The Modeling and Evaluation of Interplanetary Manned Missions Using System Architecting Techniques

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Abstract-- **The manned exploration of the rocky bodies of the solar system has long been one of the primary, if unstated, goals of the aerospace industry. In order to facilitate the design and decision process associated with the development of the systems necessary for such complex operations, an understanding of the system on the highest level must be established. This paper describes the development of a systems architecture-level model used to evaluate the in-space transportation portion of manned exploration missions to other rocky bodies in the solar system. The results of this analysis, including the "best" architectures given the metrics employed as well as technology decision impacts, will also be discussed. The MATLAB-based model utilizes the concepts of decision formulation, set partitioning problems, and technology option analysis. The set of architectures is decoupled from the launch vehicle architecture as well as destination exploration operations, allowing for an iso-performance analysis with science and exploration value while remaining independent of the launch vehicle limitations.**

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ACRONYMS

ADG – Architectural Decision Graph
AOS – Algebra of Systems
CER – Cost Estimating Relationship
DRA – Design Reference Architecture
EM – Earth-Moon
HAT – Human Spaceflight Architecture Team
HEOMD – NASA Human Exploration and Operations Mission Directorate
IMLEO – Initial Mass in Low-Earth Orbit
ISECG – International Space Exploration Coordination Group
ISRU – *In-Situ* Resource Utilization
ISS – International Space Station
JPL – Jet Propulsion Laboratory
LCC – Lifecycle Cost
LOX/LCH4 – Liquid Oxygen / Liquid Methane Propellant
LOX/LH2 – Liquid Oxygen / Liquid Hydrogen Propellant
MIT – Massachusetts Institute of Technology
MPCV – Multi-Purpose Crew Vehicle
NASA – National Aeronautics and Space Administration
NEA – Near Earth Asteroid
NP – Non-Polynomial
NTR – Nuclear Thermal Rocket
SAP – Systems Architecting Problem
SEP – Solar Electric Propulsion
SLS – Space Launch System
TRL – Technology Readiness Level

1. INTRODUCTION

With the end of the Space Shuttle era and the creation of a new Space Launch System (SLS), NASA and its global counterparts have an opportunity to renew studies of manned missions beyond the scope of the Earth-Moon system. NASA teams such as the Human Spaceflight Architecture Team (HAT) have been tasked with developing paper studies of these missions. International groups such as the International Space Exploration Coordination Group (ISECG) have similarly begun to collaborate on international studies and partnerships toward this same end [1]. Previous studies, both domestic and international, such as the Jet Propulsion Lab's (JPL) Austere Mission [2] and the Australian Mars-Oz project [3], have gone through many iterations for determining the "best" method for manned exploration. Extensive studies are necessary, as these missions would require many years for hardware and technology development as well as many billions of dollars both for development and operations. With the development of the SLS, the infrastructure for the launch and operation of complex space systems is being established. As exploration systems become closer to physical development, the understanding of the drivers of performance and cost for such systems becomes critical.

In order to facilitate high-fidelity point design studies and inform future decision-makers, the key drivers of system-level needs should be identified and quantified prior to concept down-selection. Systems architecting techniques

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allow for the analysis of the system tradespace while retaining the necessary system complexity to differentiate between good alternatives. These methods allow analysis of the broader perspective while retaining enough fidelity to distinguish between architectures. Unlike traditional point designs, systems architecture analysis allows engineers to more rigorously quantify architecture-level trades and assess the impact of design decisions within the space of possible architectures. Recent research conducted at M.I.T. has developed a set of tools for quantitative analysis of systems architectures, such as the Architectural Decision Graph (ADG) Framework [4] and the Algebra of Systems [5]. Building on these concepts, quantitative analysis of human exploration infrastructures was performed in order to inform future high-fidelity point designs.

We present a unique system model for the analysis of the in-space infrastructure for manned exploration missions. The modeling approach focuses on three components. The first component is comprised of a group of mission mode parameters that set the scientific value of the various exploration infrastructures. Second, the physical habitat and propulsion elements are established by allocating a set of fundamental invariant functions to hardware components. Lastly, a suite of technology options accompanies these system parameters, allowing for the analysis of technology impacts on the system. Although system architecture techniques have been applied to this type of system in the past, they have been limited by assumptions about the habitation and transportation tradespace. This also differs from general model-based engineering by exploring more alternatives in a lower-fidelity space. Model-based engineering is used internally in order to analyze architectures, but the focus is on high-level, architecturally distinguishing features rather than modeling of the complete system in detail. This also requires that the models are formulated for efficient analysis of many architectures, rather than efficient analysis of many system details using extensive domain knowledge. Effective exploration of this space is accomplished by utilizing vetted techniques on a unique decomposition of the manned exploration system.

The scoping of this study in the broader context is first addressed, followed by a determination of the architecturally distinguishing features present in manned exploration infrastructures. The current model's development from previous models is also discussed, along with the past and present capabilities and limitations. Validation cases are presented, in detail for Mars and briefly for the Moon and Near Earth Asteroids (NEAs). For each destination, the Pareto front of optimal architectures will be examined, along with a series of technology "switches" to understand impacts on the tradespace of specific technology options. The further impact of NASA's investment in the MPCV and SLS programs will be analyzed in terms of the reduction of the tradespace. This will be followed by a summary of the paper and final conclusions.

Specifically, the paper is structured as follows. Section 2 will discuss the study formulation, including scoping and problem formulation, along with validation cases. Section 3 will address the above mentioned results analysis, while section 5 includes the summary and conclusion. Acknowledgements, references, and biographies follow.

2. STUDY FORMULATION

Beyond the Point Design

Traditional analysis of complex space systems, particularly manned exploration systems, has been accomplished through the use of highly detailed point design studies. These studies have been conducted by teams of highly skilled and experienced individuals, often requiring months or years to accomplish. Although a great deal of unpublished background research is performed prior to these design studies, without detailed analysis of many possible systems, these types of analysis may fail to capture the best possible system-level design. The interactions between the components of these complex systems result in emergent properties that are not always readily predictable. This means that some systems with particular combinations of elements may have unexpected properties or behaviors not readily predictable prior to modeling. Therefore, without modeling the variety of system element combinations, engineers cannot be certain that the chosen set of elements for detailed point designs have the best performance characteristics. This is especially prevalent in systems without significant heritage and/or highly complex interactions between system elements. Both of these characteristics are present in large-scale human exploration missions.

To circumvent this problem and feed information to engineers about elements of point designs to be considered, architecture-level systems models are integrated into the early-stage design process. These models decompose the system of interest into architecturally distinguishing features, allowing analysis of the various combinations of these features without the need for detailed models. Due to the reduced fidelity of architecture-level models, the results are not taken to be exact, but the features are selected such that the impact of changing one or more feature results in a correct ordinal ranking along the metric or metrics of interest. In other words, the ranking of architectures resulting from the analysis remains correct regardless of the lack of detail in the model, if the simplifying assumptions posed by the model are valid. Feeding these top choices from a lower fidelity study to one of higher fidelity builds confidence that the point design reflects the best architecture in the broader tradespace.

Scoping

Downscoping of the Tradespace – As a description of the full tradespace of human exploration systems infrastructures is far too broad, downscoping of the problem at hand is necessary in order to create a comprehensible set of analysis. For this model, the overall tradespace was

downscoped by decoupling the system from parts of the mission infrastructure. The final designs for the future U.S. launch system are being established, and so an analysis of the space including a launch system is both unnecessary and overly complex. Combined with the knowledge that many case studies focus on surface operations, where the greatest design uncertainties are present, along with the fact that science payload to the surface and surface operations set the scientific value of a mission, we determined that it was most appropriate to model the in-space transportation infrastructure of the system. This allows for a deeper understanding of a portion of the system that drives both material and development costs as well as launch capability needs.

Destination Selection – NASA and its international counterparts have debated the set of destinations appropriate for human exploration missions for many years, culminating in deliberations revolving around the concept of the "Flexible Path" [1][6] within the last ten years. The three destinations described in such a path, regardless of the order in which they are explored or not explored, are generally the Moon, Mars, and Near Earth Asteroids (NEAs). In theory, these destinations allow for incremental technology development, building to systems for more and more difficult destinations to reach. NASA HAT has taken it upon itself to explore detailed point designs for missions to these destinations. The model reflects these destination choices.

