Modular Building Blocks for Manned Spacecraft: A Case Study for Moon and Mars Landing Systems

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Abstract. A method is presented for the selection of optimal modular building blocks for platforming of manned Moon and Mars landing systems employing modularity on the subsystem level; platforming shall here be defined as the reuse of designs across different systems. The motivation for platforming is the need to reduce overall Moon and Mars exploration architecture lifecycle cost by lowering spacecraft development, test, and fixed production cost, and to provide flexibility in system design to accommodate changes in the exploration architecture. The fundamental idea is to compute the surplus in functional attributes generated by using particular building block (module) sizes, then relate the surplus to a cost function, and finally select the building block sizes with minimal additional cost. Results are presented for modular crew compartments, propellant tanks, and engines. The proposed method is potentially helpful for platforming decision-making, as well as subsystem technology selection in a broad class of engineering systems.

Introduction

On January 14, 2004, President George W. Bush proposed a new Vision for Space Exploration, which provides a framework for the United States' manned and unmanned space activities in the next several decades. Among the primary objectives are a return of humans to the Moon no later than 2020, and a human Mars exploration program in the following decade [www.nasa.gov, 2004].

With the Vision for Space Exploration, a five-year budget plan, as well as a long-term budget forecast for NASA was published (see Figure 1). Both exhibit strong constraints on the resources available to NASA for the development of new exploration systems. A large part of the resources necessary for the development and acquisition of new spacecraft will be provided by retiring the Space Shuttle around 2010, and transitioning leadership of the ISS around 2016 / 17 (Figure 1).

The cumulative resources available (see Figure 1) for exploration systems development from 2004 to 2020 are between around $40 billion (FY 2004) [www.nasa.gov, 2004]; assuming a funding level of about $10 billion (FY 2004) per year, another $100 billion will be available from 2021 to 2030. These resources are comparable to the $130 billion (FY 2004) estimated for the Apollo program from inception through 1969 [www.nasa.gov, 2004]. Although it might be argued that for the Apollo program technology had to be developed that is already available today, it nevertheless appears imperative to employ design reuse between Moon and Mars exploration systems to the maximum extent feasible in order to fulfill the Vision for Space Exploration's goals within the budgetary constraints. Two separate point designs for Moon and Mars exploration systems seem to be unrealistic in terms of overall Moon and Mars exploration architecture lifecycle cost, even if they appear desirable from a technical performance perspective.

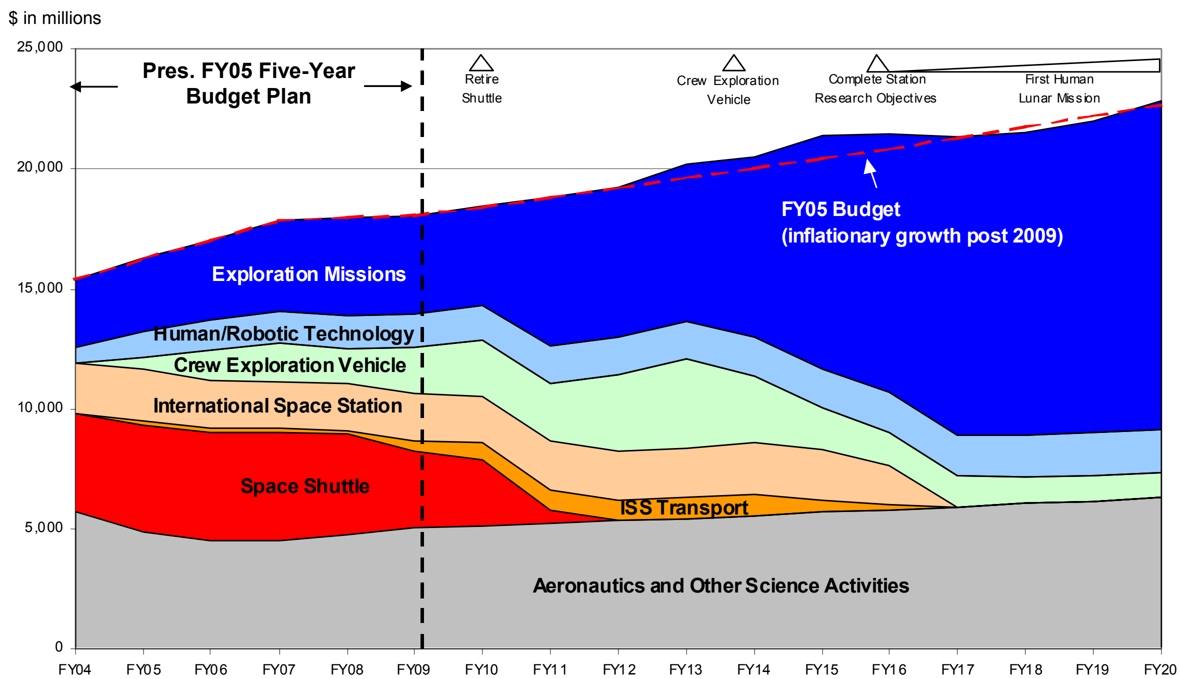

Figure 1: Overview of the NASA budget forecast for the years 2004 to 2020 [President Bush, 2004]

Figure 2 shows a hierarchical breakdown of Moon and Mars exploration systems from a landing system perspective. Although the classification of system hierarchy is naturally somewhat arbitrary, it is the authors' opinion that two major approaches to incorporating design commonality into the exploration architecture can be readily identified in this top-level view:
The first approach is reusing the exploration system point design (vehicles, propulsion stages) for Mars (Moon) to the maximum extent possible for Moon (Mars) exploration. This implies a higher level design commonality, and the vehicle or propulsion stage in question would then have to be developed for the most stringent use case (see Figure 2).

The second approach is to design the Moon and Mars exploration systems for maximum commonality on the subsystem and component level. In this case modular subsystems and components are of special relevance to enable the scaling of functionality and facilitate design reuse and commonality.

