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Development of a Lunar Architecture Simulation Environment for Evaluation the use of Propellant Re-supply

James J. Young[1], and Alan W. Wilhite[2]

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)[1] 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 re-supply available, there is the potential to increase the payload that can be delivered to the lunar surface, increase lunar mission durations, and enable a wider range of lunar missions. The addition of on-orbit propellant re-supply would have far-reaching effects on the entire exploration architecture. Currently 70% of the weight delivered to LEO by the cargo launch vehicle is propellant needed for the TLI burn. This is a considerable burden and significantly limits the design freedom of the architecture. The ability of commercial providers to deliver cryogenic propellants to LEO may provide for lower cost and better performing lunar architecture.

A model of this architecture has been developed to measure the performance, cost, reliability, mission success rate, and various other criteria for a lunar architecture built around propellant re-supply. This model will provide insight into how the addition of propellant re-supply will affect each aspect of a lunar campaign and help measure the benefits and costs associated with the development and utilization of this capability. The environment itself has been developed using parametric models of the individual architecture elements to allow for the quick evaluation of different architecture trades. A morphological matrix of the different trades considered is provided in Figure 1. This provides an easy to follow outline of the different trades studies conducted using this simulation environment. In addition to being able to quickly asses different architecture designs the simulation environment must be able to provide accurate results. A validation test case was done using the results from the ESAS. The results of this validation showed that the models could predict the results of this architecture within 2 – 3%. The analysis also includes multi-criteria decision making analysis so that a ranking of the alternatives can be found for a given weighting scenario. This allows for all of the figures of merit to be included in the decision making process.

The results of the simulation will allow a decision maker to evaluate the different architecture options against the baseline design and determine what figures of merit are improved with the inclusion of propellant re-supply and if this new capability provides for a better opportunity to meet the needs of future exploration missions.

[1]Graduate Research Assistant, School of Aerospace Engineering, Student member AIAA.
[2] NASA Langley Professor in Advanced Aerospace Systems Architecture, School of Aerospace Engineering, Member AIAA.

Nomenclature

ESAS	=	Exploration Systems Architecture Study
CaLV	=	Cargo Assist Launch Vehicle
CLV	=	Crew Launch Vehicle
SM	=	Service Module
CEV	=	Crew Exploration Vehicle
LSAM	=	Lunar Surface Access Module
FOM	=	Figure of Merit
AS	=	Ascent Stage
DS	=	Descent Stage
TLI	=	Trans Lunar Injection
LOI	=	Lunar Orbital Insertion
ISRU	=	In-situ Resource Utilization
ZBO	=	Zero-boil-off
MLI	=	Multi-layer Insulation
STS	=	Space Transportation System
LCC	=	Life Cycle Cost
LOM	=	Loss of Mission
NAFCOM	=	NASA Air Force Cost Model
CER	=	Cost Estimating Relationships

I. Introduction

In 2004 President Bush addressed the nation and presented NASA plan to further space exploration. The vision for space exploration included the completion of the International Space Station, the retirement of the Space Shuttle, the development of a crew exploration vehicle, and the return of humans to the moon by no later than 2020. The Exploration System Architecture Study team was establish to develop the baseline architecture that NASA would use to return humans to the moon. A general description of the architecture established during this study is provided in Figure 1. There are six main vehicle elements in this architecture: A Crew Launch Vehicle (CLV), a Cargo Launch Vehicle (CaLV), a Earth Departure Stage (EDS), a Lunar Surface Access Module (LSAM), a Crew Exploration Vehicle (CEV), and a Service Module (SM). A more detailed description of each elements can be found in Ref. 1. The architecture was designed using heritage space components where possible to help improve the overall cost and reliability.

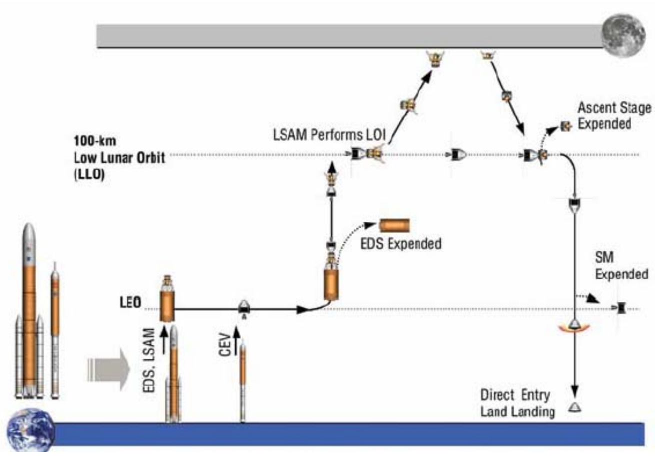

Figure 1. ESAS Baseline Architecture.

The design of the architecture focuses on maintaining a high level of crew safety while meeting the requirements of the lunar mission. This was the main driver in the development of a two launch solution, where crew and cargo are delivered to LEO separately. In addition to this the architecture has the capability to return to Earth at any point during the mission, and is required to maintain a long term stay in LEO before lunar transfer. These among many other requirements have put a strain on the design of the architecture specifically on the design of the launch vehicles and lunar lander. The CaLV is close to exceeding the physical limitation that can be manufactured with the current facilities and there is still question as to whether it can meet the performance requirements of the lunar architecture. When one considers the mass growth of historical systems[2] from concept to final production it is not difficult to imagine that the architecture elements will see considerable mass growth, especially the lunar lander which has very little historical evidence to rely on, and this could push the design up against their physical limitations.