As there are many asteroids with a wide range of characteristics that may be classified as NEAs, two asteroids that represent the widest breadth of possibilities, one "High Energy NEA" and one "Low Energy NEA", the difference being the energy needed to reach the destination from Earth from as astrodynamics point of view, were included. Although these in theory represent a set of possible destinations, the specific numerical values used to model these generic destinations come from specific asteroids also being studied by NASA HAT, namely 2000SG344 and 2008EV5 [7].

The "Flexible Path" often also includes other "non-solid" destinations, usually Earth-Moon or Earth-Sun Lagrange points. As the scientific value of these destinations is still hotly debated, as well as the fact that these are used almost exclusively as mid-points in larger, multi-mission exploration schemes, these destinations were excluded from the tradespace as final destinations. However, they were included as possible rendezvous points, meaning that exploration elements could be pre-deployed to these locations and "picked up" on the way to a further final destination.

In summary, this study includes the evaluation of round-trip missions to the Moon, Mars, and a set of two representative NEAs. Each of these missions may also use one of several rendezvous points, including the most favorable Earth-Moon and Earth-Sun Lagrange points [8].

Further Assumptions—For the purposes of limiting the tradespace and reducing computational complexity, further assumptions regarding the nature of the systems in question were necessary. These include both broader scoping assumptions as well as restrictive technical assumptions. Broad scoping assumptions reduce the overall design space and complexity by simplifying the system being modeled. Technical assumptions reduce the complexity of interactions between elements or simply reduce the number of possibilities for technology implementation.

For computational simplicity and in line with previous architectural studies, the analysis was limited to sortie-like missions only. This means that each mission is designed to deliver crew and payload to the surface of a rocky body, allow a stay on that surface for a given amount of time, and return the crew to Earth. This eliminates two other possible modes. The broader campaign of missions, where multiple crews explore a given destination or set of destinations, is not considered. Elements within an architecture are assumed to be designed for the single mission only, and no effort is given to leaving any system elements for future missions at any point. All elements are discarded after their functional requirements are fulfilled. The second possible mission mode this also eliminates is the highly controversial one-way mission, where astronauts are sent on a mission to a destination but are explicitly not returned.

Technical assumptions limited the elements of the various architectures to those that have already been developed or have development history. Technical elements were also limited to those that have sufficient data to be modeled accurately. System elements with absolutely no development history or insufficient available data were not considered. For example, LOX/LCH_4 propulsion elements were included in the system while solar sails were not. This is due to the testing of large-scale methane engines, such as the XCOR XR-5M15, while solar sails are still conceptual [9].

Evolution of the Architecture Formulation

System Architecting Problems (SAP) and the Assignment Problem – In Dr. Selva's Ph.D. work [10], a set of System Architecting Problems is proposed. These cover the majority of different formulations for the modeling of complex systems at low fidelity. Prior modeling of manned exploration missions at the architectural level have focused on the "Generalized Assignment Problem". This is accomplished by breaking down a system into a series of decisions, each decision corresponding to a particular architecturally distinguishing feature. The choices available for each decision represent the different possibilities for the distinguishing features. For example, a decision for a Mars architecture may be "How will the crewed stack circularize the orbit around Mars?" Possible choices might include: 1) propulsive breaking, 2) aerocapture, or 3) a combination of propulsive breaking and aerocapture. Each decision may

have one and only one choice selected, and these decisions and choices aggregate to define an architecture. The general assignment problem has been mathematically shown to be in the category of non-polynomial-complete (NP-complete) problems [11]. Therefore, in theory, any formulation of an architecture modeled in any NP-hard method can be formulated as an assignment problem [12]. This allows for generality in properties across many architecture-level formulations. Furthermore, the use of decisions and choices matches well with the needs of real-world decision-makers. Decision-makers must choose among the various good options in much the same way as an architecture is formulated using the assignment problem. Not only does this match well from a modeling standpoint, but this also increases acceptance of the model formulation and results by the recipients of the data.

Past Models, Their Capabilities, and Their Limitations – Hofsteter formulated manned exploration systems, specifically lunar missions, in the arrangement of an assignment problem in his master's thesis [13]. His approach focused on the "events" occurring at five nodes in the system, being most concerned with the movement of the crew from element to element. In total, Hofstetter used seven decisions to define an architecture, allowing for the full enumeration and analysis of approximately 30 architectures. Not all combinations of decisions and choices were possible, as some tightly coupled decisions were incompatible in certain combinations. For example, the choice of three crew transfers without choosing a set of three transfers in the five nodes is both illogical and impossible. These additional constraints, which required hard encoding of logical impossibilities, were acceptable due to the small number of possible architectural combinations with the set of seven decisions. However, this limited formulation prevented the analysis of anything beyond basic groupings of habitat elements. This system formulation was adequate to evaluate a set of basic habitat groupings, but any architectural features beyond that cannot be modeled by the system.

Extending that concept, Simmons developed the Architectural Decision Graph (ADG) Framework [4]. This framework allows the architect to formulate complex systems in the form of an assignment problem, allowing both the modeling of the system architecture and the evaluation of that system in the same structure. Simmons went on to demonstrate the use of this framework by formulating the Apollo Program. Once again only key groupings of habitats and propulsion stages are considered, although this reflects the discussion during the Apollo era more closely than a full set of all possible habitat and propulsion element combinations. Simmons went a step further than Hofstetter by including propellant technology options for the major stages as well as options for the number of crew. Simmons' formulation included nine distinct decisions, most pertaining to the formulation of the propulsion and habitat elements. This formulation encoded more of the constraints by cleaving the decisions along more natural lines, meaning that multiple decisions did not clearly pertain to the same logical constraint. This was not fully generalized, however, and thus still required additional post-formulation constraints.

These studies lack several major aspects that allow for the broader exploration of the manned mission tradespace. First and foremost, they each make the assumption that only a small set of possible transportation and habitation strategies can be used. The Apollo Program has been said to have used over 1,000,000 man-hours in the decision to employ the Lunar-orbit rendezvous strategy [14]. While this may have worked well for Apollo, this strategy requires resources that many missions will not have available. As such, it is important to have a consistent way to analyze the many different transportation strategies possible for these extended systems without huge resource expenditure. Habitation strategies, meaning the choice of how many habitats should accomplish the various necessary functions and which functions belong to which habitat, require a similar exploration across the design tradespce.

Within these design tradespaces, an important concept that has not gotten sufficient treatment in the past is the concept of re-use. More specifically, this is the re-use of a system element for a second or greater function as temporally displaced from the first function but still within the same mission. An example would be a propulsive stage that is used for Earth departure, sits in Mars orbit, and then is re-used for the return to Earth. The re-use requirements of all physical elements are critical to the design of the system. A method for formulating all propulsive stage and habitat element allocations allows for any kind of re-use to be present in the modeled system. Therefore, by not limiting to a small set of assumed allocations, full re-use analysis can be performed.

System Model Formulation: Capabilities and Limitations

Results from an architecture level study are highly dependent on the method of decomposition of the system of interest. As such, it is critical to determine *a priori* the key drivers of the system level metrics of interest. The drivers for the metrics of interest determine the architecturally distinguishing features of a system, therefore driving the method of decomposition. This section describes the decomposition approach, key architecture-level parameters, and the metrics upon which each architecture was evaluated.