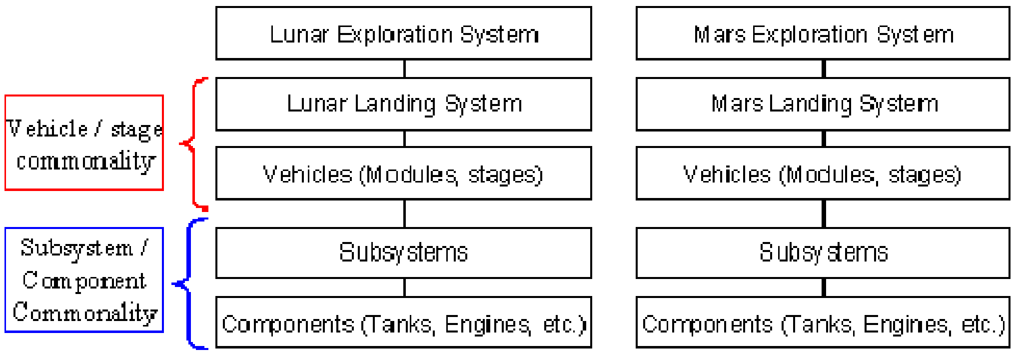

Figure 2: Commonality and modularity on different system levels

The latter approach shall be investigated here. Both are illustrated in context to the exploration system hierarchy in Figure 2. It is important to mention that the two approaches are not mutually exclusive: the modularization on the subsystem level could for example be constrained to using only tanks, engines, and crew compartments of a Mars point design as modular building blocks for designing a Moon exploration system. Also, these two approaches may not be the only ones possible: using several identical smaller vehicles to provide the functionality of a bigger one might provide a way for modularization on the vehicle / stage level. Other possible approaches to commonality through design reuse are, however, not discussed here and shall be considered topics of further work.

Conceptual Manned Spacecraft Modeling

Manned spacecraft are very complex systems due to the cutting-edge technology employed, the high level of coupling between the subsystems (e.g. Electrical Power System (EPS) and Environmental Control and Life Support System (ECLSS)), and the high degree of redundancy required, to name only a few characteristics. A detailed design therefore is a major effort. For conceptual design purposes, however, empirical and scaling models have been provided in literature to enable the computation of basic spacecraft characteristics as a function of only few input parameters [Larson and Pranke 2002; Springmann and de Weck 2004].
A manned spacecraft is assumed to consist of the following simplified elements:

1. One crew compartment (life support, power, ADCS, thermal control, pressurized volume, access hatch, etc.)
2. One or several propulsion stages (tanks, plumbing, associated structures)
3. One or several engines per propulsion stage
4. Landing gear
5. Parachutes (for Mars landers)
6. Heat shield / protection (for Mars landers)

The crew compartment mass, m_{cc}, can be modeled empirically by a power function [Larson and Pranke 2002]:

$$m_{CC} = 592kg \cdot \left(N_{Crew} \cdot \Delta t_{Mission} \cdot V_{\Pr essurized}\right)^{0.346}$$

N_{crew} is the crew member capacity of the vehicle, $\Delta t_{mission}$ is the mission duration in days and $V_{pressurized}$ is the pressurized volume in m^3. There are several approaches to compute the necessary pressurized volume; here, a polynomial of fourth order shall be used to estimate the habitable volume required as a function of mission duration:

$$V_{Habitable}\left(\Delta t_{Mission} \leq 270d\right) = 19m^3 \cdot N_{Crew} \cdot \left(1 - \left(\frac{\Delta t_{Mission} - 270d}{270d}\right)^4\right)$$

Figure 3 shows this function in relation to habitable volume requirements published in [Larson and Pranke 2002]. For mission durations longer than 270 d, the habitable volume is assumed to stay constant at about 19 m^3 per crewmember.

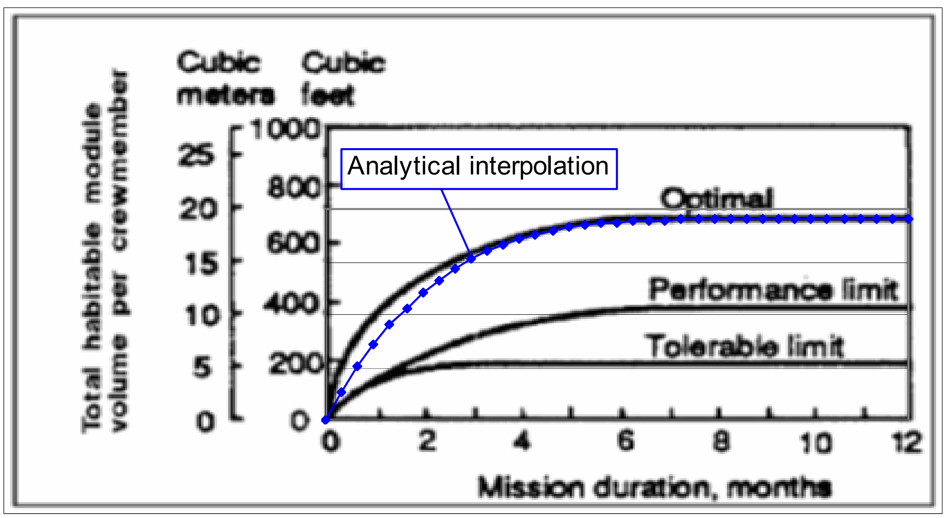

Figure 3: Analytical habitable volume model in relation to volume requirements defined by NASA [Larson and Pranke 2002]

Given the habitable volume, the necessary pressurized volume can then be calculated by the following empirical relationship [Larson and Pranke 2002]; the pressurized volume also contains the volume consumed by subsystem equipment:

$$V_{\Pr essurized} \approx 3 \cdot V_{Habitable}$$

The mass of a rocket engine can be estimated based on the thrust of the engine [Larson and Pranke 2002]: the mass of the engine generally decreases with thrust (see Figure 4). The reference data (black dots) given Figure 4 represent various engines with different thrust and different propellant combinations (liquid hydrogen / oxygen and hypergolic propellants).