The largest burden on the architecture is the delivery of the in-space propellant from the Earth's surface. This propellant makes up currently between 85 – 90% of the total payload delivered to LEO via the CaLV. If this could be augmented by another delivery option then improvements in both the LEO and lunar surface payload capabilities could be achieved as well as a reduction the overall life cycle cost of the campaign. The focus of this work will be to investigate the use of on-orbit propellant re-fueling strategies and how they could effect the design of the lunar architecture. The architecture will be evaluated against the baseline design using a set of five figures of merit (FOM) that include: Mission Success, Effectiveness, Extensibility, Risk, and Affordability. These will help to characterize the effects of re-fueling on the architecture and will also aid in establishing the propellant delivery cost that will enable to use of propellant re-fueling on future exploration missions.

The work presented in this paper will focus on the development of a simulation environment that has been developed to help investigate a series of architectures trades that utilize propellant re-fueling. The use of this simulation environment helps to increase the size of the trade space that can be considered. It also provides a more efficient means of running Monte Carlo analysis thus allowing design uncertainty to be considered in the analysis.

II. On Orbit Propellant Re-fueling

The concept of in-space propellant re-supply has been around since the 1960's when the air force began investigating how its aircraft re-fueling techniques could be applied to in-space operations[3]. The idea behind propellant re-supply is that a vehicle can gain better overall performance and lower cost if it does not need to carry all of its propellant from the start of the mission. Aircraft have been using this concept since the 1920's [4]. In this case long range cargo aircraft such as the KC-135 are able to re-fuel short ranger fighters. This greatly increasing the range of the fighter aircraft allowing them to perform a wider range of missions. This same concept can be applied to space exploration missions. In this case the propellant required for in-space operations is delivered to LEO or some other destination and stored in an orbiting propellant depot. The propellant can then be transferred to any of the in-space vehicles greatly reducing the payload requirements on the architecture. This however does add additional cost to the architecture.

There have been various concept level studies performed over the 40 – 50 years that investigate the use of propellant re-fueling and its impact on exploration architectures. These studies have generally focused on its implementation in lunar and Mars exploration missions.[(5,6,7)] The use of on-orbit re-fueling has always offered the possibility of greatly improving the payload capability for these missions, but the cost of delivering propellant to LEO has generally been considered to costly to make these architectures economically viable. In much of the previous work little has been focused on how the propellant will be delivered to LEO, and it is generally assumed that this will occur via the same launch vehicle that delivers the crew and cargo to LEO. These vehicles generally have a large launch cost associated with them and therefore would result in a high propellant delivery cost. It is more likely that the use of a low cost commercial launch vehicle would be the only means of achieving a low enough launch cost to make propellant re-fueling a viable addition to exploration architectures. One of the main goals of this work is to determine the required launch price that propellant re-fueling that would need to be achieved.

Below you will find two example cases from the literature where propellant re-fueling has been implemented into exploration architectures. Example #1 uses propellants delivered from the Earth's surface, while Example #2 uses lunar surface propellants.

A. Example #1 – Shuttle-C and ISS[5]

The study performed by Cady suggest the use of the Space Transportation System (STS) and the derivative STS-C to deliver crew and cargo to LEO and the ISS is used as the propellant transfer node. In this case the ISS is expanded to included a series of propellant storage units that are able to store enough propellant to perform the TLI maneuver. This provides for a simpler depot design as some of the ISS sub-systems could be used to operate the depot. A re-usable TLI stage is aslo designed that is capable of docking and receiving propellant from the ISS. The trade off here is whether the cost saving of the re-usable TLI stage is outweighed by the additional cost of delivering propellant. This work also investigated the possible benefits of tank swapping instead for propellant transfer. The presented a case where propellant re-fueling may benefit the design of a lunar architecture. The author however failed to demonstrate what the overall benefit would be for developing and utilizing LEO re-fueling techniques. In fact the author didn't provide any evidence to illustrate that the implementation of on-orbit re-fueling would provide a better architecture design then one without it. It was assumed that the use of propellant re-fueling must improve the design of the lunar architecture.

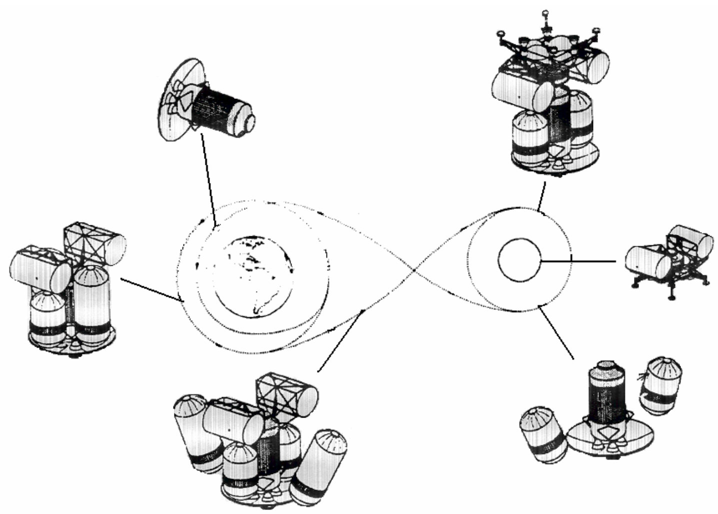

Figure 2. Lunar Architecture Concept of Operations

B. Example #2 – Re-usable Launch Systems and Lunar Surface Propellant[6]

Koelle, in his paper "Lunar Space Transportation System Options" proposed three options where propellant re-fueling could benefit lunar exploration missions, these are outlined in Figure 3. Option one and two utilize lunar surface propellants which are stored in a propellant depot located in LLO and only provide the Earth return propellants. The third option utilizes a Single Stage to Orbit (SSTO) vehicle that docks with a re-supply depot in LEO before continuing on to the lunar surface, it also utilizes propellants obtained from the lunar surface. Earth delivered propellants were not considered in any of these cases. The author provides evidence that the use of a re-fueling can possible provide a greater payload capability at a lower life cycle cost. It is however difficult to separate the benefits provided from re-fueling and those obtained from the different architecture configurations. The author points out that further investigation is required to establish the true benefit that re-fueling would have on the design of a lunar architecture.