Initial Mass in Low-Earth Orbit (IMLEO) – Mass drives cost. This has been well established in the aerospace community over the past decades [15]. Mass on the destination surface drives the mass of the EDL system, which drives the mass of the in-space system, which drives the launch cost from Earth's surface. Although mass clearly does not encompass all of the aspects of system lifecycle cost, it is a primary component of it. An IMLEO metric

perpetuates the concept of decoupling from the launch infrastructure while measuring an established cost driver. IMLEO is calculated using the basic formulation in Equation 1.

$$IMLEO = m_{pl} + m_{crew} + m_{log} + m_{prop} + m_{hab} \quad (1)$$

where m_{pl} is the payload mass
m_{crew} is the crew mass
m_{log} is the logistics mass
m_{prop} is the propulsion stages' mass
m_{hab} is the total mass of all habitat elements

Lifecycle Cost Proxy – Although IMLEO has been well-established as an indicator of launch costs, it does not capture all aspects of the cost of mission infrastructures. Introduced in Mr. Battat's masters thesis [16], the Lifecycle Cost (LCC) proxy attempts to account for the technology portfolio lifecycle cost that must be fulfilled for a given architecture. It accounts for both the development and operating costs and does not rely on Cost Estimating Relationships (CER). Fundamentally, this metric is driven by two cost factors: the readiness level of a technology, which influences the development cost, and the demand for that technology, which influences the procurement cost. While this metric does not estimate absolute cost, it does create an ordinal ranking of the architectures considering the relative investment of resources for the development and operation of the technology package embedded. The metric output follows Equation 2.

$$LCC = \sum_i C_i T_i \quad (2)$$

where LCC is lifecycle cost proxy
C_i is the cost coefficient (see below)
T_i is the technology presence coefficient
i is the index of each possible technology

In this case, the technology coefficient is simply 1 if the technology is present in the architecture and 0 if it is not. The cost coefficient is based on the readiness level of the technology and the potential for other users of that technology, according to ale, having only three levels.

Table 1. Note that the readiness level is an extremely simplified version of the NASA TRL scale, having only three levels.

Table 1 – Cost Coefficient Information

	Technology has other users?	
	NO	**YES**
Low Readiness	1.000	0.500
Relevant Demonstration	0.667	0.333
Existing Capability	0.333	0.167

Functional Decomposition Approach – The most natural approach to decompose a technical system is to cleave upon obvious physical system elements. This is most directly reflected in traditional systems engineering, where design teams are broken into the primary subsystems of a larger, more complex technical system. In the case of interplanetary human exploration systems, there is no real system heritage and only limited design reference heritage. This drives a more generalized functional decomposition, looking at what fundamental functions the system must perform. Typically, these functions are then matched with possible combinations of physical system elements that perform these various functions, referred to as functional allocations, ranging from multiple elements performing the same function (redundancy) to single elements performing multiple functions.

After many cycles of analysis on the decomposition of the system, we found that transportation systems of this level only have two fundamental functions that must be performed: 1) the "habitation" function, which provides the life support as well as living space for the astronauts, and 2) the "transportation" function, which is the physical movement of the people and cargo to the destination. However, these fundamental functions are not the only system aspects that drive the key metrics. That is to say that the metrics may be influenced by system properties that are independent of how the fundamental functions are allocated to physical elements.

Architectural Variables, Functional Invariance, and the Set Partitioning Problem -- As described, the in-space transportation infrastructure performs a set of two primary functions. Both of these functions must always be performed, otherwise the system cannot successfully operate. This concept of functional invariance is critical to the successful modeling of this system. As all architectures are meant to accomplish the same goals with the same scientific value, the functions performed within the system must likewise be constant across all architectures. The in-space portion of the transportation infrastructure must fundamentally accomplish the functions of transporting the astronauts and providing habitation for those astronauts. Although this seems like a trivial statement, these fundamental functions are difficult to establish. For one, engineers tend to focus on physical aspects of a system rather than functional aspects, making disassociation from physical forms non-intuitive. Such complex systems are also perceived as being exactly that: complex. Driving down into the fundamental nature of these complex systems means working beyond the daunting complexity.

These primary functions are further broken down into sets of seven and ten functions, respectively, which must also always be performed. The seven "habitation" functions include: 1) Earth launch (which may be accomplished by an element in the in-space transportation infrastructure), 2) deep space outbound habitation, 3) descent habitation, 4) surface habitation, 5) ascent habitation, 6) deep space inbound habitation, and 7) Earth re-entry. It is possible to

imagine that these may be further broken into a continuum of infinite possible functions, but these segments comprise the set of logical functional subdivisions of the space.

This also coincides with the common physical separations, as each function requires different performance characteristics from physical systems. This reinforces the concept that these cleavage points are correct, as they correspond to established systems and models. Similarly, the ten "transportation" functions include: 1) Earth departure, 2) outbound staging location breaking, 3) outbound staging location departure, 4) destination orbit arrival breaking, 5) descent, 6) ascent, 7) destination orbit departure, 8) inbound staging location breaking, 9) inbound staging location departure, and 10) Earth orbit breaking. No Earth surface launch segment is included due to the de-coupling from the launch infrastructure. No Earth entry transportation leg is included as it is assumed that aerobreaking in the atmosphere is always used. This last assumption indicates that any habitat element used for Earth entry must have an associated aeroshield. This leads to a trade between the desire for a single habitat to reduce additional dry mass and the need for a large aeroshield. Unlike the habitation functions that truly must always exist, it is clear that without staging locations for a given set of architectures, there is no need for transportation functions 2, 3, 8, and 9. These "points" must always be passed, in a sense, within the mission, but the cost of going through them becomes zero when there is no need to stop at any staging location. A pictorial representation of these functions in the context of a Mars mission, which is the most comprehensive mission mode, can be seen in Figure 1.

The physical elements shown on this chart are not meant to restrict the possible forms of the various habitat elements. Rather, they are meant to reinforce the concept of the habitation functions. This method of segmenting the space is reinforced by the fact that these elements of form are normally conceptualized in this fashion.

Each of the two primary functions is composed of a set of sub-functions that must always be present and must always be accomplished once. This clearly fits the criteria for a set partitioning problem [17], which, like the assignment problem, is one of the many NP-complete problems [18]. As the formulation and calculation of these problems is well known and well-studied, it is advantageous to encode this information as two set partitioning problems. It should be noted that this causes a distinct separation from the traditional assignment problem, as the model is now composed of a set of decisions and choices and two assignments to one of many set partitions schemes. After eliminating logical inconsistencies, each of these problems has a selection group of 120 and 776 options, respectively. Unlike many other problems formulated under an assignment problem decomposition, the use of set partitions with many alternative schemes creates a challenge to the engineer as well as the various outside stakeholders regarding the intelligibility of the system. However, this is more intelligible than the traditional assignment problem formulation, which would require a set of 10!+7! binary decisions (yes/no) regarding the matching of functions to formal elements in the worst case.

Along with these two set partitioning problems, a third concept of pre-deployment creates the group of so-called