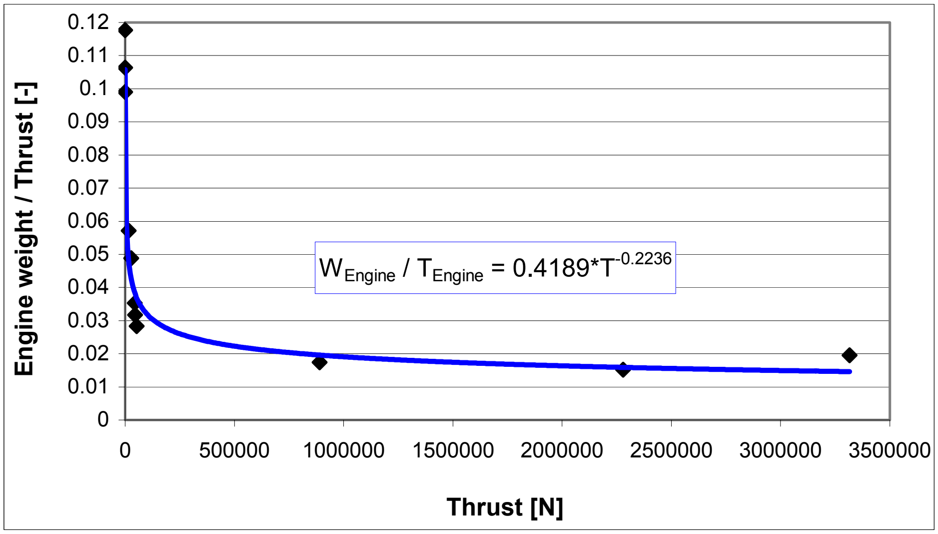

Figure 4: Engine weight to thrust ratio interpolation, reference engine data from [www.boeing.com, 2004; Northrop Grumman, 2004; www.spaceandtech.com, 2004]

These data points were interpolated with a simple power curve to provide a continuous function for engine sizing. With the initial thrust to weight ratio of the propulsion stage given, the engine mass can be related to the initial vehicle mass at the beginning of the maneuver:

$$m_{Engine} \approx \frac{T_{\max}}{g_0} \cdot 0.4189 \cdot \left(T_{\max}[N]\right)^{-0.2236} = \left(\frac{T}{W}\right)_{Vehicle,Initial} \cdot 0.4189 \cdot \left(T_{\max}[N]\right)^{-0.2236} \cdot m_{Initial}$$

The remaining dry mass of a propellant stage is comprised of tanks, plumbing and integrating structure. For conceptual design, this mass is assumed to be proportional to the fuel mass contained in the system:

$$m_{TankStructure} \approx 0.113 \cdot m_{propellant} = 0.113 \cdot \left(\rho_{Fuel} \cdot V_{Fuel} + \rho_{Oxydizer} \cdot V_{Oxidizer}\right)$$

The oxidizer and fuel volumes are coupled via the fuel to oxidizer ratio for the chemical reaction of the propellants:

$$\frac{m_{Fuel}}{m_{Oxydizer}} = \frac{F}{O}$$

This way, the tank and structure mass for the propulsion stage can be regarded as being solely a function of the fuel volume:

$$m_{TankStructure} \approx 0.113 \cdot m_{propellant} = 0.113 \cdot \left(\rho_{Fuel} \cdot V_{Fuel} + V_{Fuel} \cdot \frac{\rho_{Fuel}}{F/O}\right)$$

The landing gear mass for Moon or Mars landings is assumed to scale linearly with the mass that is landed. The proportionality factor can be determined with data from an actual design. With data given in [Gavin, 2002], the factor becomes:

$$m_{LG\,Moon} \approx 0.03 \cdot m_{Landed}$$

The landing gear mass is also assumed to scale linearly with the surface gravitational constant of the target planet. With this information, we can scale a landing gear up from Moon to Mars:

$$m_{LG\,Mars} = \frac{g_{Mars}}{g_{Moon}} \cdot 0.03 \cdot m_{Landed} \approx 0.07 \cdot m_{Landed}$$

For conceptual design, the mass for a drogue parachute on Mars is assumed to scale linearly with the parachute suspended mass. From a design example given in [Larson and Pranke 2002], we can derive the following relationship:

$$m_{Drogue} \approx 0.01 \cdot m_{Suspended}$$

Ablative heat shield masses are generally assumed to be a constant fraction of the shielded mass; a mass fraction of 20 % is regarded as a conservative choice [Larson and Pranke 2002, Messerschmid 2000]:

$$m_{HeatShield} \approx 0.2 \cdot m_{Shielded}$$

With these empirical and scaling relationships, it is possible to estimate spacecraft and component mass for short duration missions, which in turn enables the computation of optimal modular building blocks. The calculation of the vehicles mass is an iterative process.

The accuracy of the spacecraft mass calculation can be analyzed by comparing estimated values with actual spacecraft. The spacecraft selected for this comparison is the Apollo 17 Lunar Module (LM); Table 1 provides data on the Apollo 17 LM [NASA, 1972]:

Characteristic	**Value**
Crew Size	2
Crew Mass (2 crew)	160 kg
Mission duration	4 days
Sample mass to orbit	95 kg
Payload mass to surface	557 kg
Specific impulse of ascent / descent engines	311 s
Ascent velocity change	1874 m/s
Descent velocity change	2045 m/s
Descent velocity change with 20 % margin for hovering	2454 m/s
Maximum thrust / weight ratio descent	0.33
Maximum thrust / weight ratio ascent	0.5

Table 1: Apollo 17 LM characteristics [Gavin, 2002; NASA, 1972]

The actual duration of independent flight, i.e. the time from undocking to docking in LLO was 3.3 days for the Apollo 17 LM. The LM was powered up, however, during translunar coast, and also had spare lifetime for the event of a contingency in lunar orbit [NASA 1972]. A mission duration of 4 days therefore seems to be a realistic assumption. The sample and payload masses, as well as crew size and crew mass are well documented in [NASA 1972]. The thrust / weight ratios were calculated with the values for thrust given in [NASA 1972].
With these input parameters, we can calculate the masses of various LM components and compare them to the actual data of the Apollo 17 LM:

