Selection and Technology Evaluation of Moon/Mars Transportation Architectures

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Our purpose is to evaluate and select from a large family of Moon-Mars transportation architectures by integrating a general architecture network model with vehicle computational modules. A complete tradespace of 1162 unique transportation architectures for human missions to the Moon and to Mars provided by an Object Process Network based architecture generator has been interpreted and integrated with subsystem models. Three Mars and five lunar architectures are downselected based on total launch mass to LEO, risk, complexity and further criteria. Sensitivity analysis and trades of mass for different advanced propulsion types and in-situ propellant production availability are presented.

I. Introduction

This paper introduces the results of an attempt to integrate a novel modelling approach to studying space transportation system with traditional engineering models. A complete tradespace of 1162 unique transportation architectures for human missions to the Moon and to Mars provided by an Object Process Network (OPN) based architecture generator[1,2] has been interpreted and integrated with vehicles subsystem models. This method allows the rapid evaluation, comparison and analysis of a comprehensive tradespace of options in detail with physical metrics like launch mass to Low Earth Orbit, risk, cost and complexity. The options analyzed recreate previous studies of transportation systems for lunar and Mars missions[3,4] and provide insights into past decisions and their suitability for future space exploration.

A. Object Processing Network - An Architecture Generator

The Object Process Network (OPN) is a systems architecture modeling language, specifically a domain-neutral meta-language, designed to represent, generate, and manipulate simulation models. OPN represents a system in terms of a network of objects and processes. In this work, OPN is used to generate all possible operational sequences and associated hardware elements for space exploration missions.[1] It allows mission architects to declare the space of mission alternatives in terms of a finite set of repeatable states (mission phases) and finite time transitions (phase transitions). The contribution of OPN is the automation of the mechanical tasks in generating and selecting models of mission architectures. Such a rich options space demands rapid evaluation tools to sort through and rank the possible architectures. In this paper, we describe the result of linking the OPN output to general subsystem models and evaluating each architecture down to a vehicle level.

The output of OPN is a set of data structures with matrix fields representing 1162 architectures with mission elements and their operations. The rows of each matrix correspond to mission phases from launching stage to Earth surface descending state, as shown in Figure 1. The columns correspond to the various modules

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used in the mission like crew exploration vehicles, in-space transfer and surface habitats, propulsion stages and descent/ascent stages. The entries of the matrix indicate what operations the different modules are involved in during each phase. Zero means that the given module is not present in the architecture at that particular phase; 99 is reserved for disposed vehicles. Flights 1 through 5 correspond to different scenarios, as explained in Figure 2.

Phase	CEVa	CEVb	HAB1	HAB2	HAB3	HAB4	HAB4b	TMI1	TMI2	TMI3	TMI4	DSc	DS4	AS	TEI
E_Launching_State	1	4	1	2	0	0	0	1	2	0	4	2	4	4	1
E_Orbit_Assembly_State	1	4	1	2	0	0	0	1	2	0	4	2	4	4	1
E_Depart_Undocking_State	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
E_to_M_Transferring_State	1	4	1	2	0	0	0	1	2	0	4	2	4	4	1
Rendev_D_State	1	4	1	1	0	0	0	99	99	0	99	1	4	4	1
Undocking_D_State	1	4	3	1	0	0	0	99	99	0	99	1	4	4	3
M_Surface_Descending_State	1	4	3	1	0	0	0	99	99	0	99	1	4	4	3
Surface_Operating_State	4	4	3	1	0	0	0	99	99	0	99	99	99	4	3
S_Departure_Preparing_State	4	1	3	4	0	0	0	99	99	0	99	99	99	1	3
M_Surface_Ascending_State	4	1	3	4	0	0	0	99	99	0	99	99	99	1	3
Rendev_R_State	4	1	1	4	0	0	0	99	99	0	99	99	99	99	1
Undocking_R_State	4	1	99	4	0	0	0	99	99	0	99	99	99	99	1
M_to_E_Transferring_State	4	1	99	4	0	0	0	99	99	0	99	99	99	99	1
E_Return_Docking_State	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
E_Return_Undocking_State	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
E_Surface_Descending_State	4	1	99	4	0	0	0	99	99	0	99	99	99	99	99

Figure 1. Architecture Description Matrix. Mission phases are listed in the first column. Modules (mission elements) are listed in the first row. The cells contain the flight number of each vehicle in the respective phase. (Flights are defined in Fig: x) Flight 0 indicates that the vehicle is not used, whereas 99 means that the vehicle has been discarded.