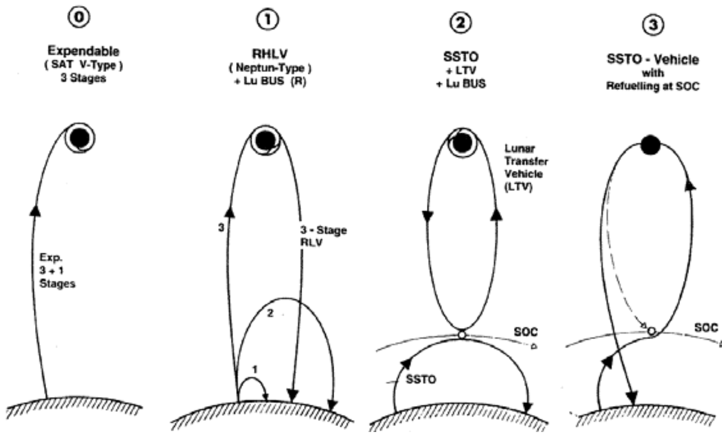

Figure 3: Lunar Transportation System Options

These are two typical examples from the literature on how propellant re-fueling could be implemented into an exploration architecture, in this case both are for a lunar missions. Much of the work that has been done in this area provides an incomplete analysis of the impact of propellant re-fueling and provides little to no evidence as to whether or not propellant re-fueling is the right solution for future exploration. The work in this study hopes to augment the work that has been done to date by applying this concept to a relevant architecture design, increase the design space to include more design choices effected by re-fueling, provide a quantitative measure as to the benefit that re-fueling may provide to the architecture, and determine the price point for propellant delivery. The remaining sections of this paper will outline the design space under consideration, discuss the methods for comparing the different designs and provide an initial look at the results from the architecture simulation.

III. Propellant Re-fueling Architecture Trade Space

The simulation environment was developed to help automate the study of propellant re-fueling and its impact on the lunar architecture. An initial investigation of the problem established nine categories that when combined with propellant re-fueling could greatly impact the design of the architecture. These categories along with the individual design choices are provided in Figure 4. The design space does not include any specific configuration changes to the baseline design, except that the AS engine is allowed to vary between the four choices provided in Figure 4. It is assumed that the majority of the baseline architecture is fixed, and only changes that are specifically altered by re-fueling are considered. This allows for a more valuable comparison to be made between the baseline architecture and various re-fueling trade studies. The simulation can be run for almost any combinations of these variables. While all combinations of the different input variables are not compatible there is still in excess of 10,000 different designs that can be created. This is a very large set of data and would be difficult if not impossible to fully explore if each design had to be calculated manually. The use of an automated process allowed the design space to be greatly increased over what was possible is previous studies. The following is a detailed description of each design variables used in this study.

Re-fueled Element	EDS	LSAM AS	LSAM DS	CLV 2nd Stage	SM
Propellant Transferred	LOX	LH2	LOX/LH2	LOX/CH4	LOX/LH2/CH4
LSAM Ascent Propellant	LOX/LH2 (pump)	LOX/LH2 (pressure)	LOX/CH4	Hypergols	
Boil-off Mitigation	MLI	Cryo-Cooler/MLI	* Considered on EDS, LSAM DS, and LSAM AS		
LOI Burn Element	EDS	LSAM DS	CLV 2nd Stage		
TLI Burn Element	EDS	CLV 2nd Stage			
LEO Orbit Stay	15 days	95 days			
Re-fuel Boil-off Propellant	Yes	No			
Propellant Transfer	LEO Depot	LLO Depot	Direct Transfer		

Figure 4. Trade Study Morphological Matrix

A. Re-fueled Element

There are four possible architecture elements that can be re-fueled. These elements along with the propellant amount considered are provided in Table I. For the EDS the propellant that is re-fueled is either burned during ascent to increase the payload capability of the CaLV or it is removed to reduce the payload requirement on the launch vehicle helping to decrease it overall size. The propellant could also be removed and replaced with additional payload, this would keep the launch vehicles capability the same, but increases the amount of non-propellant payload that can be delivered. Each stage of the lander is considered separately and can either be re-fueled at 0% or 100% for each propellant type. The 2nd stage of the CLV is delivered to LEO with no propellant remaining, it therefore must be re-fueled completely if it is t perform the TLI maneuver.

Table I. Re-fueled Elements and Quantities.

Vehicle	Fuel			Oxidizer		
EDS	0 lbs	15,000 lbs	30,000 lbs	0 lbs	50,000 lbs	100,000 lbs
LSAM AS	0%	100%		0%	100%	
LSAM DS	0%	100%		0%	100%	
2nd Stage CLV	0	~33,000 lbs*		0	~197,000 lbs*	

** Total Required TLI Propellant

B. Propellant Transferred

The architecture utilizes a series of different propellant combinations on each of the individual elements. Only three of these propellants were considered for re-fueling. Though different combinations of these propellants were investigated. The possible propellant combination that were considered are provided in Table II. It was expected that each of these combination would provide a unique understanding as to the effect of re-fueling on the architecture.