Component [kg]	**Mass calculated with model [kg]**	**Apollo 17 mass [kg]**	**Deviation [-]**
Crew compartment	2336	2427	-0.037
Ascent stage propellant	2538	2372	0.070
Ascent engine mass	114	91	0.253
Ascent stage (including crew)	**5275**	**4960**	**0.064**
Descent stage propellant	8968	8848	0.014
Descent engine mass	197	158	0.247
Landing gear	218	220	-0.009
LM total mass	**16228**	**16430**	**0.012**

Table 2: Components masses calculated with the empirical model in comparison the Apollo 17 LM [Gavin, 2002; NASA 1972]

The deviation in Table 2 was based on the Apollo 17 LM component masses. The deviation for the entire vehicle, as well as for the ascent stage is about 1 %, and well below 10 % for most components. This indicates excellent agreement with the existing design. The only components that are significantly overestimated in mass are the ascent and descent stage engines. The engines selected for landing system design in this analysis have to be restartable, and, in the case of cryogenics, might also feature additional propellant pumps and pressurization systems; a conservative engine mass estimation model therefore seems to be appropriate.

Mission Scenarios for Moon and Mars Landers

The prime motivation for design reuse in the context of manned Moon and Mars exploration is the reduction of overall Moon and Mars exploration architecture lifecycle cost by reducing development, test, and fixed production costs. This statement is based on the assumption that conceivable Moon and Mars mission scenarios and the exploration systems are already known.

The Vision for Space Exploration, however, states continuous and sustainable space exploration as the ultimate objective [President Bush, 2004]. Envisioned is a steady progression from one mission to the next, where the objective of every mission is determined by the knowledge gathered by the preceding one. In this context, it is uncertain what the most desirable mission sequence and crew sizes will be in the future. A so-called "mission type network", as illustrated in Figure 5 for manned Moon and Mars landers is intended to capture this uncertainty:

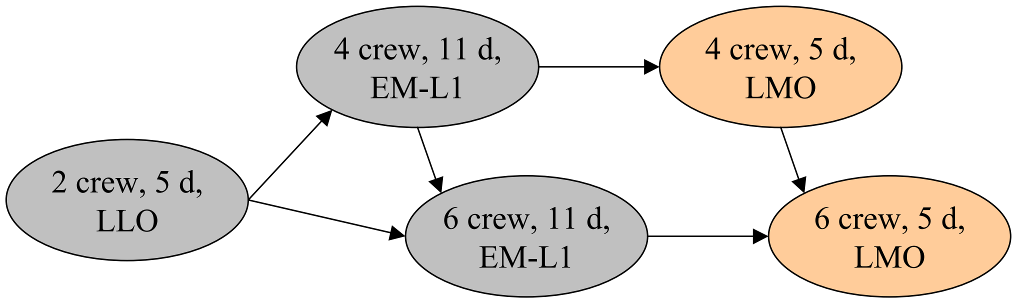

Figure 5: Simplified mission type network for manned Moon and Mars landers, (LLO: Low Lunar Orbit, EM-L1: Earth-Moon Lagrange Point 1, LMO; low Mars Orbit)

Shown are all the possible landers scenarios for a simplified campaign of Moon and Mars missions. The initial 2-crew Moon mission, and the final 6-crew long Mars mission are assumed to be the start and the end points. The arrows indicate possible follow-on missions depending on the results and findings of the preceding mission. There are several ways from the start to the end point through this network. The mission type network in Figure 5 does not preclude repeating a smaller, shorter mission even after a larger mission has been conducted.

In the case of uncertainty in the Moon and Mars exploration campaign planning, there is an even stronger motivation for commonality, because only design reuse can provide the flexibility in schedule and cost to meet the goal of sustainable long-term exploration.
The analysis presented in this paper is based on the 5 Moon and Mars lander mission scenarios illustrated in Figure 5; the associated velocity changes for the descending and ascending legs of the landers' trajectories are given in detail in Table 3:

Mission phase	Velocity change descent [m/s]
LLO to lunar surface	2083
Lunar surface to LLO	1871
EM-L1 to lunar surface	2746
Lunar surface to EM-L1	2746
LMO to Mars surface	625 (with heat shield and drogue parachute)
Mars surface to LMO	4000

Table 3: Data for Moon and Mars lander mission scenarios [NASA 1972, Larson and Pranke 2002, Farquhar 2003]

Modularization

There is general agreement that modular spacecraft are desirable because they allow tailoring of capabilities for particular missions by recombining basic building blocks. There is also general agreement that modular spacecraft and systems in general are less "optimal" or at least not as volume (packaging) and mass (launch cost) efficient as a collection of optimized point designs. The difficulty is in quantifying the "penalty" to be paid for modularity and to balance this inefficiency against the accrued benefits. The benefits stem from module reuse among missions. In this work we propose to quantify the penalty of modularity as the normalized difference in mass between a set of modular landers and their optimized, point-designed counterparts, cumulated over the set of landers required to execute the entire mission network shown in Fig.5. Formally, the generic penalty function can be expressed as follows:

$$f_{Penalty}\left(\beta_1,\beta_2,...,\beta_k\right)=\frac{\sum_i n_{Mission,i}\cdot w_{\Pr edeploy,i}\left[\underset{ModularDesign}{m_{Vehicle,i}\left(\beta_1,\beta_2,...,\beta_k\right)}-\underset{Po\operatorname{int}Design}{m^*_{Vehicle,i}}\right]}{\underset{Po\operatorname{int}Design}{\sum_i n_{Mission,i}\cdot w_{\Pr edeploy,i}\cdot m^*_{Vehicle,i}}}$$