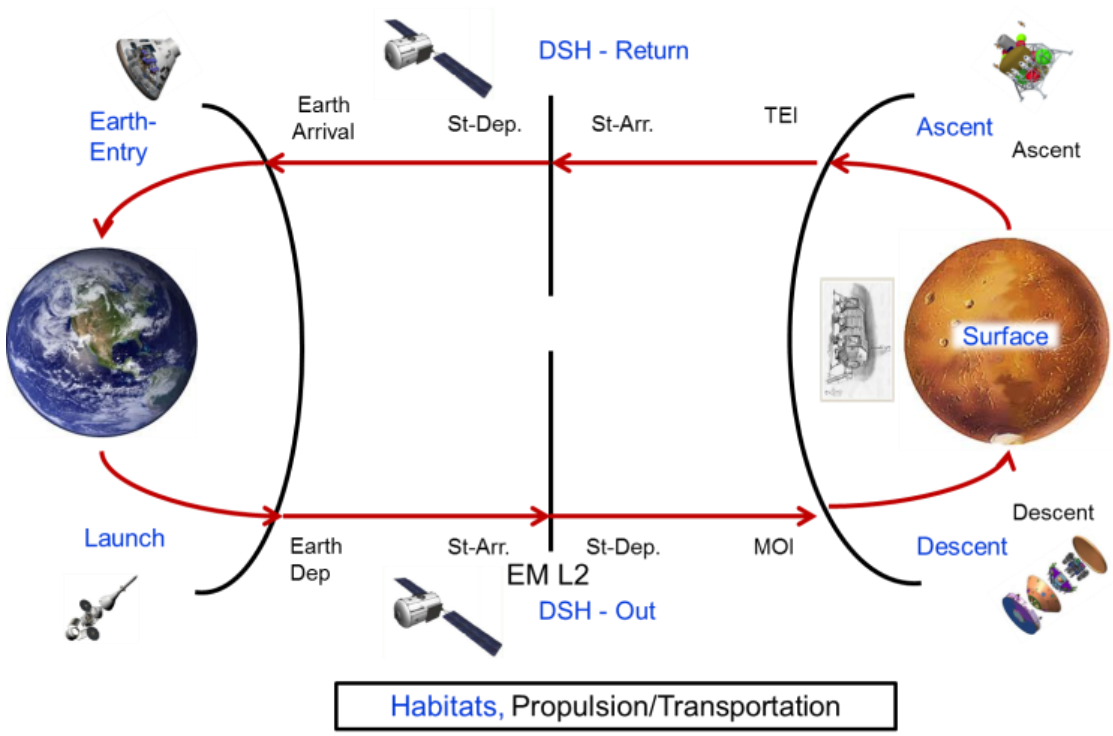

Figure 1 -- Habitat and Transportation Functions in a Mars Mission Context

architectural features for a given architecture. This name is given because these three decisions determine the majority of the large physical elements of the system, thus setting much of the hardware development cost as well as the majority of the system extensibility capabilities. In this case, pre-deployment describes the use of a low thrust system to pre-deploy cargo to the destination surface or orbit. This decision has been simplified to a binary switch (yes/no). "Yes" means that every element that can be logically pre-deployed to the destination surface or orbit will be, such as propulsion stages used after arrival at the destination orbit by the crewed stage that are not required prior to that stage. For instance, if a system consisted of three propulsion stages, being an Earth departure stage (EDS), a descent and ascent stage, and an Earth return stage, the second and third stages could be pre-deployed while the EDS could not. However, if the system was only two propulsion stages, where the EDS and descent and ascent stage were combined, only the Earth return stage could be pre-deployed.

In this system decomposition, any element pre-deployed from LEO would use a solar electric propulsion (SEP) stage, which is the low thrust system of interest to NASA [7]. Traditional cryogenic and nuclear propulsion stages that are typically used to pre-deploy elements in many of the reference designs were not considered for the pre-deployed elements. This is due to the fact that the rocket equation is linear with payload [19]. This means that, from a mass standpoint, having all the mass on one stack versus the mass being on two different stacks makes no difference with regard to propellant mass. This only works under the assumption that propulsion elements have a constant dry mass ratio. Although reality differs slightly from this case, it is not worth the additional computational complexity to model this relatively minor variation. Given that the same traditional propellant is used for both the crewed stack and the pre-deployment stack, in the metrics of interest, the architectures having pre-deployment or not having pre-deployment are identical. Low-thrust propellants, on the other hand, have very different energetic requirements and thus make a significant difference in the metrics of interest. The use of such a system also significantly impacts the development of system elements, and so it is grouped with the set partitioning problems as a member of the architectural decisions.

Advanced Technology Options – The third and final set of system decisions that define a given architecture are the advanced technology options. These fall into two groups: 1) decisions between technology options where one must always be chosen, and 2) decisions about whether or not to include certain advanced technologies. The propellant choices fall into the first category, and it was decided based on the current interests of NASA to include LOX/LH_2 and LOX/CH_4 as options for all stages, as well as nuclear thermal rockets (NTR) for the in-space stages and hypergolic propellants for the descent and ascent stages. Hypergolic propellants are too low performance to be reasonable for the in-space stages, while NTR options are unreasonable for descent and ascent due to the large dry mass necessary as well as the continuous thrust characteristics of these systems.

The set of binary decisions included in the study cover several technologies that have large impacts at an architectural level. Aerocapture for Mars missions allows the trade between a large propulsion stage for breaking into Mars orbit and a large dry mass for using atmospheric drag. *In-situ* resource utilization (ISRU) similarly trades the need for carrying large amounts of ascent propellants from Earth for the dry mass necessary to extract them from the destination surface and possibly atmosphere. Boil-off control, which limits the amount of propellant burn-off due to heating and containment issues, trades larger amounts of propellant for extra dry mass. In this case, the various boil-off control capabilities are not individually analyzed. Rather, an assumption about the reasonable capabilities for future systems is made, which matches with the assumptions made by the NASA HAT group [20].

For reference, this set of technologies takes on the following properties in Table 2 in relation to the LCC proxy previously described.

Table 2 – LCC Information for Technology Options

	Description	Readiness	Other Users?	C_i
In-Space Propulsion	NTR	Low	No	1.000
	LOX-LH2	Exists	Yes	0.167
	LOX-CH4	Low	Yes	0.500
	SEP	Demoed	Yes	0.333
Descent engines and stage	LOX-LH2	Low	No	1.000
	LOX-CH4	Low	No	1.000
	Hypergol	Exists	Yes	0.167
Ascent engines and stage	LOX-LH2	Low	No	1.000
	LOX-CH4	Low	No	1.000
	Hypergol	Demoed	No	0.667
	Boil-Off Control	Low	Yes	0.500
	ISRU	Low	No	1.000
	Aerocapture	Demoed	Yes	0.333

Iso-Performance Analysis – In addition, each architecture also includes a set of mission mode parameters. These set the scientific performance level of the mission architecture, characterized by five mission aspects. These include 1) the destination choice, 2) the number of days on the surface, 3) the number of crew in the mission, 4) the possible outbound staging location (rendezvous point), and 5) the possible inbound staging location, which may differ from the outbound location. Although some science value can be altered by operations in the in-space portion, the vast majority of scientific value is set by the combination of surface operations and these mission mode parameters. Therefore it is adequate to assume that all architectures under these parameters have equal science value. For the analysis presented here, each set of architectures for a given destination has the same set of these variables.

Altogether, an architecture is defined by a set of 16 decisions, including five science value decisions, three architectural decisions, and eight technology options. These are most easily viewed in the form of a morphological matrix, seen in Figure 2.

Validation

The evaluation of architectures for each of the destinations was validated against the most relevant design case. Two clear methods for validation of such a model exist. The first method, used in this study, is to model an reference mission and check the output on the metrics of interest, determining the match with the detailed point designs. The second method would be to analyze the full design space and determine the placement of the point designs in the overall space. This method is difficult to apply in this situation, as it requires knowledge of how the point design fits in the overall space in order to verify its placement in the model. As full analysis of the overall space has not been rigorously conducted to a sufficient level of fidelity, this method would not adequately validate the model.

Mars Case – Design Reference Architecture (DRA) 5.0 – As there is no flight history for manned Mars missions, the most detailed and vetted design study was taken to be the baseline for comparison in the validation of the Mars case. Design Reference Architecture 5.0 is NASA's latest Mars reference mission, published in 2009 [21]. Once the reference mission was encoded in the formulation required for the established model, it was found that there were fundamental assumption differences between DRA 5.0 and the model baseline.

Mission Mode Parameters										
Destination	**Mars**	**Moon**	NEA (High Energy)	NEA (Low Energy)						
Number of Crew	3	**4**	5	6						(any user input)
Surface Mission Duration	**7**	14	21	30	90	180	360	**500**		(any user input)
Possible Outbound Staging Locations	EML1	**EML2**	SEL1	SEL2	Phobos	**None**				
Possible Inbound Staging Locations	EML1	**EML2**	SEL1	SEL2	Phobos	**None**				
Enumerated Features										
Architectural Features										
Habitation Distribution	120 Combinations									
Propulsion Element Distribution	776 Combinations									
Pre-Deployment (via SEP)	Yes	No								
Technologies										
Boil-off control for propellants	Yes	No								
ISRU	Yes	No								
Aerocapture	Yes	No								
TDI Propellant Type	LOX/LH2	LOX/LCH4	NTR							
Descent Propellant	LOX/LH2	LOX/LCH4	N2O4/MMH							
Ascent Propellant	LOX/LH2	LOX/LCH4	N2O4/MMH							
TEI Propellant	LOX/LH2	LOX/LCH4	NTR							
(Pre-Deployment Propellant)	SEP									

Figure 2: Architecture Morphological Matrix, with mission mode parameters presented in the results in bold