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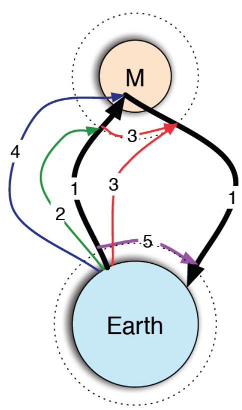

Flight 1 The crew is always on flight 1. Flight 1 progresses from the Earth surface to M surface and back.

Flight 2 Modules predeployed to M orbit for descent to M surface.

Flight 3 Modules deployed to M orbit for rendezvous and return to Earth.

Flight 4 Modules directly deployed to M surface (ex: surface habitat).

Flight 5 Modules left in Earth orbit (for rendezvous and re-entry).

Figure 2. Flights definition: A flight is a grouping of modules (modules that transfer together to destination). "M" indicates either the Moon or Mars. Note that the flight number does not indicate the launch order.[1]

B. Methodology Discussion and Motivation

The benefits of the rapid evaluation of a large architecture space are many. First, understanding sets of options and where families of architectures stand allows immediate focus on key aspects, for example, technologies that are necessary for feasibility or that fulfil primary objectives. On the other hand, looking at a large tradespace in detail allows the architect not to take key decisions early, but weigh all trade-offs against each other. Rapid architecture evaluation also allows sensitivity studies with respect to risk and technology readiness. Some architectures, while more costly upfront might be desirable as altogether in an uncertain future (with respect to stakeholder objectives). Some of the technology switches explored in this paper are in-situ propellant production versus payload and advanced propulsion types versus traditional chemical propulsion.

The following section describes the interpretation of OPN architecture description into operations and linking to vehicle subsystem modules.

II. Integration Methodology

The goal of the integration tool is to link architecture description output from OPN with vehicle modules comprised of subsystem models. As a result, each architecture is evaluated with pre-defined metrics. For metrics, we present launch mass to LEO as part of the analysis and discuss risk and complexity while comparing selected architectures.

A. Integration tool concept and structure

The idea of the integration tool is to extract flight durations and ΔVs from the architecture description matrix. Flight durations and destination information are used to calculate habitat masses. Then habitats and other assets are assembled per flight and used as payload information together with ΔVs to calculate masses of propulsion stages.

The integration tool contains one major routine which specifies high-level mission information, like crew size, propulsion and transfer trajectory types, fraction of propellant produced in the surface, mission surface duration, and scientific payload. Then the architecture matrix is read, modules are initialized depending on the matrix information (flight entries) and their operational sequences (if any) are recorded. The next step is to compute flight durations and delta Vs per phase per module, using two trajectory computational tools for the Moon and Mars respectively, described in section B. These provide options for Earth to Moon or Mars transfer with orbital or L1 rendezvous, and free return from destination for various propulsion types. The baseline analysis uses chemical propulsion. Once space durations are available per module per phase, the habitat masses are calculated, using habitat engineering models, described in C. After the habitat masses are known those are added per flight to calculate the necessary propulsion stages. An example is given in Figure 3. Finally, mass breakdown of the architecture, per vehicle, per subsystem is outputed together with any other relevant budgets.

B. Trajectory tools

To evaluate the architecture in terms of initial mass to LEO (IMLEO), which is a proxy metric for the cost, information about ΔV and flight time per mission phase is necessary. MATLAB scripts for both lunar and Mars trajectories have been developed. The input to these tools is mission information about departure and destination coordinates, orbital altitudes, propulsion system (chemical propulsion system / low thrust propulsion system) and sometimes trajectory types (ex: free return for abort options). The output is ΔVs and times of flight for the entire transportation system which are used as inputs to vehicle design.

