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MODELING THE MARS-EARTH LOGISTICS AND TRANSFER ARCHITECTURE: FOCUS ON CONDUCTING INEXPENSIVE EXPLORATION THROUGH VARIANCE OF KEY MISSION REQUIREMENTS

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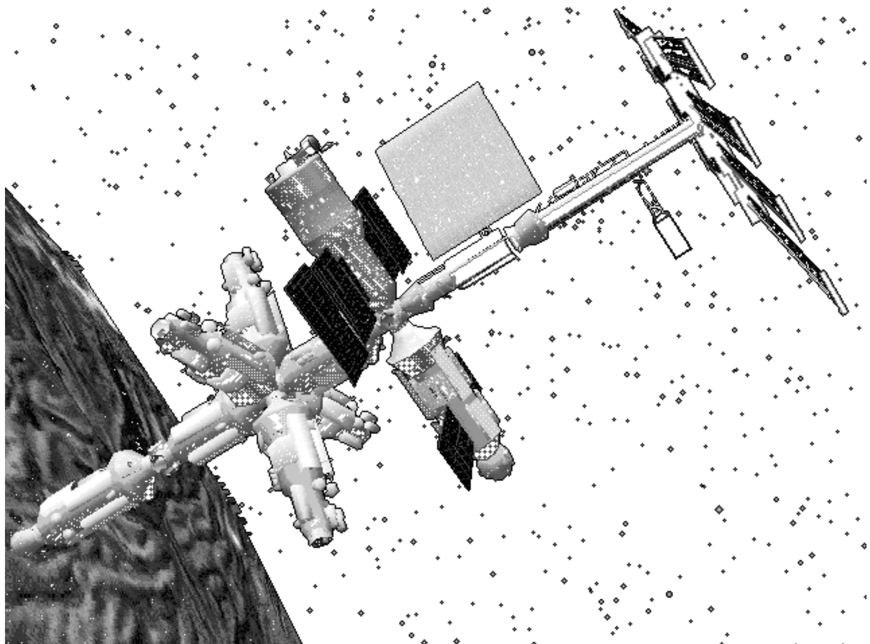

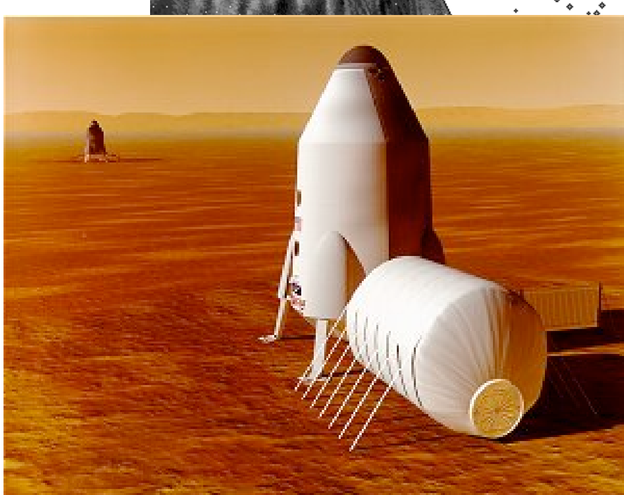

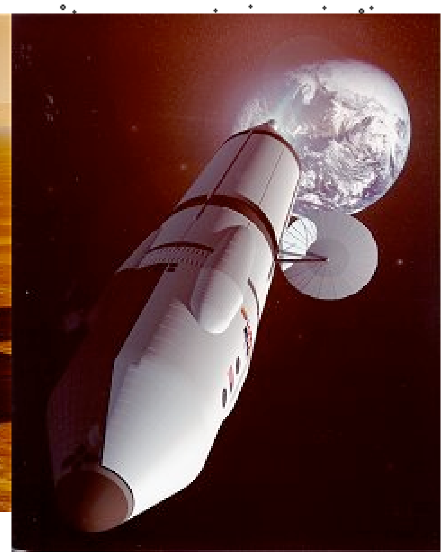

ABSTRACT

This paper addresses a concept-level model that produces technical design parameters and economic feasibility information addressing future Mars Exploration platforms. In this context, the platforms considered include those currently chosen in the NASA Mars Design Reference Mission.