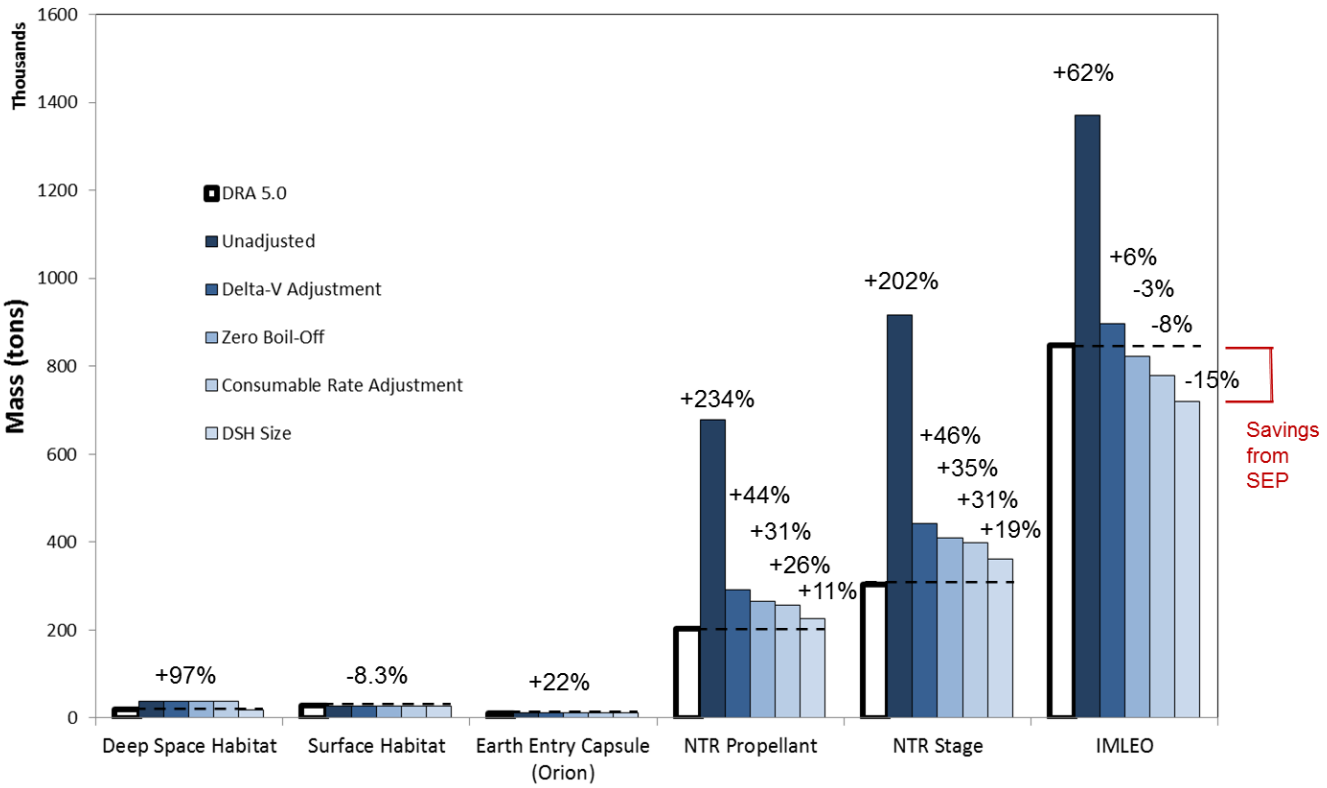

Figure 3: DRA 5.0 Validation Case

When unaccounted for, these assumption differences caused a significant shift in the model results away from those established by the design study. However, when adjusted in the model to match that of DRA 5.0, it can be seen that the mass results from the model match well with those of the design reference architecture. The assumption differences found to have the greatest impact include the ΔV requirements, propellant boil-off rate, consumable usage rate, and the size of the deep space habitat. These were all dependent on input assumptions to the model. Had the model been formulated in the same fashion as DRA 5.0, there would not have been a discrepancy between the model output and the design reference mission.

In the detailed reference study, NASA assumed that the mission would launch on a highly favorable launch date, requiring significantly less ΔV capability from the system. The model, on the other hand, assumes a more average energetic requirement, although it still assumes that the mission will be launched within the favorable portion of the Mars launch windows [22]. DRA 5.0 also assumes that the propulsion stages will have zero boil-off capability, meaning that no propellant will be lost in the system during the long in-space segments. NASA's more recent HAT studies as well as outside group studies [23] have shown that this is in fact not a reasonable assumption, and that boil-off rates should be expected to be closer to those reflected in the model. NASA JPL produces a consumable intake estimation tool, and this tool is integrated into the framework of the model calculations. However, DRA 5.0 makes the assumption that consumables will be used at a lower rate. As these consumables are present during the entirety of the mission (in some portion, obviously in relation to how they are consumed), the ripple effect of even a relatively small change in consumable mass creates a much bigger impact on the overall system IMLEO. The final difference to be accounted in the validation study was the sizing of the deep space habitat. The most recent sizing estimates coming out of the global space community were used in the model [24]. DRA 5.0, on the other hand, used ground-up creation of custom habitats. The model incorporated a parametric relationship to establish the total habitable volume required based on the duration of the mission and number of crew present [25], as well as the type of functions that each given habitat must perform. This is then matched with historical data regarding the mass of rigid habitats on a mass per volume basis. This establishes the overall mass of the habitat systems. The design reference mission, on the other hand, was able to design the habitats to a component level. It should be noted that this study makes the assumption that all habitats will be rigid, as inflatable habitats are in many respects still in their infancy, and no reliable parametrics for sizing of inflatable habitats can be established without significantly more flight history.

The aggregate information on the validation of the model versus DRA 5.0 can be found in Figure 4. This figure shows a series of bars for various system elements that drive IMLEO. The left-most white bar represents the reference

mission. The blue bars represent the progression of assumption adjustments from no adjustment to the final validation case. The overall IMLEO difference of 15% is believed to be associated with the savings from pre-deployment with SEP instead of NTR as present in DRA 5.0. This, even without this knowledge, is within reasonable error bounds for both the study and DRA 5.0.

Moon and NEAs – The validation for the Moon case was performed against the obvious candidate of the Apollo missions. Once encoded, the original validation without any adjustment came within 16% of the overall IMLEO. However, it was again found that there were fundamental assumption differences between the model and the actual Apollo missions. The principal difference was, once again, the energetic requirements. The model assumes that a Moon mission would require full lunar access, which innately requires a greater ΔV capability than the Apollo equatorial access requirements [26]. Once adjusted, the total IMLEO from the model estimate came down to +3.7%. However, a 20% difference in mass in the command module also affected the system, for similar reasons as the Mars deep space habitat. This resulted in a final IMLEO difference between the model and Apollo data of -3.4%.

Validation studies were also performed for the NEAs against preliminary data from NASA HAT. This validation study showed adequate agreement with the point designs. The generality of the assumptions in the model is more pronounced in the NEA designs due to the variety of possible destinations in the two primary categories. It is therefore re-emphasized that these are meant to be representative of the broader class of asteroids. [27]

3. RESULTS ANALYSIS

Full Feasible Tradespace Analysis

Analysis Setup – In this instance, the feasible tradespace is a restriction by destination on the maximum logical IMLEO that could be supported by a space program. All architectures whose mass is less than this limit remain within the tradespace for evaluation, regardless of other architectural traits. This is cropped from the full combinatorial space of all possible decision options, as described in Section 2. By destination, this upper bound is: 900 mt for Mars (~2 ISS masses [28]), 450 mt (~1 ISS mass) for the Moon and high energy NEAs, and 225 mt (~1/2 ISS mass) for low energy NEAs. Within this feasible region, several forms of analysis were performed, including a traditional review of the non-dominated architectures along the IMLEO-LCC Pareto front and an analysis of technology "switching" impacts, where particular