Delta V and time of flight for a lunar trajectory are obtained by solving three-body problem considering the Earth, the Moon, and the spacecraft. Several approximation methods to calculate the lunar trajectories are available[5,6]. Lunar trajectory information used in the tool is listed in Table 1.

Table 1. Lunar Trajectory information

LEO to LLO	Departure Delta V : 3.15 km/sec Trip Time : 3.50 days Arrival Delta V : 0.85 km/sec
Descending to the Lunar Surface	Delta V : 2.1 km/sec
Ascending from the Moon	Delta V : 1.9 km/sec

Mars trajectories can be calculated by solving a two-body boundary value problem, called *Lambert problem*.[7] Opportunity to go to Mars occurs about every 2.2 years with different ΔV and time of flight values. Some examples of Mars trajectory information used for short-duration mission and long-duration mission are presented in Table 2.[8]

Table 2. Delta Vs and times of flight table for Mars mission

Opportunity	dV(TMI), km/s	$T_{outbound}$, days	Venus flyby, yes/no	dV(Mars orbit), km/s	$T_{surface}$, days	dV(TEI), km/s	$T_{inbound}$, days	Venus flyby, yes/no
Nov 20, 2013	3.8	256	0	1.8	60	3.1	311	1
Nov 23, 2015	4.9	221	0	3.9	60	3.3	250	1
May 17, 2018	3.5	235	0	1.2	515	1.3	191	0
Sep 06, 2022	4.1	181	0	2.1	540	1.7	190	0
Oct 12, 2024	4.0	194	0	2.7	515	1.4	210	0

C. Modules and Subsystems Summary

The mission elements are defined in Table 3.

There are four general module types used to design four types of vehicles. The *HabitatModule* is used to design all the habitats, practically the life support subsystem of CEVa, CEVb and HAB1, HAB2, HAB3, HAB4, HAB4b. The *EarthEntryModule* is used to add the re-entry capability to the CEVa and CEVb vehicles. The *DescentAscentModule* is relevant for the descent stages DSc, DS4 and the ascent stage AS. Finally, the *OrbitTransferModule* is used to model all propulsion stages TMI1, TMI2, TMI3, TMI4 and TEI. There is also a lighter version of the propulsion module called *OrbitalTransferStage* which is used as an extra fuel tank (kick-stage).

Each module type is comprised of relevant subsystems, including ACS, avionics, communications, docking mechanisms, ECLS (environmental control and life support), EDLS (entry, descent and landing subsystem), payload, power, propulsion, structures and thermal.

Table 3. Mission Elements Description

CEVa	Crew Exploration Vehicle A: launches crew, always exists
CEVb	Crew Exploration Vehicle B: ascends crew from surface, optional
HAB1	Habitat module on flight 1
HAB2	Habitat module on flight 2: pre-deployed for descent
HAB3	Habitat module on flight 3: pre-deployed for return
HAB4	Surface Habitat module on flight 4: pre-deployed directly to surface
HAB4b	Ascending Habitat module on flight 4: dedicated to ascent (and sometimes return) only
TMI1	Trans M Propulsion Module on flight 1: TMI injection on flight 1
TMI2	Trans M Propulsion Module on flight 2: TMI injection on flight 2
TMI3	Trans M Propulsion Module on flight 3: TMI injection on flight 3
TMI4	Trans M Propulsion Module on flight 4: TMI injection on flight 4
DSc	Descent stage (for crew): provides propulsion for crew descent to M surface
DS4	Descent stage (for surface assets): provides propulsion for descent pre-positioning of assets on the surface (ex: habitat)
AS	Ascent stage: provides the propulsion for the crew return to M orbit from M surface
TEI	Trans Earth Propulsion Module: performs the TEI burn, correction burns and Earth Orbit injection for the crew return trip

D. Mass flow calculations

In this section we give an example of mass flow calculations based on an architecture description generated by OPN. The example chosen is one of the Mars architectures chosen for further analysis from the pool of 1162 architectures.

The architecture shown in Figure 3 (left) has a CEVa which launches the crew and goes to M orbit, another CEVb which acts as an ascent habitat and serves as a re-entry vehicle. CEVb is pre-deployed directly to the surface together with the ascent stage AS. A return habitat (HAB3) together with a TEI propulsion stage are pre-positioned in M orbit for rendezvous after ascent and return. The surface habitat is also the crew transfer habitat to M (HAB1). All three flights, with crew (1), direct to surface (4) and to M orbit (3) are powered by separate TMI stages, TMI1, TMI3 and TMI4 respectively. Figure 3 (right) shows the tree of mass flow calculations.