Table II. Propellant Combinations Considered.

Propellant Combination	Element Considered
LOX	EDS, AS, DS
LH2	EDS, AS, DS
LOX/LH2	EDS, AS, DS, CLV
LOX/CH4	AS
LOX/LH2/CH4	AS

C. LSAM Ascent Propellant

The LASM ascent engine is the only propulsion system allowed to change during this study. There are currently four propulsion systems being considered for use on the AS. The first is the use of NTO/MMH (Hypergols). These engines have the lowest performance of any of the engines considered, but they would required limited development and have no significant propellant boil-off. The second engine being considered in a LOX/CH4 engine. This engine would require the most advanced development work, but a higher performance can be achieve over that of Hypergols with a much lower boil-off rate then a LOX/LH2 system. This engine could also be more easily adapted to a Mars mission where ISRU may be required. The final two engines are LOX/LH2 systems, both pressure and pump feed engines are considered. The Pump feed engine has a higher overall performance, but is considered a riskier system because of the additional complexity involved. These systems have the highest propellant boil-off because of the high boil-off rate of hydrogen. The propulsion system on the CaLV, CLV and DS are considered to be finalized and will remain fixed to the baseline design.

D. Boil-off Mitigation Methods

Boil-off mitigation is the method used to reduce or eliminate the propellant boil-off from the storage tanks. Two mitigation systems are considered in this study. The first is MLI blankets, in this case the propellant is maintained at a lower temperature by insulating the propellant from the outside environment. This is a relatively simple system that is currently used in space system today. This system only limits the boil-off and can not eliminate it completely. The second system is cryo-coolers/MLI, in this case the cryo-coolers actively remove heat from the tanks keeping the propellant at a constant temperature. A small number of MLI blankets is also use to reduce the heat that the cryo-coolers must remove from the system. This system can be design to eliminate the boil-off completely. In this study it is assumes that the cryo-cooler system achieves ZBO. Cryo-coolers tend to be a better option for longer duration missions where substantial boil-off can occur. The simulation has the option to put either system on the each of the elements that store propellant for use during the mission. Since each elements carries propellant for different time frames each favors the selection of a different mitigation option.

E. Trans-lunar Injection Maneuver

In the baseline architecture the TLI maneuver is performed by the EDS. This was done to utilize the storage capacity and engine already design for the ascent phase of the mission. The upper stage of the CLV also has the propellant storage capacity and the high thrust engine to perform this maneuver. However, in the baseline design this stage does not have the available propellant once reaching LEO. If this stage was re-fueled once in LEO then it has the capability to provide the delta-V for this maneuver. The upper stage of the CLV can transfer in excess of 300,000 lb to LLO if completely re-fueled. If this could be accomplished it would greatly reduce the payload requirement on the CaLV, reducing its size and total cost.

F. Lunar Orbit Insertion Maneuver

In the baseline architecture the LOI maneuver is performed by the DS of the lunar lander. This requires an increased dry weight to be carried down to the lunar surface that is not used during the lunar mission. It would be a more optimal lander design to have the LOI maneuver performed by another vehicle in the architecture. This maneuver could be performed by either the EDS or upper stage of the CLV, and could lead to as much as a 40% reduction in the gross weight of the DS. Improvements in the gross weight of the lander can have a substantial trickle down effect on the rest of the architecture.

G. Re-fuel Propellant Boil-off and LEO Orbital Stay

One of the major concerns with the two launch architecture is that there may be delays that occur that push back the launch of the second vehicle. This can lead to substantial propellant boil-off or the need for large cryo-cooler systems on the EDS. If the time between the launch of the CaLV and CLV exceed the design limits of the EDS the mission may need to be scrapped. This would be a complete loss of the EDS and LSAM a very high cost to pay. The simulation can design the EDS for either a 15 or 95 day stay in LEO. The 95 day requirement puts a heavier burden on the design of the CaLV but provides the highest chance for a successful mission. The 15 day mission design could experience considerable mission delay cost over the life of the campaign. A simulation of the two launch solution shows that only about 75-80% of the missions could launch both vehicles with in the 15 day requirement verses an over 95% success for the 95 day design.

The in-space elements (EDS and LSAM) can be designed to carry only the required mission propellant. In this case the boiled-off propellant would be provided to these elements after the CLV has launched and the architecture is ready for lunar transfer. This would eliminate any loss of mission due to delay in the CLV. This is a relatively small amount of propellant, but would have the potential to save billions of dollars is mission delay cost.

H. On-orbit Re-fueling Location and Method

There are two decisions that need to be made when deciding how the propellant transfer should be handled. Where will the transfer take place and from where is the propellant transferred. In the first case there are two options that are considered here though there are others options that have been discussed in previous studies. The transfer can either take place in LEO or LLO. When using a LLO transfer much of the trade space discussed above is eliminated, thought it could make a much greater use of lunar resources. The second aspect of the design is how the propellant is transferred. There are two options considered here. The first is through the use of a propellant depot which stores the propellant for long periods of time in a orbital storage facility. The second case is the use of a direct transfer where the delivery vehicle directly transfer propellant to each element. The second case requires an additional scheduling requirement as the delivery of propellant must occur on the same schedule as the launch of the lunar architecture elements. There are four possible categories here, but it is likely that direct lunar transfer will be eliminated based on its impractically.