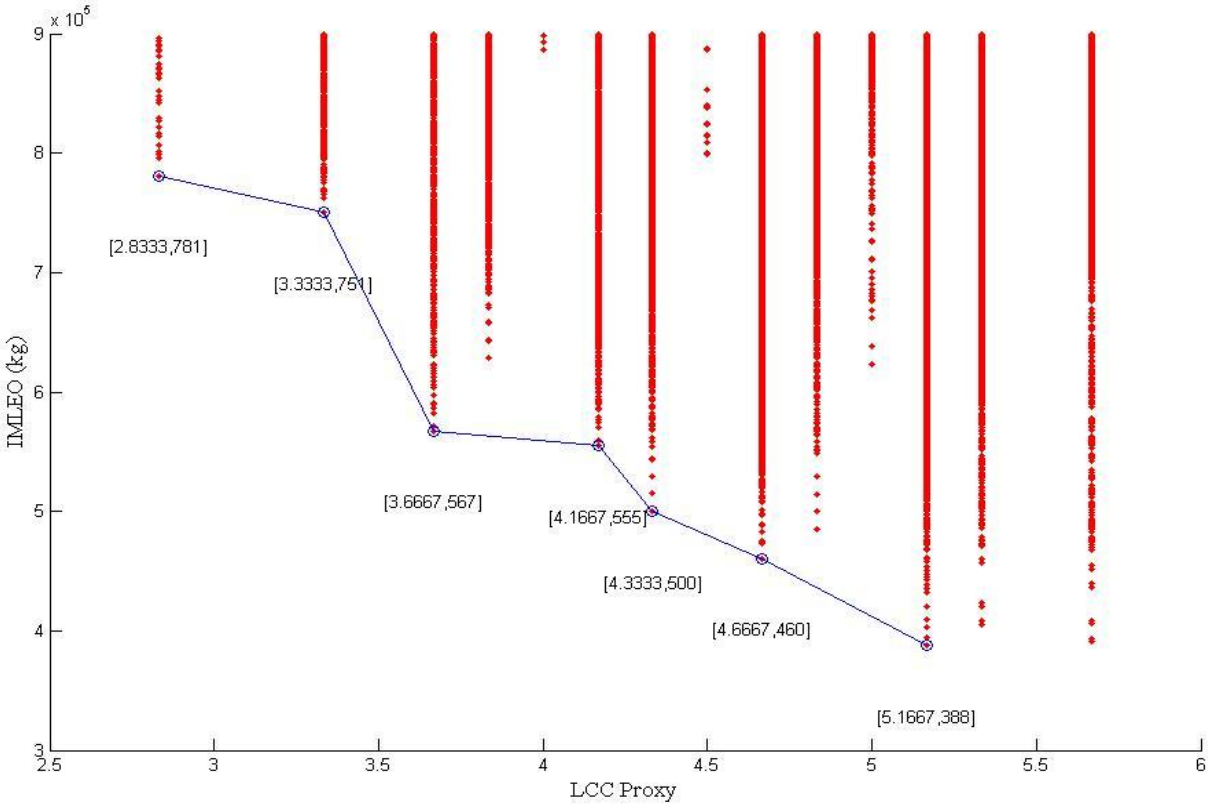

Figure 4: Mars Full Tradespace Pareo Front

technology options are disallowed in the tradespace.

Non-Dominated Pareto Fronts — For each destination, we will look at the Pareto front for the trade between IMLEO and the Life Cycle Cost proxy. The feasible tradespace for Mars architectures is shown in Figure 4. Architectures in the complete tradespace are shown in red, while the Pareto front architectures are circled in blue with an approximate front line between architectures.

We see that in the Mars case the Pareto architectures range in IMLEO from 388mt to 781mt, with a similar scaling of LCC from 2.833 to 5.166. Thus, there is a clear trade between IMLEO and LCC, where both approximately double at opposing ends of the tradespace. At the low end, a minimal IMLEO Mars architecture would require at least three launches of an upgraded SLS-130, while an architecture with the lowest LCC proxy requires seven launches.

There is a similar scaling to the Lunar architectures seen in Figure 5. A new feature arises in this tradespace, however, as there is a clear cusp at 4 LCC where there are negative impacts (in terms of IMLEO) associated with increased LCC. This corresponds to a point where the fractionation of propulsive and habitation elements results in higher total IMLEO than those architectures with more monolithic elements. For Moon missions, the lost IMLEO according to this low-fidelity analysis would be somewhere close to 170mt. Keeping in mind that this includes the propulsive stage required to leave LEO, which comprises a considerable amount of the total mass, this total mass in orbit is as expected. Reducing the number of or increasing the usage of more developed technologies results in a minimum IMLEO at minimum LCC of 255mt. The full range of architectures along the Pareto front for Lunar missions all require the use of two SLS-130 launch vehicles, although the lower massed architectures would be more open to mixed launch vehicles.

Low-energy NEA architectures in Figure 6 have similar characteristics to Moon missions but with significantly lower LCC. Along the Pareto front, the mass ranges from 158mt to 183mt, while the LCC proxy ranges from 0.50 to 1.34. High-energy NEA architectures, on the other hand, have more similar characteristics to Mars architectures. The tradespace for these arrangements are shown in Figure 7. Like the low-energy NEA architectures, these points also correspond to arrangements with much lower LCC proxy