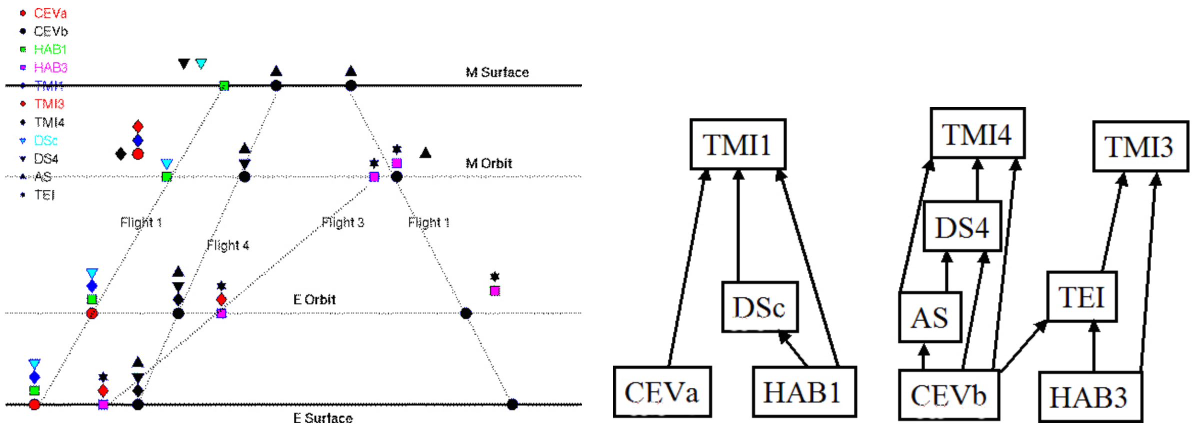

Figure 3. Mass flow calculations example

First, all habitat masses (CEVa, CEVb, HAB1, HAB3) are calculated using ΔV and flight durations information from our trajectory tools (section B). CEVa and CEVb also include some propulsion for re-entry, as well as heat shield mass. Since the mass of HAB1 is known, the descent stage payload is determined and its mass can be calculated leading to DSc in the tree. Finally, CEVa, HAB1 and DSc comprise the payload for the TMI1 stage, so this is the next module whose mass can be determined. This concludes the calculations along flight 1. Next, CEVb is the only module ascending from the surface, so its propulsion stage (AS) mass can also be calculated. CEVb and AS descend together powered by DS4, so this stage mass is determined and finally, altogether CEVb, AS and DS4 transfer to M on TMI4, which concludes the calculations on flight 4. Flight 3 can be traced similarly.

E. Assumptions, constants, switches

1. Modeling assumptions

- Aerocapture is used at Mars (required to make free-return feasible)
- Trajectories
  1. Crewed vehicles traveling to Mars use free-return trajectories for long duration missions
  2. Assets in lunar orbit during surface stay include sufficient propellant for bi-elliptic transfer to enable anytime return from anywhere on the lunar surface

     i. Separate trade identified bi-elliptic plane change transfer as most mass efficient - superior to simple plane change or EML1 rendezvous point
- In-Situ Propellant Production (ISPP)
  1. Mars ISPP: Each mission carries required production plant, power source, and chemical feedstock (unique site per mission scenario)

     i. Significantly less infrastructure required relative to moon due to simplicity of acquiring and processing CO2 from atmosphere on Mars
  2. Moon ISPP: Production equipment and power source are positioned once to produce oxidizer, return fuel is brought from earth (single base/outpost scenario)

     i. Propellant production, excavation, and power systems could be amortized over series of missions
- Propulsion Technology

1. Liquid Hydrogen/Liquid Oxygen is used for Earth departure in Chemical Propulsion case
2. Nuclear Thermal propulsion is used for Earth departure for all elements in NTR cases
3. Nuclear Electric / Solar Electric is used for Earth departure of un-crewed elements in EP cases
4. Liquid Methane/Liquid Oxygen is used for all operations near M (ascent, descent, earth return) due to boil-off

- L1 and Mars/Moon orbit are modeled as the same destination for this analysis.