β_k is the value of the functional attribute of the k-th building block (i.e. module); for an engine building block, for example, this would be the thrust. As the vehicles operate at different locations, there masses at those locations cannot be compared: a difference of one ton between modular and point design has a different effect in lunar orbit and in Mars orbit. The individual vehicle masses are therefore transformed in to IMLEO (Injected Mass in Low Earth Orbit) by means of the mass-multipliers $w_{Predeploy,i}$ in order to be comparable. The mass-multipliers represent the mass of predeploy propulsion stages. The values used here are shown in Table 4; they are based on liquid hydrogen / liquid oxygen propulsion with a specific impulse of 450 s. Please note that the multiplier for Earth-Moon-L1 predeployment also contains a conservative overhead for additional station-keeping and rendezvous maneuvers expected in the Earth-Moon-L1 halo orbit; additional analysis is necessary to understand the exact implications in terms of velocity changes of staging at the Earth-Moon-L1 point:

Predeployment to	Weighting Factor
Low Lunar Orbit	$W = 3.6$
Earth-Moon-L1	$W = 6.4$
Low Mars Orbit	$W = 9.9$

Table 4: weighting factors for predeployment of landing spacecraft to various locations; common reference point is LEO

Also, it is not expected that some lander missions will be executed more often than others; this can be taken into account by multiplying the individual vehicle masses with number of missions anticipated, $n_{Mission,i}$. As the exact mission sequence of a space program spanning decades is not yet known, we will assume all missions in the mission type network to be executed exactly once. In summary, we can say that the penalty function is designed to quantify mass penalties over the whole lifetime of the Moon and Mars exploration system. The effect of changes in the overall exploration plan on the penalty function can be captured by sensitivity analysis of the individual influence factors.

The mass penalty of modularization is due to two effects: if one building block has to be used to satisfy all possible requirements, it will generally not be possible to fulfill all of them exactly. The mass penalty is therefore caused by a surplus of functionality provided. The only exception to this observation arises, if all the requirements have a common denominator, and the building block value is this denominator. The second effect that causes modular spacecraft designs to be heavier than custom-designs is the additional mass for the interfaces needed to connect the modules to the remainder of the system.

For the following analysis, we assume that the spacecraft are used exactly once, and that they are assembled and tested on the ground. Therefore no heavy interfaces and reversible connections for in-space assembly are needed. Hence, the mass penalty due to module interfaces is assumed to be much smaller than the penalty caused by surplus functionality, and is subsequently neglected. For future analysis, the interface penalty could be incorporated by applying a certain mass overhead to the modules used.

We will now apply the above definition to specific design examples. First, we will analyze the modularization of only one component: the pressurized crew compartment. For this particular case, the penalty function takes the following form:

$$f_{Penalty}\left(V_{Pressurized,BB}\right)=\frac{\sum_i w_{Predeploy,i}\left[\underset{ModularDesign}{m_{Vehicle,i}\left(V_{Pressurized,BB}\right)}-\underset{Point\,Design}{m^*_{Vehicle,i}}\right]}{\underset{Point\,Design}{\sum_i w_{Predeploy,i}\cdot m^*_{Vehicle,i}}}$$

Before we start calculating the penalty function, it is necessary to apply constraints to the pressurized volumes eligible for the modular building block. The upper limit of the eligible pressurized volume is assumed to be the maximum pressurized volume needed for the point designs; for the mission type network of Figure 5, this is the 11 day, 6 crew mission to the lunar surface from EM-L1. Also, for geometric reasons, no more than three building blocks shall be used to provide the maximum pressurized volume required. This rule determines the lower boundary of the volumes eligible for the building block. Figure 6 shows the penalty function plotted over the pressurized building block volume. The constraints specified above are represented by vertical dotted lines; the eligible building block volumes are between the two constraints.

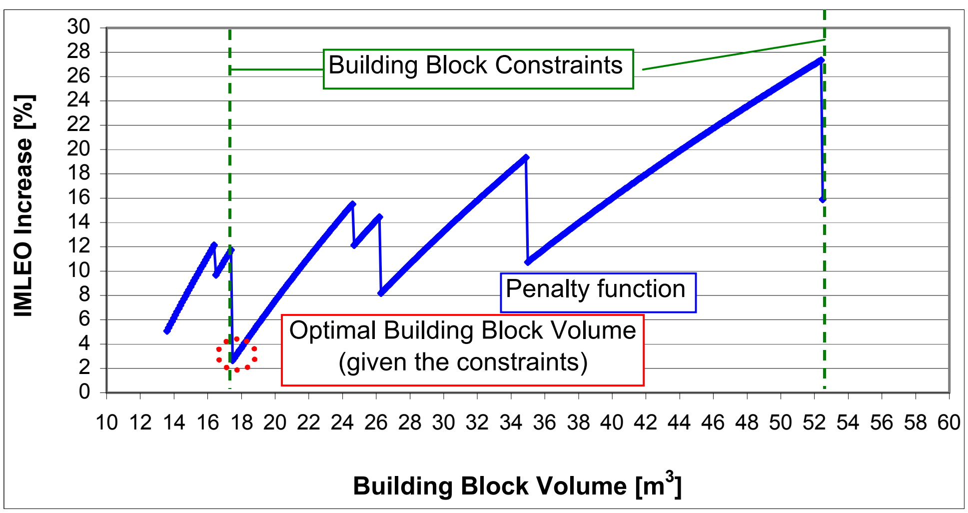

Fig. 6: Normalized mass penalty function (IMLEO increase) for the modularization of pressurized compartments

The penalty function in Figure 6 shows two key characteristics: firstly, it features several sharp drops. The drops occur, when the pressurized volume building block selected is a denominator of one of the pressurized volumes required by the mission type network.

Secondly, not counting the drops caused by arriving at a common denominator of one or more requirements, the penalty function generally increases with increasing building block volume. This is intuitively understandable: if we select a very small building block volume and use it in great numbers, we will be able to approximate the required volumes with only a small surplus; the bigger the building block, the more surplus we generate. The optimal building block volume, given the above constraints, is about 17.5 m^3 and causes an IMLEO increase of about 2.5 % (circled in red).