IV. Simulation Environment

The lunar architecture propellant re-fueling simulation environment was developed using the ModelCenter[8] integration frame work developed by Phoenix Integration. ModelCenter is a commercially availiable software that provides a simple to use architecture for passing information between different analysis models. In this case the models have been developed using Excel©, but any software application could have been used. The ability to automatically pass information between the different models greatly reduces the time required to run each simulation. In fact after an initial setup any number of simulations can be run without any human interaction. A illustration of this environment is provided in Figure 5. There are six analysis models and one simulation controller.

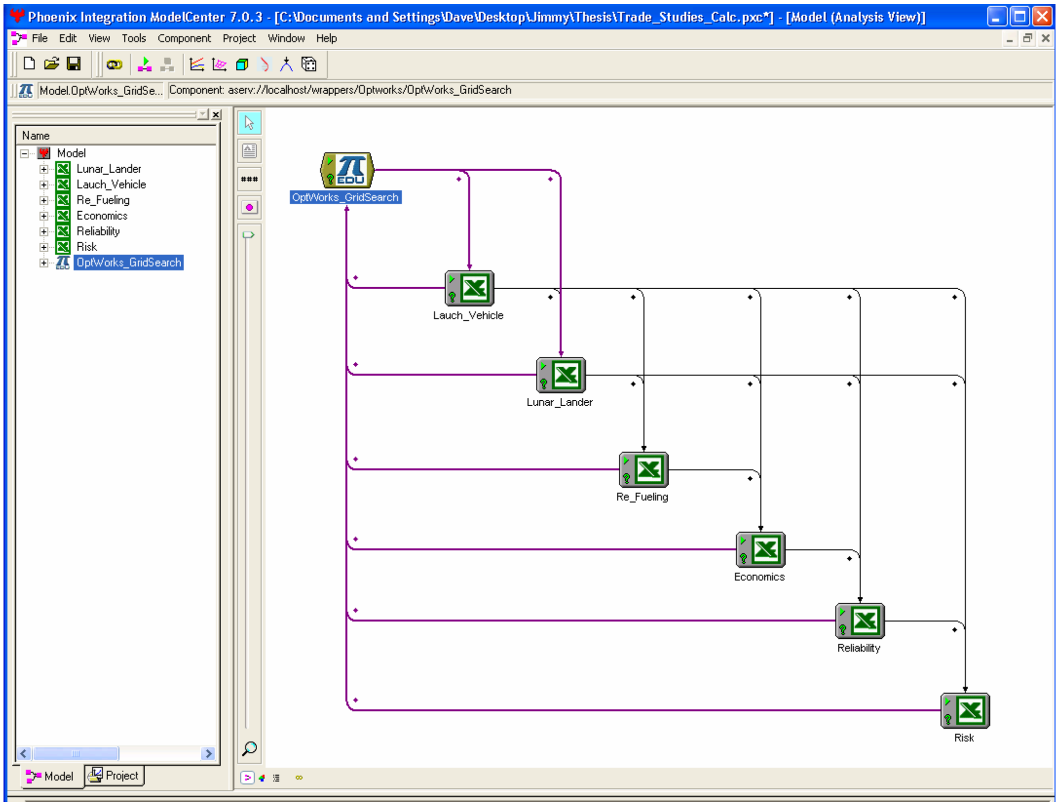

Figure 5. Lunar Architecture Simulation Environment

The simulation runs from the top left to bottom right and passes information between the models signified by a black dot. The passing of information between the Launch_Vehicle and Lunar_Lander can be switched depending on how the architecture is to be design. In one case the launch vehicle is designed to deliver specific payload to LEO and the lander is sized based on this delivered weight. In the second case the lander is sized and the CaLV is design to deliver a lander of this gross weight. The controller in the top left is the link between the design and the simulation. It provides the I/O between the user and the simulation. To familiarize the reader with the details behind each analysis a decryption is provided.

A. Launch_Vehicle

The launch vehicle uses a parametric design to scale the size of the vehicle depending on its configuration and mission. Historical mass estimating relationships (MER) are used to estimate the dry weight of the vehicle. These MERs have been calibrated to provide the same results as provided in the ESAS final report. In this design most of the input design variables are fixed to that of the CaLV limiting the design changes to only those associated with propellant re-fueling. The engine type, number of engines and stage configuration all remain fixed. A validation of this design against the baseline CaLV can be found in Reference 9, the results from this validation show that the dry weight, gross weight, and payload capability are all within 5% of the baseline design.

In order to simplify the design and speed up the simulation the models utilizes a response surface to approximate the changes in configuration, aerodynamics and the trajectory, these are the first three components of the design structure for the model shown in Figure 6. The use of this approximation is part of the validation that was done to match the results for the CaLV, so there is a high level of confidence in the use of this approximation.

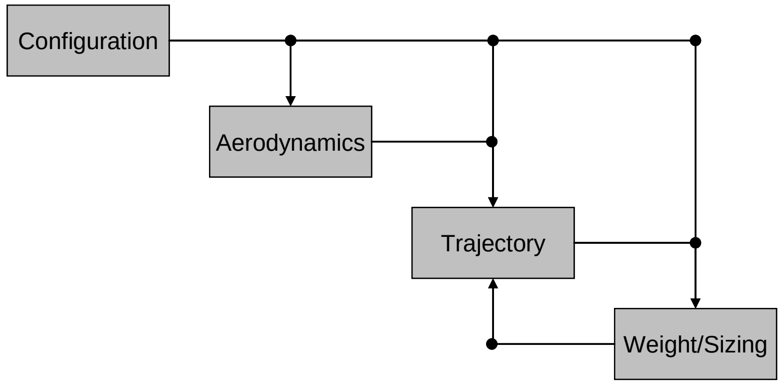

Figure 6. Launch Vehicle Design Structure Matrix

The design of the launch vehicle can be sized by two methods, in both cases the TLI propellant is provided to the launch vehicle once it arrives in LEO. In case #1 the size of the launch vehicle is allowed to decrease as it must only deliver the lander and not the additional 230,000 lbs of TLI propellant. Case #2 increases the burn of the 2^{nd} stage to include the TLI propellant, this increases the overall payload capability of the launch vehicle.

