iteration continues until the two converge on a closed design. The lunar payload (LSAM + CEV) is then fed to the lunar block to determine the size of the LSAM and the amount of payload delivered to the lunar surface.

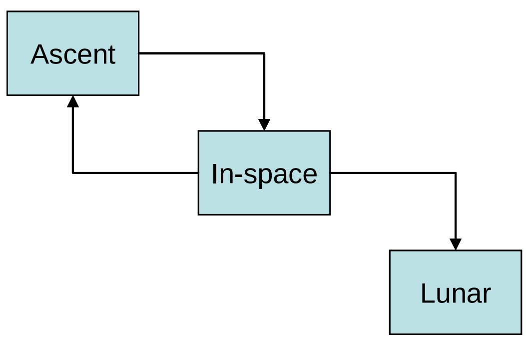

Figure 2: Lunar Architecture Design Space Matrix

These vehicles were designed and calibrated to match the ESAS results so that these models can provide a direct comparison when the propellant re-supply elements are added to the architecture. The vehicle weights, propellant requirements, and payload delivered were all within a few percent of the ESAS results. These results are provided in the following sections.

1. Cargo Launch Vehicle

The cargo launch vehicle (CaLV) is used to deliver large non-human cargo elements to LEO for lunar exploration missions. It is designed to deliver in excess of 320,000 lbs of payload (LSAM + TLI propellant) to LEO. The 2nd stage of the CaLV is also used as the EDS and performs the TLI burn once mated with the CEV and service module (SM). A weight comparison between the CaLV and the model was within 2%. A summary of these results is provided in Table I.

Table I: Cargo Launch Vehicle Comparison

	1st Stage			2nd Stage		
	ESAS	**Model**	**% Error**	**ESAS**	**Model**	**% Error**
Dry Mass w/ growth (lb)	194,997	194,600	-0.20	42,645	42,500	-0.34
Gross Mass (lb)	2,430,643	2,432,000	0.06	640,171	650,800	1.66
Ascent Propellant (lb)	2,215,385	2,217,000	0.07	264,690	263,600	-0.41
Stage Length (ft)	210	210	0.00	75	75	0.00

The ESAS report also provided a set of trajectory plots for the ascent phase of the CaLV, which provide another means of comparison between the two models. The main difference between the two trajectories is that the model pulls up out of the dense part of the atmosphere later and thus is exposed to a higher dynamic pressure for an additional 10s. The dynamic pressure peak remains the same at around 550 psf. The other aspects of the trajectory, (altitude, velocity, and acceleration) match up well with the published ESAS results. A comparison of the trajectory profiles are provided in Figure 3 and Figure 4.