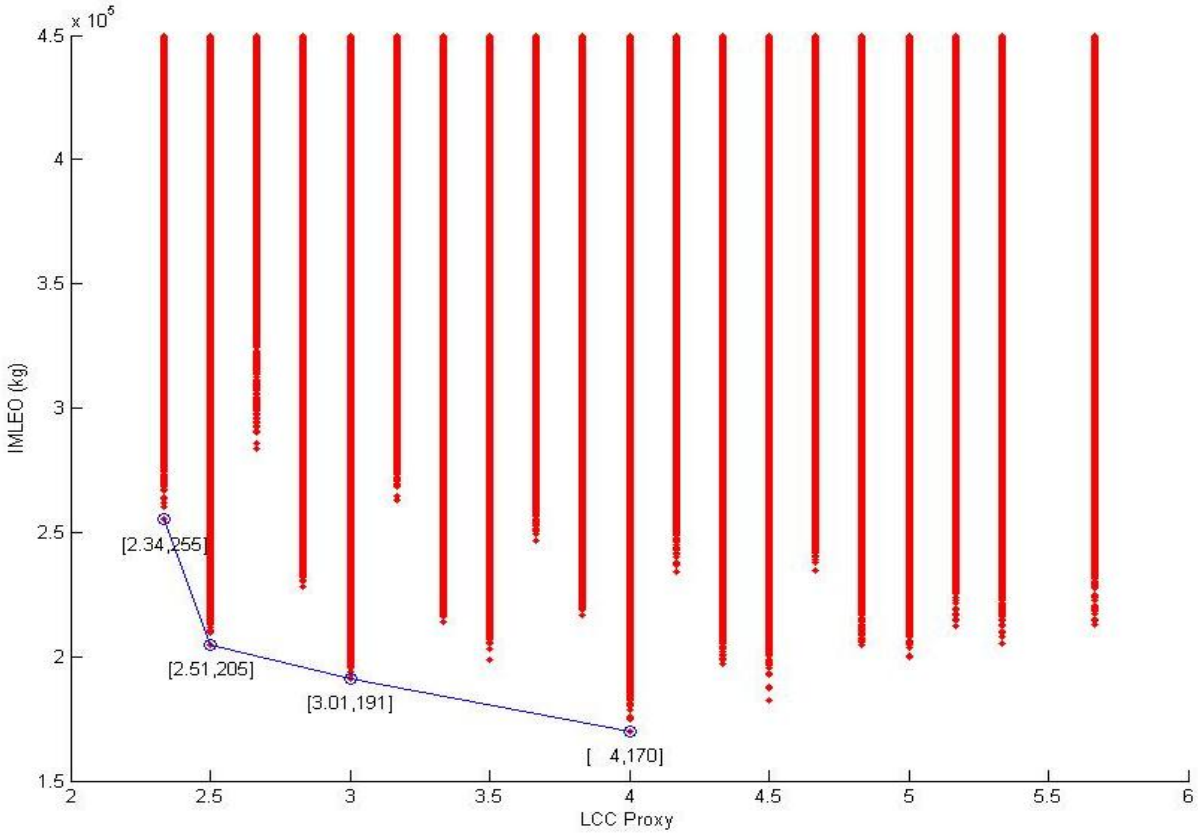

Figure 5: Moon Full Tradespace Pareo Front

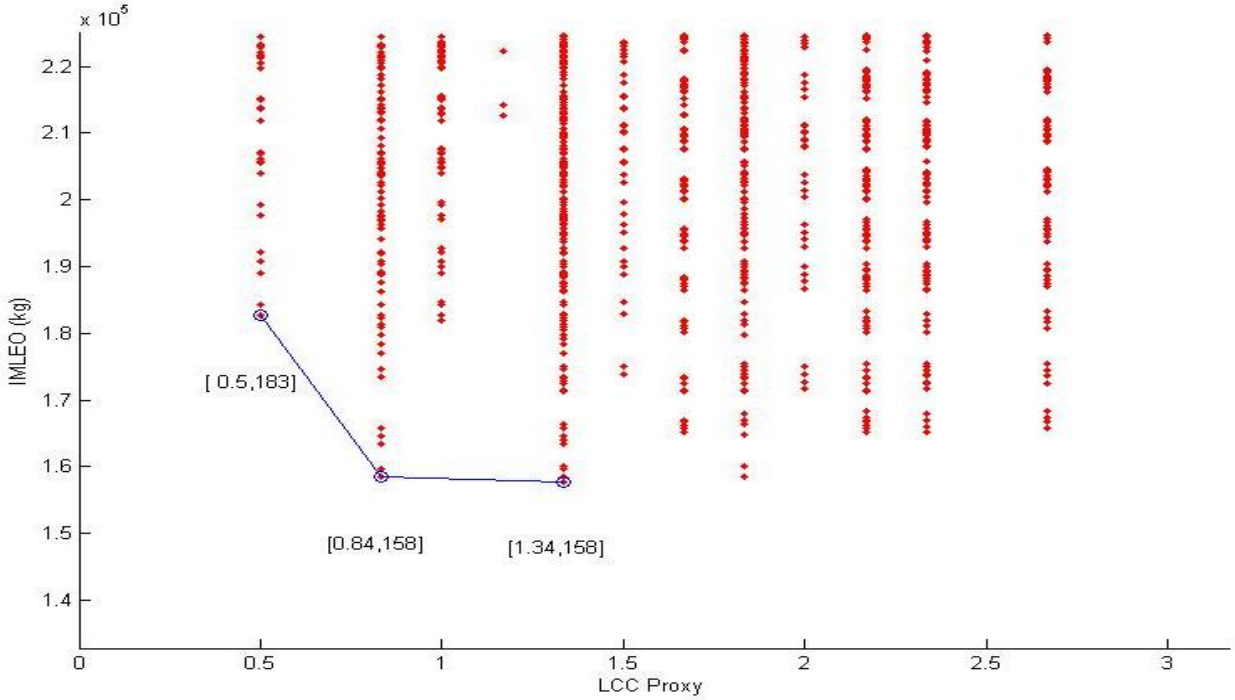

Figure 6: Low-Energy NEA Full Tradespace Pareto Front

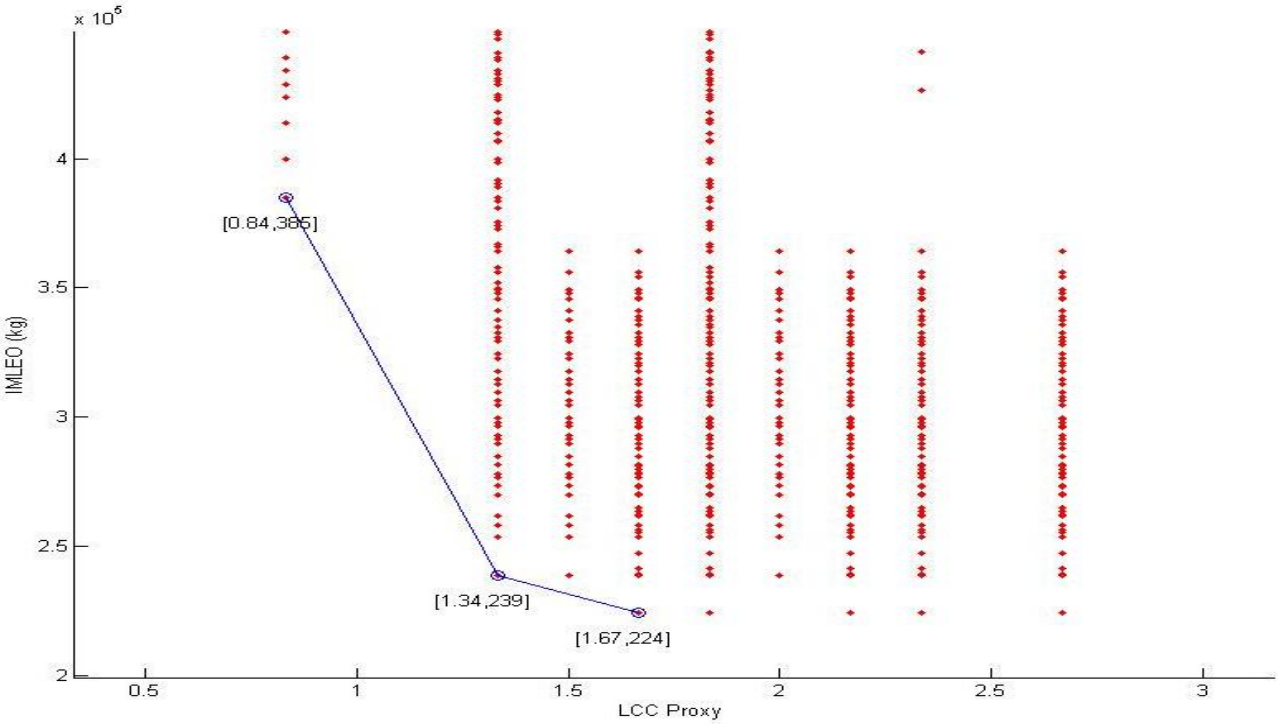

Figure 7: High-Energy NEA Full Tradespace Pareto Front

but with IMLEO closer to Mars missions, ranging from 224mt to 385mt. Thus, each of the destinations falls into a quadrant of the IMLEO-LCC Proxy tradeoff.

Technology "Switch" Impacts – One common piece of analysis in regards to interplanetary mission architectures is to look at the impact of individual technologies. With point

studies, this usually corresponds to the impact *on that architecture* when a technology is disabled. In this case, we look at the impact on the *tradespace*, meaning what happens

to the Pareto front when the option to use a technology is removed. For the sake of space we will look only at IMLEO optimality in this section. For each destination, we will examine the best architecture by IMLEO when several technology options are disabled. We will examine the impact of aerocapture, NTR propulsion, ISRU, and pre-deployment. The full list of "switches" is seen in Table 3. You will notice that there are several repeated architectures, meaning that the best architecture in those corresponding cases excluded all technologies under analysis. It should also be noted that NEA architectures, by definition, cannot have ISRU, and therefore these numbers are not included. Furthermore, to avoid confusion, aerocapture has an impact on both Mars orbit insertion and Earth orbit insertion, meaning that it may impact architectures for destinations without atmospheres.

Let us first examine the Mars architectures. Some surprising results occur in this case. We first notice that aerocapture has the greatest impact on IMLEO of all technology switches. This means that, in order to have a low cost mission, aerocapture must be developed for Mars architectures. In fact, the best architectures under this restriction has an even greater LCC Proxy than the overall best architecture, meaning that this architecture is heavily dominated in the full tradespace. It should be noted that, although the results are not herein presented, boil-off control is by far the most dominant technology need, where architectures without boil-off control do not exist at all within the feasible region. The second very interesting point is the low level of impact from NTR. If we choose not to develop nuclear propulsion, we, by necessity, increase the total IMLEO of missions to at least 486mt. This is a comparatively small trade with regards to other technology options. ISRU and pre-deployment have comparable impacts on the overall architecture mass. Again, this does not mean that we can develop an architecture scheme, decide to not invest in a technology, and then have these results. Rather, this analysis requires an up-front decision about particular technologies. Under those restrictions, there exist mission architectures with these properties, although they are very different from each other.

Lunar missions have fewer surprises. Aerocapture and NTR are both not used in the best overall architecture, and thus they make no difference under the restricted tradespace. ISRU has a relatively minor impact, while pre-deployment makes the most significant change. The benefits of low-thrust propulsion are clear for cis-Lunar space while they are less clear for farther destinations, interestingly enough. For farther destinations, use of high performance propellants reduces the overall advantage of pre-deployment elements on low-thrust trajectories.

Low-energy NEA architectures have small variation due to the exclusion of these technology options, with the worst case (aerocapture) resulting in an 8% increase in IMLEO over the base case. High-energy NEA missions see the greatest benefit from the inclusion of NTR propulsion and pre-deployment. In this case, aerocapture makes the least difference, with a 7% increase in mass versus the 72% increase for the exclusion of NTR and pre-deployment.

From this analysis it is clear that the investment in technologies should be highly dependent on the desired destination, as different technologies are key to the various destination choices.