2. Architecture selection criteria

1. Architectures with complex docking and undocking sequences in Moon/Mars orbit are removed from consideration in order to minimize the mission risk.
2. Architectures that require reconfiguration of vehicle modules on the surface of the Moon or Mars are eliminated.
3. Architectures which do not bring a TEI stage to Moon orbit with the crew are eliminated. Having a TEI stage with the crew on the way to the Moon enables free return from lunar orbit without the necessity of landing or rendezvous. (Without the TEI stage, the crew could be stranded in lunar orbit if rendezvous with the TEI stage fails.)
4. Architectures requiring ascent vehicles mounted on top of a surface habitat are removed from consideration because a combined ascent vehicle and surface habitat will result in coupling between the two vehicle designs, which limits individual element evolvability and complicates the development process.
5. Mars mission architectures with descent habitats designed for short durations are eliminated because of the danger of landing far from the long duration habitat. Supplies for a few days would be insufficient to support the crew until they reach the long duration habitat. This is only a requirement for Mars missions because of the availability of anytime return-to-Earth in lunar missions.
6. Architectures with greater initial launch mass to LEO (IMLEO) but no added benefits (as compared to other architecture designs) are removed from consideration.

III. General Results Summary

IMLEO of all OPN-generated architectures were evaluated for the following specifications.
”Mars Baseline” / ”Moon Baseline”
Propulsion Type: Chemical / Chemical
Crew Size: 5 / 5
Surface Stay Time: 515 days / 4 days
Cargo to the Mars (Moon): 8300 kg / 100 kg
Returning payload: 100 kg samples / 50 kg

The technology options in our analysis are different propulsion types, including chemical, electric propulsion with solar and nuclear power and nuclear thermal propulsion, the availability of in-situ propellant production and different power options like batteries types, fuel cells and RTGs for each vehicle. These technology options are traded for missions with different crew size, payload and surface durations. The baseline architectures use chemical propulsion for all flights, crewed and non-crewed, outbound for prepositioning and inbound. Five-member crew is considered in all baseline calculations. The baseline short Moon mission includes 4 days on the lunar surface. The baseline Mars mission is long, with 515 days on the surface, a number which is chosen based on the orbital dynamics. The short Moon mission carries 100 kg science payload to the destination and 50 kg back. The long Mars mission carries 1200 kg to and 100 kg back respectively.

Figure 4 shows the IMLEO for the Moon baseline architectures (left) and the Mars baselines (right). Clearly, there is great variation in masses, indicating the superiority of some architectures. The Moon architecture masses are obviously more uniform and less sensitive to logistics than the Mars masses (notice the scales are different). Despite the great variation in the Mars case, it can be observed that the 200 lightest