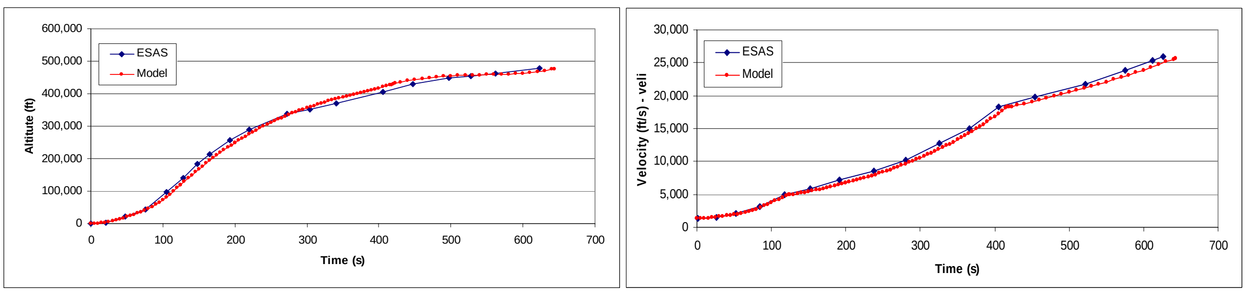

Figure 3: CaLV Altitude and Velocity Profile Comparison

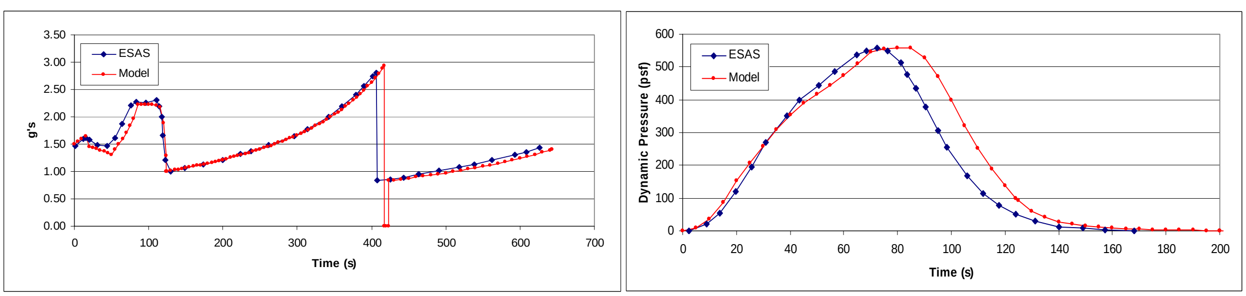

Figure 4: CaLV Acceleration and Dynamic Pressure Profile Comparison

2. *Crew Launch Vehicle*

The crew launch vehicle (CLV) is designed to provide reliable transportation of humans from the earth's surface to LEO. The CLV is similar to the CaLV in that it uses Shuttle-derived launch vehicle components. Its first stage is powered by the same 4-segment Solid Rocket Boosters (SRB) used on the Shuttle and the 2nd stage is powered by a single SSME. This 2nd stage engine has been changed to a J-2S in current design, but this study will continue to use the original SSME design. The goal of designing the CLV with Shuttle-derived components is that it will improve the reliability of transporting humans to LEO by a factor of 10.

Table II: Crew Launch Vehicle Comparison

	1st Stage			2nd Stage		
	ESAS	Model	% Error	ESAS	Model	% Error
Dry Mass w/ growth (lb)	180,399	180,400	0.0	38,597	39,400	2.06
Gross Mass (lb)	1,309,305	1,293,500	1.25	474,739	472,800	0.40
Ascent Propellant (lb)	1,112,256	1,107,800	0.40	360,519	362,700	-0.88
Stage Length (ft)	133	133	0.0	105	105	0.0

The model of the CLV again matched the results provided in the ESAS report to within a few percent. These results are provided in Table II. A more extensive look at the CLV model can be found in a 2005 paper published by the Space Systems Design Lab (SSDL)[2]. The model developed from this work was used in generating the results of this study and follows the same design methodology discussed previously.

3. Lunar Surface Access Module

The Lunar Surface Access Module (LSAM) provides both access to the lunar surface and a habitat for humans during exploration missions. Once in Low Lunar Orbit (LLO), the crew transfers to the LSAM for descent to the surface. The crew utilizes the LSAM as a base of operations while on the surface, at the end of the mission the LSAM separates and a smaller ascent stage is used to return the crew to the CEV in LLO. In this study, it will be assumed that the mission ends at lunar landing and that all of the remaining architecture elements remain constant and are not affected by the addition of propellant re-supply capabilities.

Table III: Lunar Surface Access Module Comparison

	Dry Weight (lb)		Gross Weight (lb)		Propellant (lb)	
	ESAS	Model	ESAS	Model	ESAS	Model
Descent	13,500	13,600	77,300	80,200	55,300	56,400
Ascent	11,300	11,700	23,800	24,400	10,400	10,600

The model again provides a close match to the ESAS reports, a summary of these results can be found in Table III. The largest difference between the two models is seen is in the gross weight of the descent stage. There is a 3.5% error between the two models that comes from a slight increase in the propellant requirements calculated for the mission, which in return causes the size of the vehicle to grow slightly. The error is still small and should not provide any significant bias in the final finding of this study.

It is concluded that these models provide a good representation of the ESAS results and will provide a good baseline of comparison when looking at how each element of the architecture changes with the addition of propellant re-supply capabilities. These models will be used in a series of trade studies to see how the architecture performs when propellant re-supply is added. These trade studies will include: burning addition propellant in the CaLV 2nd stage to increase the payload capability, removing the TLI propellant from the CaLV and scaling down the vehicle to deliver only the LSAM payload, and the case where the CaLV is completely removed from the architecture and the CLV or an EELV is used to deliver cargo elements to LEO. There are many other trade studies available that could be considered, but these will provide for an initial understanding of how the architecture performs with the addition of propellant re-supply capabilities.