B. Lunar_Lander

The lunar lander also uses a parametric design to scale the size of the vehicle based on the input configuration and mission type. The results from this model were also verified against the baseline ESAS LSAM, and can also be found in Reference 9. There is no aerodynamics model needed for this design and the trajectory is done using simple delta-V approximations taken from the ESAS final report. The model is able to design for various engine configurations and propellant types. The DS engine system remains constant, but the AS engine is allowed to vary between four engine types. The engine is defined by the ISP, thrust, T/W_e, expansion ratio and propellant type. This allows almost any engine to be designed and used on the lander.

Re_Fueling

The re-fueling module estimates the cost of providing propellant to LEO and sizes the propellant storage and transfer systems needed to provide propellant to the individual architecture element. In order to determine the cost of propellant delivery the model runs a simulation that determines the numbers of launches required to provide the total propellant for a given mission. This is dependent on the capability and reliability of the launch vehicle and the propellant required for the mission. The simulation accounts for the fact that a launch failure may occur during any launch and assumes that the cost of a failure is just the loss of that vehicle. The total cost of delivery can then be calculated based on the launch cost and the total number of launches required for the given vehicle. Monte Carlo analysis is also used in this module to account for the uncertainty in the cost and reliability of each launch vehicle. A 90% confidence value is used in determining the propellant delivery cost.

The propellant transfer system mainly focuses on the design of a propellant depot capable of long term propellant storage. The depot is designed to scale the size of the system based on the propellant required for a given mission. The model includes design of the structure and storage tanks, power systems, docking mechanism, maneuvering system and cryo-coolers. The model determines the overall dimensions of the system and the total cost to deploy the depot, more information on this model can be found in Reference 10, several additions have been made to this design since this paper was published, but the basis function has remained the same. This information is then passed to the economics model for use in calculating the total life cycle cost (LCC) of re-fueling the architecture.

C. Economics

The Economics module calculates the entire LCC of the lunar architecture. This includes the development cost of the different architecture elements, the yearly operating cost of each lunar mission, the cost of delivering propellant to the architecture and any cost associated with a loss of missions (LOM). The development and production cost are calculated using historical weight based cost estimating relationships (CERs) obtained form the NASA Air Force Cost Model (NAFCOM) and therefore are a function of the individual architecture elements. Monte Carlo analysis is again used to take into account the high level of uncertainty associated with all economic predictions. Distribution are put on the estimates for development, production, and operating cost. A 90% confidence value is used to represent the total LCC of the lunar campaign.

A secondary aspect of this model is to determine the chance that the mission is delayed or scrubbed because the delay in launching the CLV exceeds the design of the EDS. A plot of the probability of a successful launch based on the designed LEO stay time is provided in Figure 7. The 15 day design was selected to be just before the large drop off see in this figure, but has a significantly lower probability of success than the 95 day design. Monte Carlo is again used to simulate the effect that these probabilities will have on the overall LLC of the campaign.

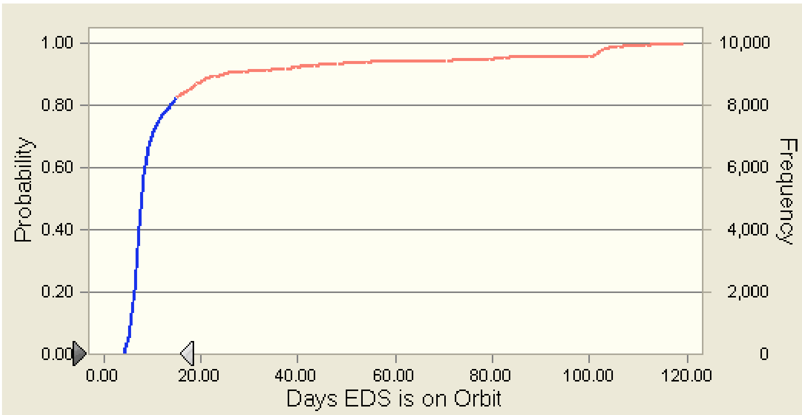

Figure 7. Probability of a Successful Mission Based on 2-Launch Architecture

D. Reliability

The reliability module calculates the overall LOM for the lunar architecture. The model uses fault tree analysis similar to that used to determine the LOM for the Apollo lunar missions. There are six phases of the architecture that are used to determine the overall reliability. These phases are: the launch of the CaLV and CLV to LEO, Earth orbit operations, propellant re-fueling, lunar transfer, lunar orbit operations, and lunar mission. The equation used for this calculation is provided in Equation 1. A dynamically changing reliability model[11] is used to determine the reliability of the CaLV and lunar lander based on the information received from the previous modules. The reliability of the CLV is taken from ESAS final repot and held constant at 1/433. The remaining phases use the data from the Apollo Mission Mode Study[12]. Since it is difficult to determine the actual reliability values Monte Carlo analysis was again used to help eliminate some of the uncertainty in this calculation. A calculation of the reliability is also made where the Apollo reliability assumptions are increased by a factor of 10.