Reduced Tradespace Analysis

Although we would all like to work in an ideal world where the best system design will be the one used in reality, the truth is that few complex systems are clean-sheet designs. Heritage components are often a necessary part of large systems, and the in-space transportation infrastructure for interplanetary missions is no exception. In this case these are not heritage systems *per se*; rather, they are components that have already been partially developed for segments of the overall missions. In the case of NASA, this includes two primary assets, the Multi-Purpose Crew Vehicle (MPCV) and the Space Launch System (SLS). As this model is decoupled from the launch architecture by design, the inclusion of the SLS system in the overall mission design makes no difference to the architectures under analysis. The

Table 3 – Technology Switches

	Moon		Mars		Low Energy NEA		High Energy NEA	
	IMLEO (mt)	**LCC**	**IMLEO (mt)**	**LCC**	**IMLEO (mt)**	**LCC**	**IMLEO (mt)**	**LCC**
Overall Best	170	4.0	388	5.2	158	1.3	224	1.7
No Aerocapture	170	4.0	751	5.8	171	1.3	239	1.5
No ISRU	191	3.0	555	4.2	--	--	--	--
No NTR	170	4.0	486	4.8	158	1.3	385	0.8
No Pre-deployment	237	3.0	572	4.3	163	0.8	385	0.8

following work examines the impact of the inclusion of the MPCV on the overall exploration tradespace. There are two methods for assessing this impact. The first is to assess the portion of the tradespace that includes architectures that incorporate an MPCV-like vehicle in the overall design. The second is to assess the impact of sizing these MPCV-like vehicles as the predicted actual MPCV. Only the former will be herein discussed, as the MPCV design is still under revision.

To assess the portion of the tradespace which includes MPCV-like vehicles, we filter the full combinatorial tradespace to only include architectures that have a small re-entry vehicle included in the design. For Mars missions, this does not include small descent vehicles, as the MPCV is not explicitly designed for the Martian atmosphere. The overlap of the full tradespace and the reduced tradespace for MPCV-like vehicles for Mars missions is shown in Figure 8. Any areas where red can be seen indicate that these are regions where there are no MPCV-like vehicles included in the architectures. In this case, we note that this reduced tradespace still covers approximately 83% of the total tradespace, indicating that the inclusion of the MPCV as a partially developed piece of hardware is not overly restrictive on mission designs for the in-space portion. More importantly, all architectures along the Pareto front are still included in the reduced tradespace.

Lunar mission results are even more encouraging, as 100% of the architectures in the feasible region include an MPCV-like capsule. For the sake of space, this graph and the NEA graphs are not included in this paper. For low-energy NEA missions, 89% of the tradespace is covered, including all Pareto-optimal architectures. For high-energy NEAs, this is much the same, with 87% coverage with all Pareto-optimal points included.

4. Summary and Conclusion

In order to gain a thorough understanding of the necessary components of exploration systems for manned missions to other rocky bodies, it is necessary to explore a more complete tradespace of mission architectures than traditional point designs. Through a functional decomposition of the high level abstraction of the in-space portion of these manned exploration infrastructures, a broader view has been gained from previous architectural studies on this matter.

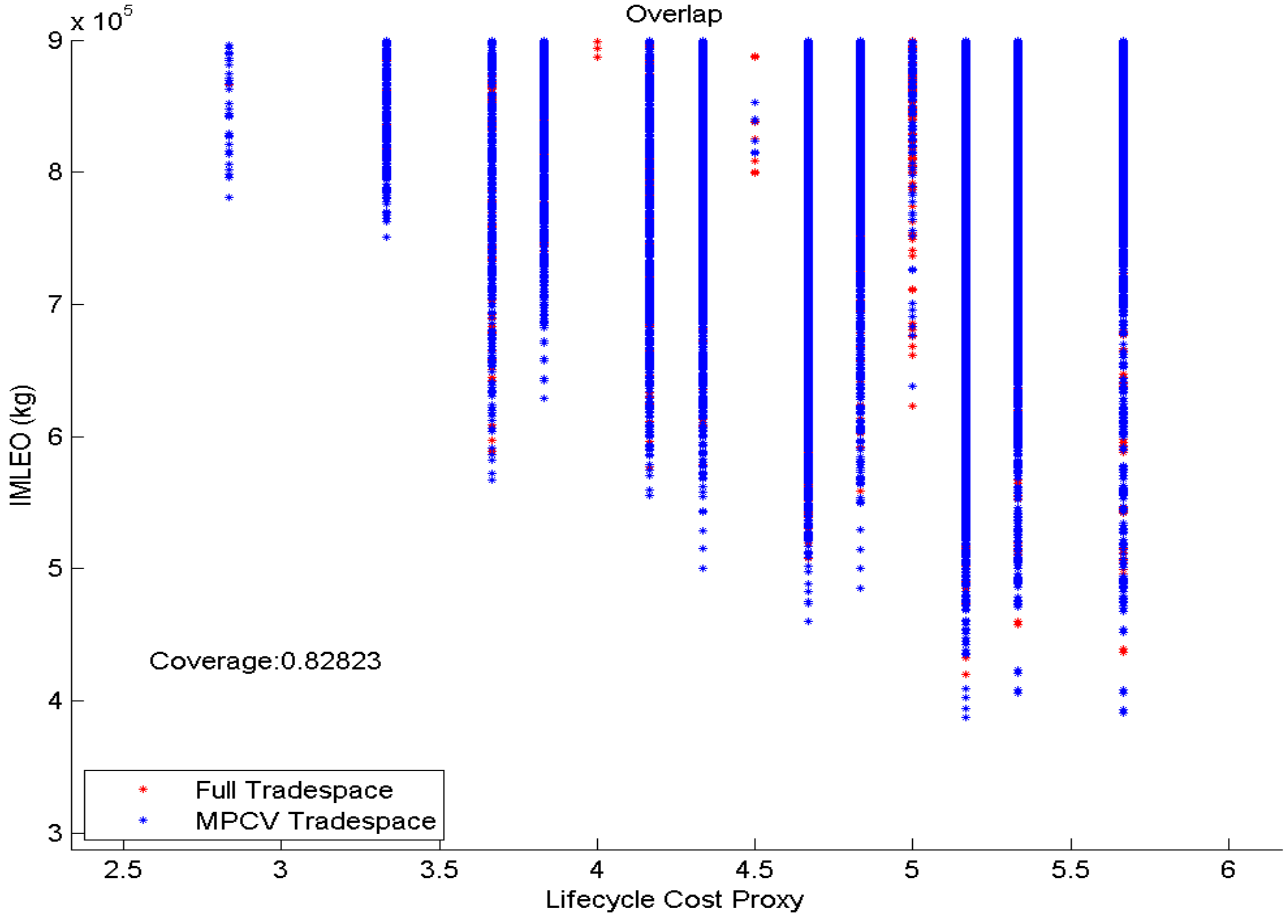

Figure 8: Reduction of Tradespace Due to MPCV Restriction

Coupling this functional decomposition with a suite of technologies of interest to the international space community, a technology impact analysis was performed to inform the technology investment process. Key to this analysis is the decoupling from science and exploration value by not considering the surface operations or launch infrastructure for any given system. The iso-performance nature of the tradespace allows the architectures to be compared on equal grounds. The model was also shown to be representative of the leading point designs as well as heritage systems.

Full tradespace analysis showed that there is a clear trade-off between mass in orbit and lifecycle cost, particularly for Mars infrastructures. For all mission architectures, even with the largest theoretical launch vehicle (SLS-130), at least two launches will be required for mission success. Technology "switch" analysis has revealed that aerocapture is the most important technology (of the four investigated) for Mars and low-energy NEA architectures, pre-deployment on SEP is the most important technology for Moon missions, and NTR is most critical for high-energy NEA architectures. This demonstrates the need for consensus of destination choice(s) early in the design and investment phase in order to properly determine technology portfolios.

Analysis of the impact of invested technologies, specifically the MPCV, shows that such an element is present in all Pareto-optimal architectures to the four destination choices. It also showed that greater than 80% of all feasible architectures retain an MPCV-like element. This supports NASA's investment in the MPCV program and demonstrates that it will be key in future mission success.

5. ACKNOWLEDGEMENT

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BIOGRAPHIES

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