III. Propellant Re-supply Trade Options and Results

A. Increase in CaLV 2nd Stage Propellant Usage

The first trade study was to determine what the maximum payload the CaLV could deliver to LEO. An increase in payload could be achieved by increasing the ascent propellant used by the upper stage. This would require some to all of the TLI propellant to be used during the ascent phase leaving the EDS empty upon reaching LEO. To avoid confusion here, the term mission payload will be used to represent the LSAM and any additional payload carried to the lunar surface, while the term payload will refer to the mission payload plus TLI propellant. The current CaLV can deliver a payload of 325,000 lbs to LEO, of which 101,000 lbs is mission payload. When all of the EDS propellant is used to reach LEO, the payload delivered is 352,000 lbs, which represents an 8.5% increase over the current design. This is a fairly small increase, considering the 2nd stage propellant usage was almost doubled, this is due in part to two reasons. The first is that the gross lift off weight of the vehicle increases by 250,000 lbs to account for the additional mission payload while still carrying the same propellant load. This significantly decreases the capability of the core stage, which reaches a velocity of 4,000 ft/s less at separation. The second is that the upper stage T/W is decreased from 0.84 to 0.61, making the stage less efficient than the baseline design. Putting these issues aside, there is still an increase of the payload delivered to LEO and a significant increase in the mission payload. The main concern with delivering this amount of payload is that it cannot physically fit on the current CaLV configuration. An increase in the fairing diameter will allow for larger payloads to fit on the CaLV; however, the use of a larger fairing decreases the overall payload capability of the vehicle do to an increase in the weight of the fairing and an increase in drag losses. An optimization was done in order to determine the maximum payload that could be delivered by the CaLV. The optimizer was allowed to vary both the fairing diameter up to 22ft and ascent propellant used while keeping the configuration of the vehicle constant (height constraint). The payload density was also constrained to remain the same as that calculated from the baseline design. This resulted in the maximum mission payload decreasing from 352,000 lbs to 260,000 lbs and the fairing diameter increasing to 22 ft. These results are provided in Table IV along with the ascent propellant required and the propellant needed for the TLI burn. This final case provides for more realistic payloads.

Table IV: Maximum Payload Delivered, Fairing Diameter 22ft

	Mission Payload (lb)	Ascent Propellant Used (lb)	Remaining TLI Propellant (lb)	Required TLI Propellant (lb)
ESAS	101,000	264,000	224,000	224,000
Model	259,300	413,000	86,400	433,200

The propellant required for the TLI burn varies directly with the mission payload as noted in Figure 5. As the mission payload increases, the cost to deliver sufficient TLI propellant also increases. An important question arises as to when propellant re-supply becomes to expensive to effectively deliver an additional pound of payload to LEO. An ongoing study is being conducted to look at how increasing the payload delivered to LEO affects the economics of the lunar architecture. The results in Figure 5 includes various payload fairing densities and so the maximum payload capability is greater than that provided in Table IV, which is for a payload density equal to the baseline design.

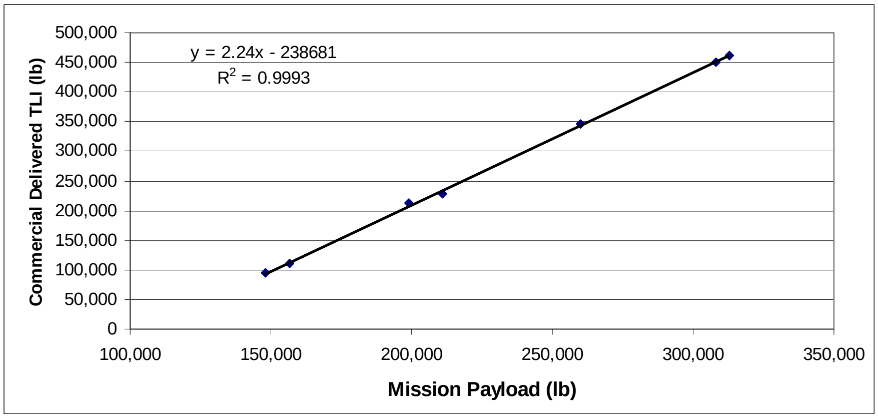

Figure 5: Mission Payload vs. Propellant Delivery Requirements

The increase in mission payload delivered to LEO does not translate directly into payload delivered to the lunar surface. The increase in mission payload is for the entire LSAM system, when the payload to the lunar surface increases so the propellant requirement for LOI and descent (both are done by the LSAM). Therefore the increase in mission payload must be balanced between an increase in propellant and lunar surface payload. The baseline design is capable of delivering 3,000lbs of payload to the lunar surface this solution provides a lunar surface payload capability of 85,500 lbs about 50% of the increase in mission payload.