$$LOM = [R_{CaLV} * R_{CLV}] * R_{EOO} * R_{\text{Re}-fueling} * R_{TLI} * R_{LOO} * R_{Lander} \qquad \text{Equ. 1}$$

E. Risk

The risk module provides an estimation of the technological development risk associated with maturing the technologies that are currently not at TRL 9. A list of the technologies that factor into the risk calculation is provided in Table 3 along with the TRL number assumed for each. Any other technology is assumed to effect all designs and such will not effect the relative comparison of the risk score. The risk associated with developing each technology is assumed to be an exponentially increasing function of TRL. The lower the TRL the greater the risk to fully mature the technology. The total risk score for each design is a summation of the individual risk score for each technology.

Table 3. Technology TRL Levels

Technology	TRL	Risk Score
Ascent Engine		
Hypergol	7	5.04
LOX/CH4	5	2.32
LOX/LH2 (pump)	8	7.15
LOX/LH2 (pressure)	8	7.15
Mitigation Method		
MLI	8	7.15
Cryo-coolers	4	1.46
Re-fueling	4	1.46

The simulation takes approximately 6 seconds to run, most of which is required to run the 10,000 Monte Carlo calculations.

V. Sample Results

This section will outline two samples cases that use this simulation environment to investigate how propellant re-fueling effect the design of the lunar architecture. The first sample case increases the propellant that is burned on the upper stage of the CaLV and the second case removes the LOX from the EDS and allows the CaLV to adjust its over all size. These two cases should provide the reader with a better understanding of the results and information that can be achieved through this simulation.

A. Sample Case #1

For this example a set of 80 design cases were run using the four input variables provided in Table IV. The simulation required approximately eight minutes to run.

Table IV. Sample Case #1. Increase Upper Stage Propellant Burned Inputs

Variable	Inputs				
AS Mitigation	MLI	Cryo-cooler			
Ds Mitigation	MLI	Cryo-cooler			
Additional Propellant Burned	0 lb	25,000 lb	50,000 lb	75,000 lb	100,000 lb
Lander Stage Re-fueled	None	DS	AS	AS and Ds	

The results of this simulation are provided in Figure 8. Taking a look at the set of blue dots first, which represent the total LCC for the lunar architecture. There are three trends that can be seen in this data. The first is the local set of four data point that in general form a horizontal line with little variation in total LCC. These points represent the four combinations of mitigation methods that can be used on the ascent and descent stages of the lander. Note that they tend to have little effect on the overall cost of the architecture. The second trend to note is the five sets of these four points that increase in LCC. These points represent an increase in the amount of propellant that is re-fuelled in the architecture. It is not surprising that the cost of the architecture increases, the goal of these cases is to improve the capability of the architecture not lower the cost of the campaign. The final trend is the results for the different re-fueling strategist on the lander. Due to the additional propellant delivery requirements the more re-fueling that is required the greater the cost of the architecture.

The average cost of delivering propellant to the lunar surface shows similar trends as that seen in the LCC data. The results however tend to decrease with an increase in propellant delivery. This suggest that the architecture's ability to deliver payload to the lunar surface become more efficient with an increase in propellant delivery. There is also a lower limit to this efficiency that can be achieved no mater how much propellant is delivered to the architecture. This is due to the fact that there is a certain amount of infrastructure that must be developed in order to accomplish the lunar mission. The limit on the $/lb to the lunar surface is approximately $5,000/lb of payload. Due to this limitation there is little improvement that can be achieved through propellant re-fueling on the lunar lander, unless a greater payload capability is all that is desired. Again the mitigation method selected had little impact on the lunar surface payload efficiency.

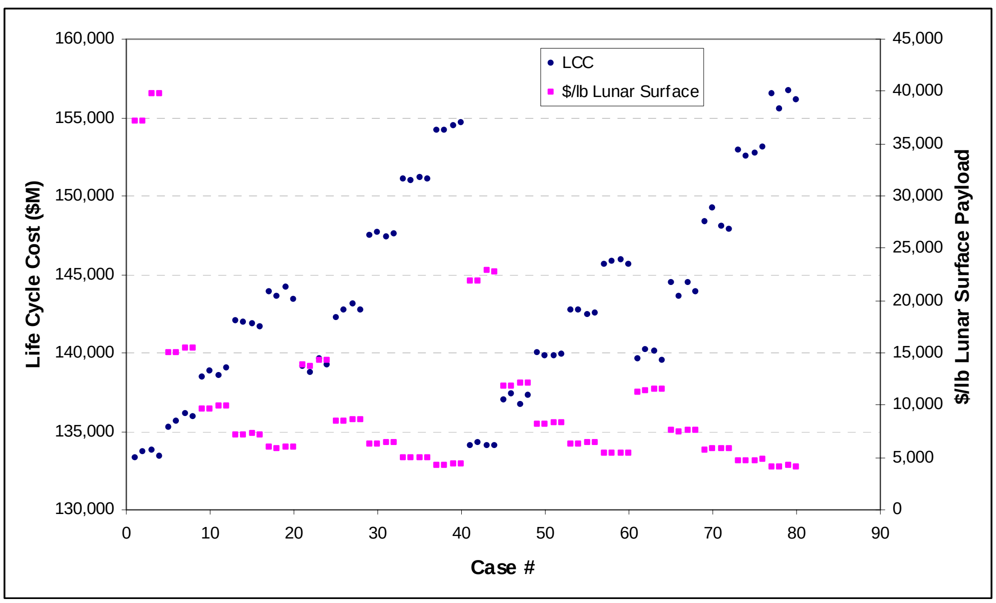

Figure 8. Sample Case #1. Increase Upper Stage Propellant Burned

B. Sample Case #2

For this example a set of 40 design cases were run using the 3 input variables provided in Table V. The simulation required approximately 4 min to run. The LOI maneuver refers to what stage performs this maneuver, in this simulation only the EDS and DS were considered.