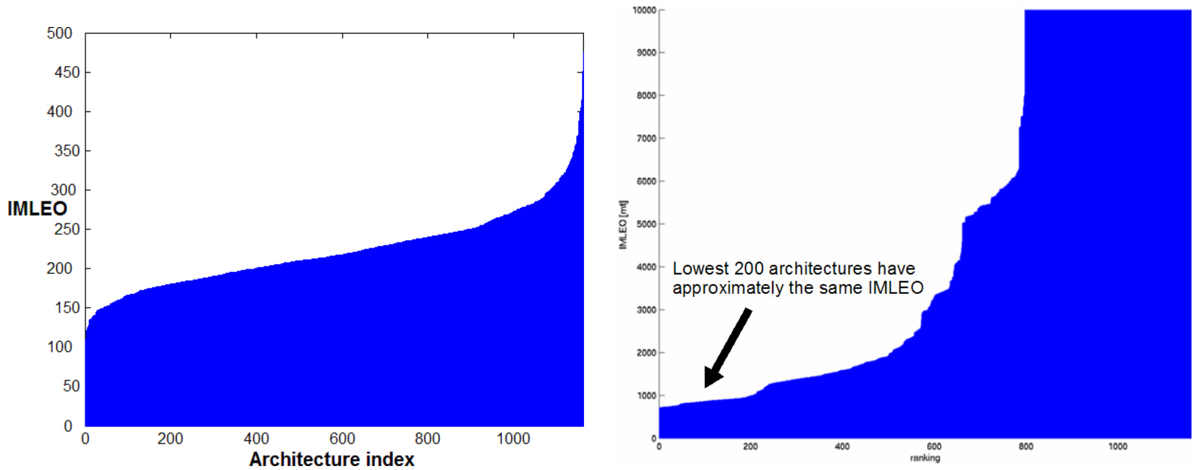

Figure 4. Moon (left) and Mars (right) baseline launch masses in metric tons; 5-member crew with chemical propulsion and no ISRU

architectures have almost constant mass. This smaller set of lighter architectures was considered for further analysis with the additional set of criteria described in section E.

After applying these criteria, three Mars and three lunar architectures are chosen to be further analyzed, described in the following section.

IV. Architecture Evaluation, Trades and Conclusions

A. Candidate Mars architectures

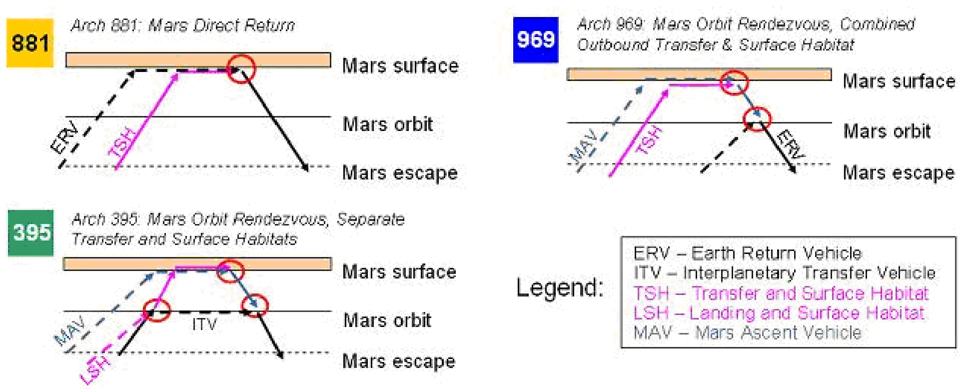

Figure 5. Selected candidate Mars architectures

Architecture 881
This architecture represents a "direct return" scenario which uses ISRU. The Earth return vehicle is prepositioned on the surface. The crew launches in CEV A, and travels to Mars with a transfer and surface habitat and the Mars descent stage.

Architecture 969
This architecture is similar to the NASA JSC 1993 Mars Design Reference Mission. The Earth return transfer habitat and TEI stage are prepositioned in Mars orbit to rendezvous with CEV B. CEV B is the only vehicle returning to the Earth while the TEI stage and transfer habitat are discarded prior to Earth entry. CEV B and the ascent stage are prepositioned directly to the surface. The crew launches in CEV A and travels to Mars together with the transfer / surface habitat and descent stage.

Architecture 395
CEV A and in-space transfer habitat travel to Mars orbit and stay in orbit ready to rendezvous before

return. CEV A performs a rendezvous maneuver in Mars orbit with the descent stage and surface habitat. The Mars ascent vehicle is prepositioned on the surface.

Technology switches - ISPP and advanced propulsion system

ISPP enables architecture 881 (feasible only with ISPP) while does not make much impact on the IMLEO if architecture 969 and 395. This implies that nuclear power may also be necessary to enable architecture 881, which has highest crew safety. Solar Electric Propulsion (SEP), Nuclear Electric Propulsion (NEP), and Nuclear Thermal Propulsion (NTR) technologies are also studied for the candidate Mars architectures. Chemical propulsion system yields the highest IMLEO values. SEP and NEP are similar to each other. Nuclear thermal has the lowest launch mass. Nuclear thermal propulsion system combined with ISPP can yield IMLEO less than 400 metric tons. Figure 6 presents the IMLEO values for the candidate architectures with different technology switch options.