B. Scale Down CaLV to Deliver Lass Payload

This second study considered the case where all of the TLI propellant is removed from the EDS Stage, it is assumed to be delivered separately. This reduces the payload delivered to LEO by 70%, and results in a smaller launch vehicle requirement. In order or determine how small the launch vehicle could be and still deliver the required mission payload the number of engines on both the upper and core stages were decreased. This reduction continued until there was a single J-2S on the upper stage and 2 SSMEs on the core, decreasing the number of SSMEs further results in a non-closed trajectory. The configuration still provided more than the needed amount of thrust, to decrease the thrust further the two 5-segment SRBs were replaced by two 4-segment SRBs. This final configuration provided for a closed design and meet the mission payload requirement of 101,000 lbs. A comparison of the resulting vehicle and the baseline design are provided in Table V.

Table V: Scaled Down CaLV Comparison

	Length (ft)	Diameter (ft)	SSME	J-2S	SRB
ESAS	357	27.5	5	2	5-segment
Model	242	20	2	1	4-segment

In addition to decreasing the number of engines the length and diameter of the vehicle also became smaller than the baseline design. Originally the diameter was left the same as the baseline design, but this configuration had a very poor length/diameter ratio. The diameter of the vehicle was decreased to 20 ft to improve this, while the fairing diameter was left at the original 27.5ft so that it could continue to accommodate the baseline LSAM. An optimization could still be done on designing a CaLV that was required to deliver only the mission payload, in this design the baseline configuration structure was kept in tact and only reduced in size until it couldn't be reduced anymore. A more optimal design could be found if different engines were investigated, but this study provides an initial look at how the size of the vehicle can change if the TLI propellant is delivered by a separate means.

A look at how the configuration of these two designs compare to the baseline design is provided in Figure 6. These two designs provide opposing looks at how propellant re-supply capabilities can affect the design of the lunar architecture. The first trade study provides an increase in the mission payload while the second study reduces the overall size of the vehicle while providing the same mission payload.

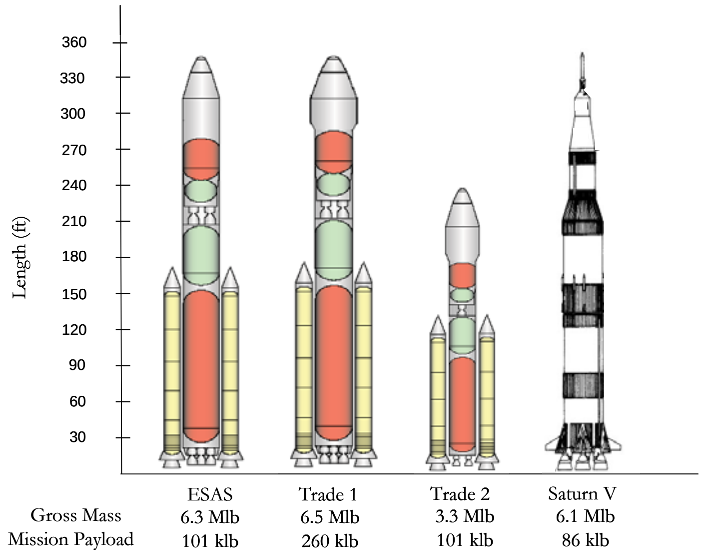

Figure 6: CaLV Trade Study Comparison[1]

C. Remove CaLV from Lunar Architecture Design

The final study investigated was to look at the case where the CaLV would not be developed and the CLV would be used to deliver both the CEV and LSAM to LEO (separate launches would be required). The benefit of this would be to free up the resources needed to develop and build the CaLV. This would amount to billions of dollars in saving over the entire lunar campaign. In addition to delivering the CEV to LEO the CLV would be required to deliver the LSAM to LEO and perform the TLI burn. The CLV has the capability to deliver 60,000 lbs to the reference orbit. The gross weight of the LSAM is 101,000. This includes 55,000lbs of LOX/LH2 descent propellant. The LSAM can be delivered to LEO with only 25% of the descent fuel and be within the payload capacity of the CLV. The remaining fuel can be provided on orbit in the same manner that the EDS stage is re-fueled. The 2nd stage of the CLV must also be able to perform the TLI burn. The TLI burn requires a 10,333 ft/s delta-v maneuver, this equates to a propellant requirement of 230,000 lbs. The 2nd stage of the CLV can hold a maximum of 360,000lbs of propellant, well above the require propellant mass.