Table V. Sample Case #2. Allow CaLV to Decrease in Size Inputs

Variable	Inputs				
AS Engine	Hypergol	LOX/CH4	LOX/LH2 (pressure)	LOX/LH2 (pump)	
LOX TLI Propellant Removed	0 lb	25,000 lb	50,000 lb	75,000 lb	100,000 lb
LOI Maneuver	DS	EDS			

There again is some useful information that can be taken from the results of this simulation that help to characterize how propellant re-fueling effect the design of the lunar architecture. The two over all trend lines seen in Figure 9 suggest that if the LOI maneuver is performed by the EDS that a lower life cycle cost can be achieved. In this case the additional LOX required for this maneuver is provided through propellant re-fueling once the EDS reaches LEO. This reduction in LLC is due to the decrease in size of the lander that is achieved by reducing the total propellant that that must be carried on the lander. Reducing the size of the lander has a drastic effect on the size of the CaLV which is one of the most costly elements of the architecture.

Increasing the amount of propellant that is provided to the architecture has little effect on the over all cost. When the TLI propellant is removed there is a decrease in the development and production cost of the architecture, but this is countered by the additional cost of providing the propellant to LEO. The major improvement here comes from allowing the EDS to perform the LOI maneuver and not from decreasing the amount of TLI propellant that is carried on the CaLV. These results show that the LOX/LH2 pressure feed engine has the highest LLC of all foru engines considered.

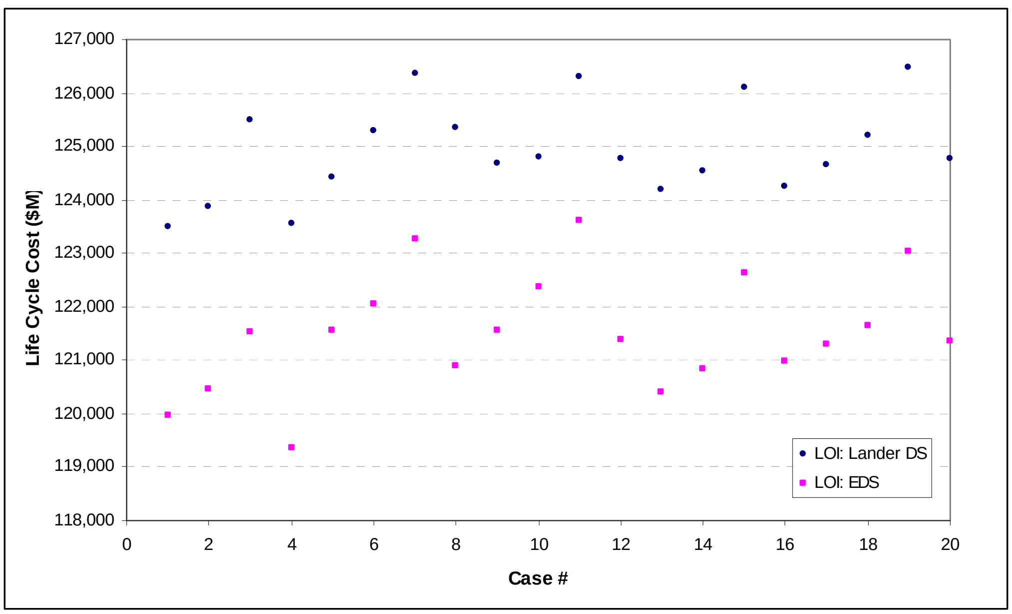

Figure 9. Sample Case #2. Allow CaLV to Decrease in Size

The results from these two studies show that an improvement in the architectures efficiency and payload capability can be achieved by increasing the amount of propellant burned during the ascent, but comes at an increase in the overall cost of the architecture. The cost of the architecture can be reduced by allowing the EDS stage to perform the LOI maneuver and only a small improvement is seen when the TLI propellant is removed from the CaLV and the vehicle is allowed to decrease in size.

VI. Conclusion

This paper described the development and implementation of a lunar architecture simulation environment capable of evaluating the use of propellant re-fueling. A discussion of the work done to date applying propellant re-fueling to lunar architecture, and the short comings of this work was addressed. A trade space was then presented that would fully investigate how this capability could be applied to the current lunar architecture and determine if such a capability would improve the overall design. The use of the simulation is required if the entire trade space is to be investigated within a reusable time frame. Using the simulation the entire trade space can be evaluated within a single day. If these cases were done manually it would take weeks or even months to fully evaluate all the combinations of interest. Two example studies were provided to describe the outputs of the simulation and provide an initial look at the results of implementing this capability into the architecture. It was shown that both a lower cost and/or greater payload capability could be accomplished through the use of on-orbit propellant re-fueling technologies.

A follow on paper will be presented at the 2007 International Astronautical Congress that will describe in much greater detail the results obtained from the entire trade space. It will also discuss the required cost of re-fueling that must be achieved in order to make this capability a viable option for future exploration missions. This is a key aspect of this work and will be the main focus of the follow on paper.

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