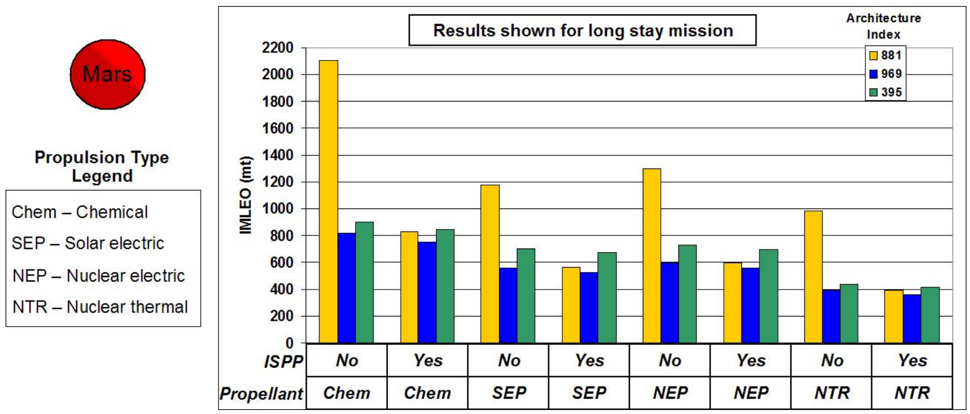

Figure 6. Technology Switches Analysis for Mars

Trades for Mars candidate architectures

Architecture 881 has low mission and safety risk because there is no Mars orbit rendezvous and the return habitat is accessible on the surface, so it can serve as safe haven and can be inspected and maintained by crew on Mars. On the other hand, higher development risk is involved because ISPP is required for this architecture. Architecture 969 has moderate mission and safety risk which could be partially mitigated by prepositioning an additional surface habitat; however, this would impact IMLEO. Moderate development risk is envisioned, as the mission is feasible without ISPP. Architecture 395 has higher mission and safety risk due to the two Mars orbit rendezvous operations. The development risk is low, since ISPP is not necessary for mission feasibility and the developments of the in-space and surface habitat are decoupled.

B. Candidate Lunar architectures

Looking across all the technology options and using the screening criteria, three lunar architectures (five, including variants) of interest were identified.

Architecture 1/6

Architecture 1 is same as the architecture proposed in NASA's First Lunar Outpost study in 1992. All modules transfer in one stack, including the CEV, descent stage, ascent stage, and TEI stage. Stages are disposed of as they are used. Architecture 6, a variant of architecture 1, is the same as architecture 1 with the exception that the descent and ascent stages are prepositioned in lunar orbit and rendezvous with the crewed flight for descent.

Architecture 12/41

Architecture 12 is similar to architecture 1 except it leaves the return TEI stage in lunar orbit, thus requiring a rendezvous operation prior to return. As in architecture 1, all modules transfer to lunar orbit together. The CEV goes to the surface and performs a rendezvous with the return propellant in lunar orbit. In architecture 41, a variant of architecture 12, the descent and ascent stages are prepositioned in lunar orbit, waiting to rendezvous for descent.

Architecture 67

Architecture 67 is Apollo-like with a dedicated lander and CEV that remains in orbit. CEV A takes the

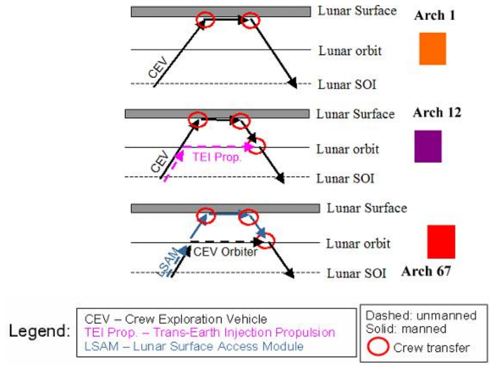

Figure 7. Candidate Lunar Architectures

crew to lunar orbit and back to the Earth. A separate CEV B is prepositioned in lunar orbit to be used for descent/ascent. In lunar orbit, the crew transfers to CEV A for the return journey.