The concern with placing the LSAM on top of the CLV is that the current LSAM configuration can not realistically fit on the top of the CLV. The current fairing diameter is at the upper limit of what can be placed on top of the solid first stage and the LSAM is considerably larger. It would be difficult to design the LSAM to with fit on the CLV with out decreasing its current capability or requiring multiple launches to deliver the vehicle. Other launch vehicles were considered to replace the CaLV. These vehicles have the performance capability to deliver the LSAM to orbit, but they have the same fairing limitation as the CLV. The results from this study show just how much the lunar architecture can de affected by the development of propellant re-supply capabilities.

IV. Propellant Re-supply Capabilities

A. In-space Propellant Re-supply Options

There are two main options for in-space propellant re-supply. The first is to deploy a propellant depot to Earth orbit. This depot is capable of holding large amount of propellant for an extended period of time. The benefit of a propellant depot is that the delivery of propellant is not directly linked to the exploration mission[3]. The propellant is delivered to orbit in such a way that sufficient propellant is always available for the next exploration mission. Many designs for propellant depots include the use of cryogenic coolers which limit the amount of propellant that is lost to the environment. This allows propellant to be stored on orbit for extended periods. The additional benefit is that the delivery vehicle never needs to come in contact with the specific mission elements helping to keep the design of the delivery system simple and low cost. The down fall of a propellant depot is that additional hardware must be designed, deployed and operated, each of these increase the cost of the overall architecture. The alternative design is the use of direct propellant transfer. In this case the propellant is delivered directly from the delivery vehicle to the EDS. This is a simpler design, but requires a stricter launch schedule since the propellant can't be delivered until the EDS is in orbit. This becomes more of a concern as the number of launches required to re-fuel the EDS increases. Additional requirements will be place on the delivery vehicle as it comes into contact with the very expensive lunar architecture elements. This will likely increase the cost and complexity of the delivery vehicle. The benefit of this design is that it does not require the development and deployment of a propellant depot and so offers a lower infrastructure cost, but increases the requirements on the delivery vehicles. Additional work in this area will look at the economic point where each of these solutions makes sense and what effect does the additional regulation play on the effectiveness of using direct transfer options.

B. Propellant Re-supply Delivery Options

The key to propellant re-supply becoming apart of the lunar architecture and future exploration missions is the development a low cost and reliable delivery option. In the near future this is likely to come from the development of a small to medium commercial launch vehicle capable of launching many vehicles within a short period of time. There is an emerging field of launch vehicles that is generally referred to as "responsive space" access vehicles. The goal of this new launch vehicle industry is to reduce launch vehicle complexity and government involvement resulting in a lower cost and higher reliability design. There are many companies currently designing such vehicles, but none of these companies currently have a launch vehicle in the market. The closest company is Space X who plans to launch a test vehicle in November of 2006. Other options available for propellant re-supply are current EELVs, such as the Atlas and Delta family of launch vehicles. These vehicles provide a medium to large payload capability, but the cost per mission is much higher than expected from the responsive space community; however the reliability of these vehicles is much more proven. The following table compares the Space X launch vehicles to the current Atlas 552. The last column is the number of successful launches required to deliver the TLI propellant of the baseline design.

Table 6: Propellant Delivery Vehicle Comparison

Vehicle	Payload Capability (lb)	Cost/Launch ($M)	Required Launches
Falcon I	1,250	6.7	189
Falcon 9	20,500	27	12
Falcon 9S9	54,500	78	5
Atlas 552	41,900	150	6

It is important to include the cost of launch vehicle failures when looking at the total cost of propellant delivery. This has been done through the use of Monte Carlo simulation. A simulation is created to model the number of launches required to compete each re-supply mission. The simulation is run 10,000 times to generate a distribution of the launch cost based on the number of failures experienced during each mission. A triangular distribution is also placed on the launch cost as the accuracy of these values is less certain. The results for the Atlas V and Falcon 9 launch vehicles are provided in Figure 7. The two curves on each plot represent different levels of assumed launch vehicle reliability. The 95% confidence value is also shown in each figure.