This paper uses a design methodology and analytical tools to create feasible concept design information for these space platforms. The design tool has been validated against a number of actual facility designs, and appropriate modal variables are adjusted to ensure that statistical approximations are valid for subsequent analyses. The tool is then employed in the examination of the impact of various payloads on the power, size (volume), and mass of the platform proposed.

The development of the analytical tool employed an approach that accommodated possible payloads characterized as simplified parameters such as power, weight, volume, crew size, and endurance. In creating the approach, basic principles are employed and combined with parametric estimates as necessary. Key system parameters are identified in conjunction with overall system design. Typical ranges for these key parameters are provided based on empirical data extracted from actual human spaceflight systems.

In order to provide a credible basis for a valid engineering model, an extensive survey of existing manned space platforms was conducted. This survey yielded key engineering specifications that were incorporated in the engineering model. Data from this survey is also used to create parametric equations and graphical representations in order to establish a realistic range of engineering quantities used in the design of manned space platforms.

Using this tool a sample Mars exploration architecture is formulated with emphasis on cost minimization through variance of key mission requirements. This paper is based on work Dr. Reynerson recently completed at George Washington University in fulfillment for the degree of Doctor of Science in Astronautics. Dr. Mike Griffin, former head of NASA's Human Mars Mission, was a member of the dissertation committee.

INTRODUCTION

The design of human spaceflight systems is not a field that has an overabundance of design examples. But from over 40 years of experience there exists enough quantitative data on which to attain initial estimates for future designs. On this premise the following paper is written. Prior work has been done in this area relative to space station designs. In the doctoral dissertation work performed by Dr. Charles Reynerson an engineering model was created for preliminary space station design. The model is general enough in that it can be used for any spacecraft, regardless of it's purpose. Typical spacecraft with only sensors for payloads would be a specialized case of the model. The generalized model incorporates the addition of humans and their associated life support and habitation systems.

It is the intent of this paper to show that the same model can be used for elements needed for human transportation and settlement on extraterrestrial bodies. This model can be easily adapted to have applicability to elements such as transportation vehicles, non-earth orbiting space stations, and surface habitations. The model also provides a rough cost estimate for future manned missions.

As an example, the NASA Mars Design Reference Mission (DRM) will be used as a baseline architecture which will be perturbed by altering the desired payload amount during transfers and landings. This variation will show the impact of entire system mass required as well as system cost.

HUMAN SPACE SYSTEM MODEL

A human space system model was created that is composed of both an engineering model and a cost model. This model treats two basic types of payloads: humans and space hardware (i.e. sensors, communications, manufacturing,…). The model flow is shown in Figure 1. Five inputs are put through the engineering model and spacecraft power, mass, and volume are output. The cost model uses spacecraft weight as an input since it tends to drive about 80% of cost on typical space systems.