We will now extend the modularization to two other components: the rocket engine and the propellant management system, in addition to the pressurized volume structure. The penalty function for concurrent modularization of these three components can be expressed as follows:

$$f_{Penalty}\left(V_{Pressurized,BB}, T_{BB}, V_{Fuel,BB}\right) = \frac{\sum_i w_{Predeploy,i}\left[\underset{ModularDesign}{m_{Vehicle,i}\left(V_{Pressurized,BB}, T_{BB}, V_{Fuel,BB}\right)} - \underset{Point\,Design}{m^*_{Vehicle,i}}\right]}{\underset{Point\,Design}{\sum_i w_{Predeploy,i} \cdot m^*_{Vehicle,i}}}$$

The constraints for the pressurized volume are the same as for the single-modularization case above. For the engine / thrust building block, the upper constraint is the largest thrust for needed for the point designs, and the lower boundary by limiting the number of building block engines required to generate the maximum thrust to four. In the case of the fuel tanks, the upper boundary for the building block tank volume is defined by the maximum required volume needed for the point designs. The number of building blocks needed to contain the maximum fuel volume shall not exceed six. As the fuel and propellant volumes are coupled (see above), the oxidizer building block volume is determined by the choice of the fuel building block volume.
Table 6 shows the values for the optimal building blocks for pressurized volume, engine thrust, and fuel tank volume, as well as the minimum IMLEO increase (normalized mass penalty) given the above constraints.

Building block	**Size / Value**
Pressurized volume	17.973 m^3
Engine	147608 N
Fuel volume (methane)	2.882 m^3
IMLEO increase [%]	38.43

Table 5: Optimal modular quanta for modularization of pressurized compartments, engines and fuel tank volumes

As the modularization is taking place in three dimensions, the penalty function can no longer be displayed in a single diagram. If we select the optimal building block for two components, however, we can plot the penalty function over different values of the third building block. This yields the three curves that are shown in Figures 7 – 9.

The results indicate that for the modularization of more than one subsystem, large mass penalties (about 38 %) can arise, even in the case of negligible interface masses. Furthermore, it is apparent that for the modularization of the fuel tanks the penalty function varies only between 40 – 60 %, and for the engines stays close to 40 %, whereas for the modularization of the

pressurized volumes, it varies between 40 – 80 %. This indicates that the pressurized volume surplus has the strongest impact 'gradient'.
Future quantitative analysis will determine the sensitivity of the penalty function to changes in the mission type network, an in the number and type of modularized subsystems components. The method presented above provides the basis for this analysis.

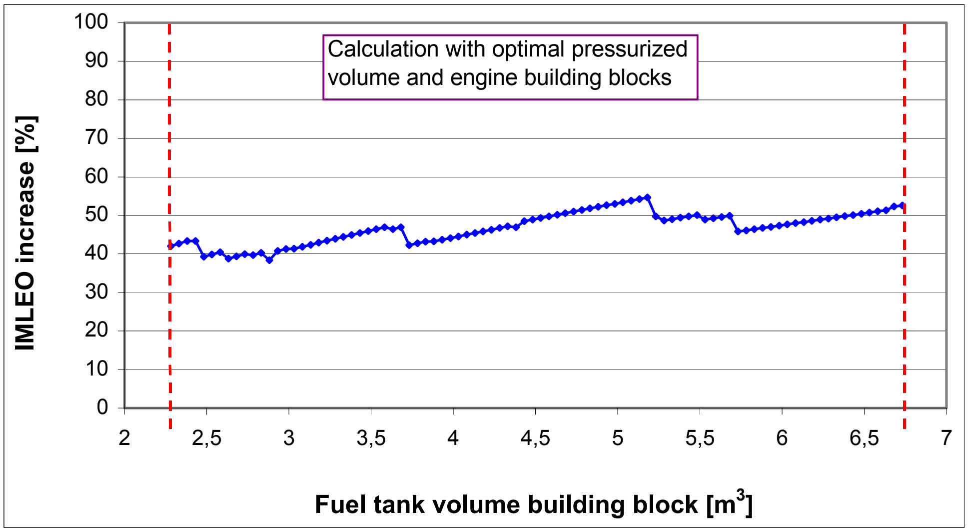

Figure 7: Normalized mass penalty (IMLEO increase) as a function of fuel tank building block volume; the engine building block thrust and the pressurized volume building block are held constant at their optimal values

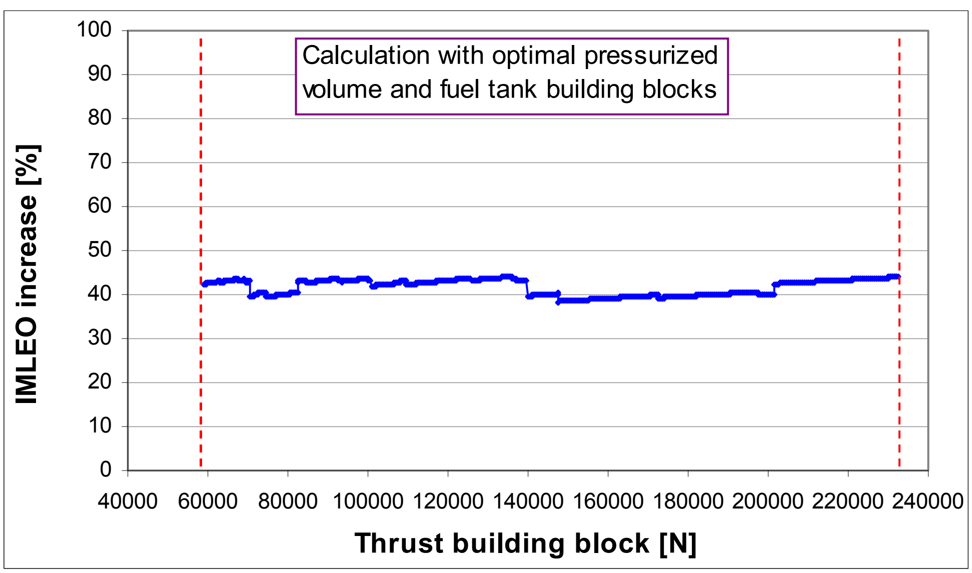

Figure 8: Normalized mass penalty as a function of engine building block thrust; the fuel tank building block volume and the pressurized volume building block are held constant at their optimal values