Technology switches - ISPP and advanced propulsion system

ISPP lowers IMLEO of architecture 1 significantly. But it affects architecture 67 only slightly. As for advanced propulsion systems, NTR results in the lowest mass. In general, SEP dominates NEP; however it would entail deployment challenges because of the high power requirements (>500 kW-e). The advanced propulsion analysis on lunar candidate architectures shows high variations in IMLEO for seemingly similar architectures, like architectures 1 / 6, and architectures 12/ 41.

Trades for lunar candidate architectures

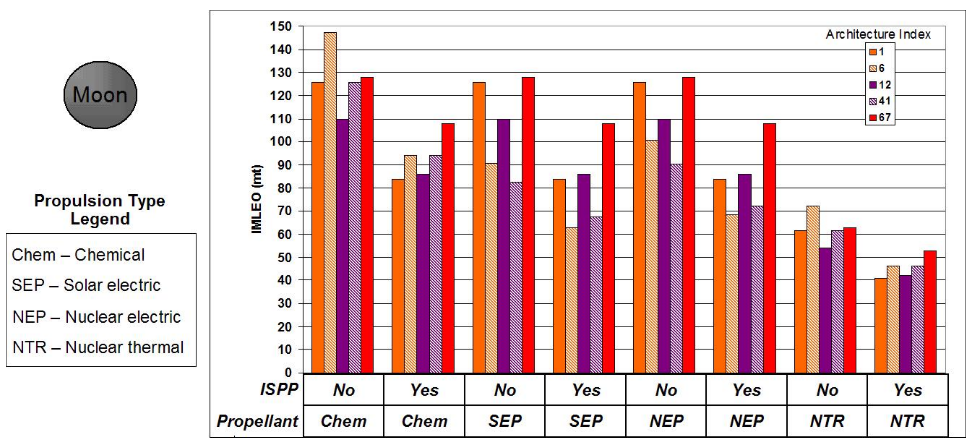

Figure 8. Technology Switches for Lunar Architectures Analysis

Architecture 1 is the lowest mission and safety risk architecture, since it does not involve lunar orbit rendezvous and the CEV is accessible on the surface for maintenance and monitoring. Building redundancy in the power and ECLS system is recommended in order to enable survival for Apollo-13-like emergencies. This architecture has low mass in general in the baseline case, and has the lowest mass when ISPP is used. Architecture 12 has higher mission risk due to the lunar orbit rendezvous. If anytime return is desired, then bi-elliptical transfer may be required for any landing site. Since the return propulsion remains in LLO, automated monitoring and additional orbit control propellant are required. In general, this is a low mass architecture with or without ISPP. Architecture 67 has a moderate mission and safety risk. It requires lunar orbit rendezvous like architecture 12 and probably additional propellant and guidance to maintain the orbit of the CEV A/TEI stage stack. Also, more hardware is needed since two habitats are designed separately.

V. Related Work and Conclusions

In this paper, we presented a method to generate a large number of space transportation architectures using OPN and how these OPN-generated architectures can be evaluated in terms of IMLEO using an integration tool. We also presented criteria to select interesting architectures other than IMLEO and several technological switches to see the effect of the specific technology on the architecture. Using proposed tools and criteria, several interesting architectures for Mars mission and lunar mission were selected and studied for trades. Selected Mars architectures have multiple trans-Mars flights and prepositioning. (881: Earth Return Vehicle, 969: Transfer HAB and TEI stage, 395: Mars Ascent Vehicle) On the other hand, selected lunar architectures are single trans-Moon flight architectures. Two of the selected lunar architectures have a CEV to the lunar surface (Architecture 1 and 12) and one is an Apollo-like architecture (architecture 60).

More in-depth study on the CEV destination for the lunar mission was carried out by Wooster, et al.[10] The result of the study indicates that for lunar architectures, having the CEV (defined as the crew launch and reentry capsule) travel to the lunar surface is the best approach from a mass, risk, and cost perspective, and allows for both a maximum degree of commonality with Mars and flexibility across architectures. While the Apollo architecture was appropriate for the requirements of the time, it is no longer optimal (given current requirements to access the entire lunar surface with long duration stays and anytime return capability).

Also since all vehicle elements depart from LEO, optimal launch packing analysis is necessary. Launch vehicle selection and sizing for lunar and Mars missions based on optimal grouping of mission elements has been studied by Gralla et al.[11]

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