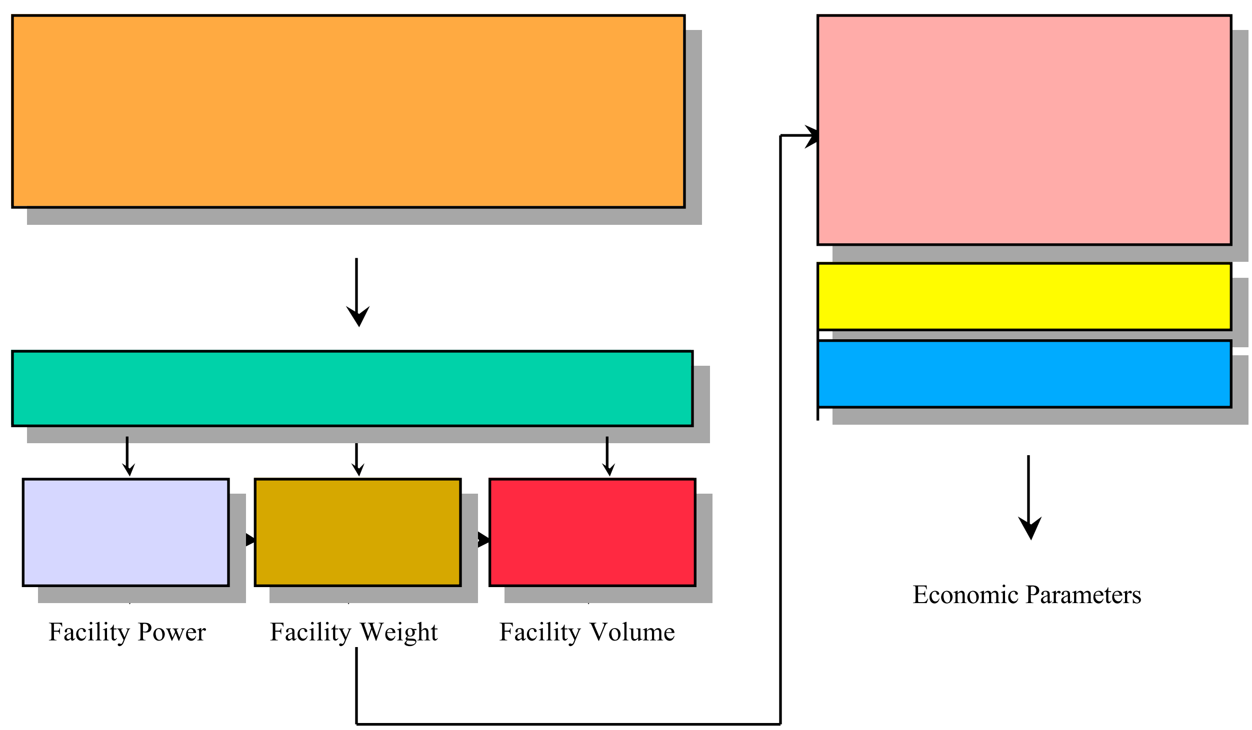

Fig 1. Modeling Flow Diagram

ENGINEERING MATH MODEL GOVERNING EQUATIONS

The inputs to the model are the following variables:

W_p = payload power (payload being defined as space rated hardware and equipment used to create revenue for the space business park)

V_p = payload volume

P_p = payload user power (most commonly referred to as user power on space stations)

N_c = number of crew members

E_c = designed endurance limit for the crew. This time factor will also be the assumed resupply interval for consumables calculations.

Assume the outputs to our model are the following variables:

W_f = facility weight
V_f = facility volume
P_f = facility power

Assume that the output variables are some linear combination of the input variables. In reality the input variables may be raised to some arbitrary power as follows.

$$
\begin{aligned}
W_f &= f(W_p, V_p, P_p, N_c, E_c) = \alpha \cdot W_p^a + \beta \cdot V_p^b + \chi \cdot P_p^c + \delta \cdot N_c^d + \varepsilon \cdot E_c^e \\
V_f &= g(W_p, V_p, P_p, N_c, E_c) = \phi \cdot W_p^f + \varphi \cdot V_p^g + \gamma \cdot P_p^h + \eta \cdot N_c^i + \iota \cdot E_c^j \\
P_f &= h(W_p, V_p, P_p, N_c, E_c) = \kappa \cdot W_p^k + \lambda \cdot V_p^l + \mu \cdot P_p^m + \nu \cdot N_c^n + o \cdot E_c^o
\end{aligned}
\qquad (1 - 3)
$$

If a linearized form of equations 1 through 3 are used then the following approximations can be made:

$$
\begin{aligned}
W_f &\cong \alpha \cdot W_p + \beta \cdot V_p + \chi \cdot P_p + \delta \cdot N_c + \varepsilon \cdot E_c \\
V_f &\cong \phi \cdot W_p + \varphi \cdot V_p + \gamma \cdot P_p + \eta \cdot N_c + \iota \cdot E_c \\
P_f &\cong \kappa \cdot W_p + \lambda \cdot V_p + \mu \cdot P_p + \nu \cdot N_c + o \cdot E_c
\end{aligned}
\tag{4 - 6}
$$