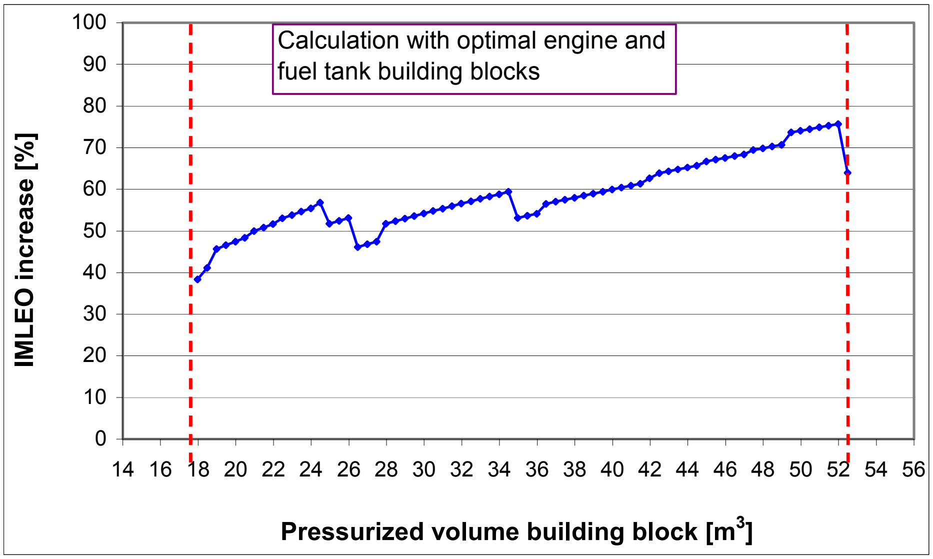

Figure 9: Normalized mass penalty as a function of pressurized volume building block; the fuel tank building block volume and the engine thrust building block are held constant at their optimal values

Figure 10 shows a set of possible conceptual cross-sections (approximately to scale, Apollo LM for comparison) of the manned Moon and Mars landers for the missions specified in the mission type network above. The Mars landers are presented in the actual Mars landing configuration, i.e. without parachutes and heat shield. Please note that the overall vehicle height seems to be a challenge for crew operations in the case of the vehicles utilizing EM-L1 as a staging point. This could be mitigated by not stacking the tank modules vertically, but adding them laterally to the vehicles.

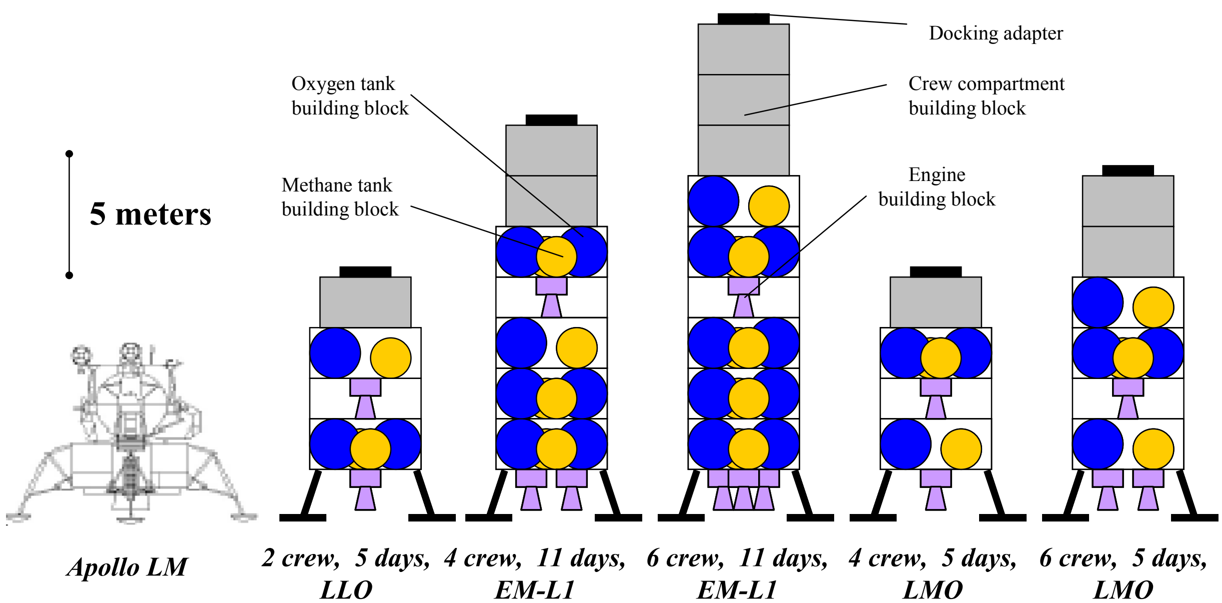

Figure 10: Overview of modularized Moon and Mars lander designs for the mission type network of Figure 5 [Gavin 2002]; Mars heat shields are not shown

Conclusions

Results. A straightforward process for the identification of optimal modular building blocks for manned Moon and Mars landers was described. The process is based on the assumption that the subsystems in question are going to be modularized; it can then provide the suite of modular quanta with the lowest mass penalty. Spacecraft wet mass serves as a surrogate metric for cost.
For every subsystem, one modular building block type was employed[1], which can be used $n_{ss,}i$ times to provide the functionality needed. The process steps through all combinations of building blocks for the modular subsystems, and calculates the mass penalty for each combination relative to point design solutions. With the penalties known, the building block combination with the lowest additional mass can be identified.