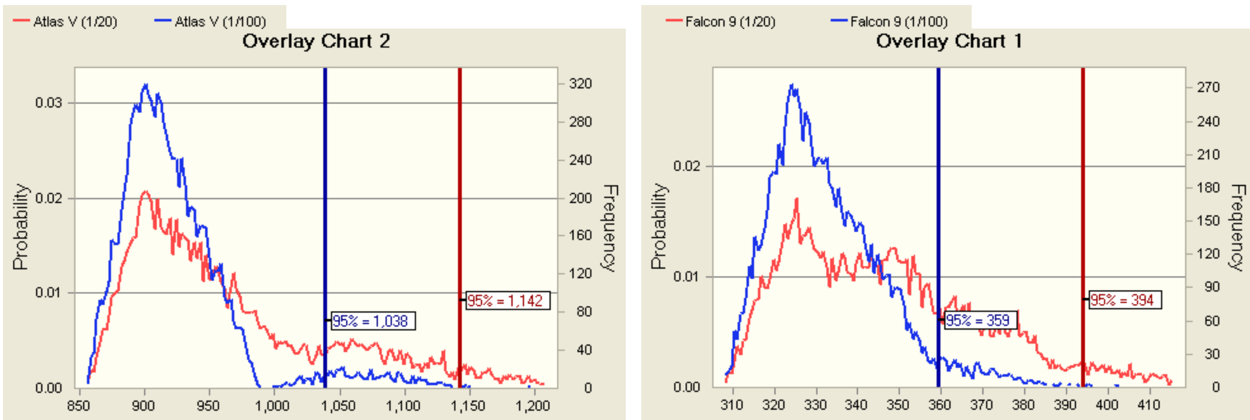

Figure 7: Monte Carlo Mission Cost Simulation Results

These results significantly lean toward the use of responsive space vehicles as they provide a much lower cost with a similar launch reliability. The concern with these vehicles is that their development is not secure and only time will tell if the development of these vehicles will come to fruition. Future studies will look at both vehicle types and make a recommendation as to what level of reliability and cost must be reached before propellant re-supply can be effectively implemented into exploration missions.

V. Future Work

The work presented in this paper is part of an on going study at the Georgia Institute of Technology under the direction of Dr. Alan Wilhite. The goals of this study are aimed at determining the point at which an architecture utilizing propellant re-supply becomes a better solution than current architecture models. The term "a better solution" is dependent on more than the performance of the architecture, cost, reliability, risk and launch schedule must also be considered. This papers discussed the possible performance benefits that propellant re-supply can provide to exploration missions. Currently an economic and reliability model are being developed to provide additional insight into these other figures of merit. The economic model will include a year by year analysis of all economic activities associated with the operation of a lunar campaign, including develop, production and operating cost. The model will also look at the cost associated with delays in the CLV that could lead to the loss of already deployed assets such as the EDS stage. The reliability model will calculate an overall architecture reliability based on the reliability of the individual elements with the architecture. The addition of a propellant re-supply element adds additional complexity and decreases the overall reliability of the architecture. Other changes in the configuration may also effect the reliability of the architecture, such as the number of engines on the CaLV. A level of risk will also be considered to account for any addition development that may be required, such as in-space propellant transfer and zero-boil off technologies. These different figures of merit will be used to evaluate the use of propellant re-supply in future exploration mission and to determine at what benefit to cost ratio does this architecture become a better solution.

VI. Conclusions

The study presented in this report provided some initial evidence as to the benefits that propellant re-supply can provide to future exploration missions. Opportunities for improvement in both performance and cost can be achieved if a low cost and highly reliable delivery method can be developed. Current delivery systems provide proven reliability, but come with a high price tag. The development of responsive access vehicles offers would provide both reliability and low cost, but there is much uncertainty in the development of these vehicles. The increase in complexity of adding an additional element decreases the over reliability of the architecture, but changes in the configuration may help to counter act that reduction. This project is continuing to develop a complete architecture model that will classify all of the costs and benefits of adding propellant re-supply option to exploration mission. The goal is to determine at what point propellant re-supply becomes a better solution than current exploration architectures.

VII. References

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