For the crew compartment structure modularization, the wet mass (IMLEO) overhead given the above building block volume constraints was found to be 2.5% of the overall mission type network lander mass, which seems to be small even compared to the design uncertainty expected at this level of system resolution. For the modularization of pressurized structures, propellant storage systems, and engines, the minimum IMLEO overhead was found to be about 38 %, which is considerable. From Figure 10, however, it can be seen that eliminating missions utilizing the EM-L1 point are driving the number and size of modules. Elimination of these missions from the mission type network is therefore expected to lower the IMLEO overhead, because the remaining requirements could be approximated more closely.

In order to allow a full-factorial analysis of multiple design parameters, a simplified model has been employed that does not specifically include the mass overhead for module interfaces. However, as the number of modules is small in all cases, and as the vehicles described here are assumed to be assembled on Earth and to be launched integrally, the additional mass overhead seems to be comparatively small compared to the module masses, and the simplification therefore justified. This assumption would no longer hold in the case of on-orbit assembly with the requirements of module docking and berthing interfaces.

The process described here can be employed for any finite number of modular subsystems; as the number of combinations is multiplicative, however, the computation for the mass penalties can become quite time-consuming. If subsystems are decoupled from others in the system design in question, they can be modularized independently. The penalty for this modularization can then be displayed in a diagram as a function of the building block size, as illustrated above for the case of the pressurized volume.

Multiple building blocks for one subsystem. So far, it was assumed that for every subsystem only one building block is selected, and for providing different levels of functionality, different numbers of the same building block are used. For a short-duration mission with a small crew, one building block can provide all the necessary pressurized volume. For a mission with longer duration and additional crewmembers, three building blocks are connected to provide the additional volume needed.

[1] Larger landers require multiple of these building blocks in the same instantiation of the system.

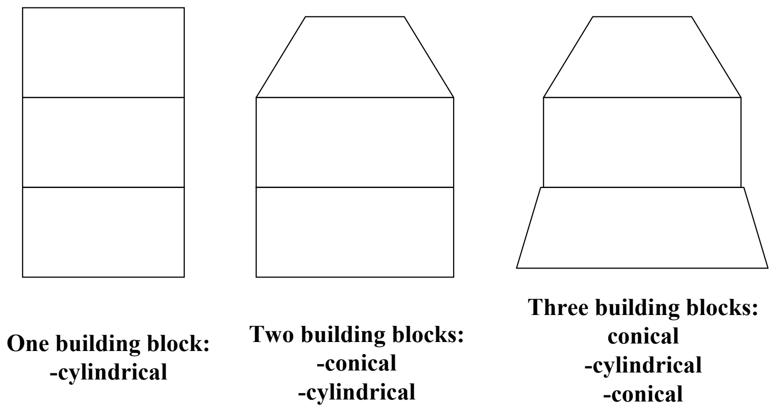

Figure 11: Crew compartments with multiple pressurized volume building blocks

It is also possible to use more than one building block to provide the subsystem functionality: Figure 11 illustrates that the same pressurized volume can be provided by a stack of identical cylindrical modules, a combination of two cylindrical and a conical module, or a by an assembly of three different building blocks. Increasing the number of modular building blocks for one function introduces additional degrees of freedom into the system design. If we take this concept to the extreme, we could introduce a building block size for every requirement for a specific functionality, and would thereby arrive at the point design solutions. It is therefore intuitively clear that additional building blocks for the same functionality decrease the modularization penalty, because less surplus functionality will be provided. Every additional building block, however, represents also an increase in development cost and system complexity. The process described can contribute to find the optimum number of building blocks to provide one function by assessing the reduction of the penalty metric.

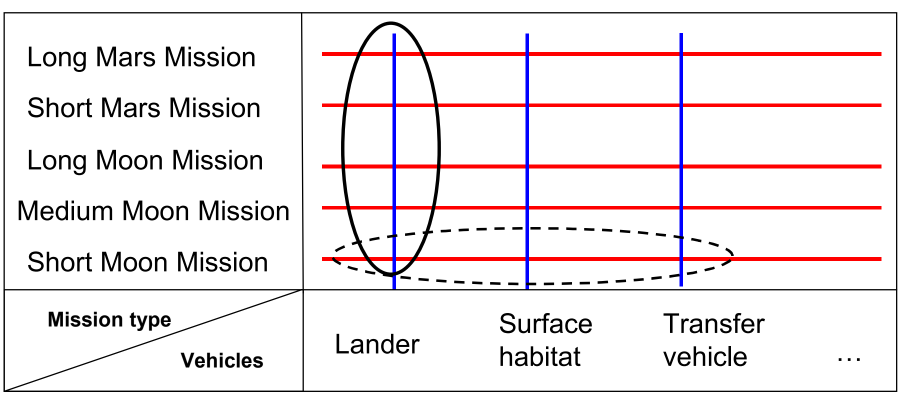

Figure 12: Commonality between missions and architectures for planetary exploration

Modularization across the elements of an exploration system architecture. The example cases for modularization described so far were all set in the context of maximizing commonality between vehicles (landing systems) with identical overall functionality: to transport human beings from a staging point in the vicinity of the of the destination planet (Moon, Mars) to the surface, and back. As the Moon and Mars landers described have comparable overall velocity changes and crew-days assigned, it is likely that they employ the same technology to provide subsystem functionality. This approach is indicated in Figure 12 by the solid vertical ellipse: commonality between vehicles of the same type for different missions types. There is, however, another possibility for commonality. For Moon and Mars exploration systems, usually several manned vehicles are needed: a lander, a surface habitat, a transfer vehicle, etc. If all the vehicles use the same technology to provide a certain subsystem function (for example fuel cells for power generation), this functionality could be provided by a common building block.

Summary. The process described can be employed for the modularization of any system, provided a model (analytical, numerical) is available to assess the cost penalties of the modularization. The primary objective for using the process is to provide a basis for platforming decision-making by supplying information about the penalties. The savings induced by reduced development, test, and fixed production cost, and the additional launch and variable (per unit) production cost [Enright, Jilla, Miller, 1998] need to be assessed independently.

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