In matrix form:

$$
f = \mathrm{A} \cdot p \qquad \text{where} \quad f = \begin{pmatrix} W_f \\ V_f \\ P_f \end{pmatrix}, \quad \mathrm{A} = \begin{bmatrix} \alpha & \beta & \chi & \delta & \varepsilon \\ \phi & \varphi & \gamma & \eta & \iota \\ \kappa & \lambda & \mu & \nu & o \end{bmatrix}, \text{ and } \; p = \begin{pmatrix} W_p \\ V_p \\ P_p \\ N_c \\ E_c \end{pmatrix} \tag{7}
$$

The above equations form the basis for the engineering concept model.

In reality the components of matrix ***A*** will vary depending on the payload input values in vector ***p***. Thus the system is not truly linear; nonetheless, we will approximate it as such. Each component of ***A*** will be discussed in the following sections.

When focusing on the meaning of each component in relation with the physical system some of the ***A*** matrix coefficients can be neglected. For the facility volume calculation, the payload weight term does not affect the payload volume term if they are assumed to be independent. Similarly, in the facility weight calculation the payload volume term is independent of the payload weight term. For the facility power calculation, only the payload power and crew number terms are assumed to have an effect on payload power. Therefore, the ***A*** matrix becomes,

$$
\boldsymbol{A} = \begin{bmatrix} \alpha & 0 & \chi & \delta & \varepsilon \\ 0 & \varphi & \gamma & \eta & \iota \\ 0 & 0 & \mu & \nu & 0 \end{bmatrix}. \tag{8}
$$

Through careful accounting of typical spacecraft weight, volume, and power budgets the above assumptions can be shown to have validity.[2]

FACILITY WEIGHT

The weight of the facility can be considered in terms of its basic segments. The most general division is that of payload and bus. The payload in this case is considered to be both space hardware and people. The spacecraft bus must be designed to support both the hardware and the crew. Since the effect of varying the amount of space hardware and crew on the facility is to be considered, the bus is broken into two portions: that which supports the space hardware, and that which supports the crew. The third major weight category is consumables. This can also be subdivided between consumables for the spacecraft bus and consumables for the crew. The facility weight equation is given by:

$$W_f \cong \alpha \cdot W_p + \beta \cdot V_p + \chi \cdot P_p + \delta \cdot N_c + \varepsilon \cdot E_c \, . \qquad (9)$$

The corresponding terms are defined as followed:

$\alpha \cdot W_p$ = payload weight + payload bus support weight - power system weight required to support payload

$\beta \cdot V_p = 0$

$\chi \cdot P_p$ = power system weight required to support the payload

$\delta \cdot N_c$ = spacecraft subsystem weight required to support the crew (includes power) and crew weight

$\varepsilon \cdot E_c$ = weight of consumables for the crew and the facility.

The second term is assumed to be zero since the payload effect is included in the first term. This might be useful if the assumed payload density is not comparable to current systems. The above equation coefficients can be shown to relate to actual spacecraft weight budgets.

FACILITY VOLUME

For the volume equation, the terms are analogous to those of the weight equation except for the fourth term. For the crew members, there is an additional volume allocation for habitability. This was neglected in the weight equation since the weight of the air within that volume is small relative to the other terms. The facility volume equation is given by:

$$V_f \cong \phi \cdot W_p + \varphi \cdot V_p + \gamma \cdot P_p + \eta \cdot N_c + \iota \cdot E_c \qquad (10)$$

where the corresponding terms are defined as followed:

$\phi \cdot W_p = 0$

$\varphi \cdot V_p$ = payload volume + payload bus support volume - power system volume required to support payload

$\gamma \cdot P_p =$ power system volume required to support the payload

$\eta \cdot N_c =$ spacecraft subsystem volume required to support the crew (includes power) + crew habitable volume allocation

$\iota \cdot E_c =$ volume of consumables for the crew and the facility

The first term is zero since the payload effect is accounted for in the second term. This might be useful if the assumed payload density is not comparable to current systems. The above equation coefficients can be shown to relate to actual spacecraft volume

FACILITY POWER

The facility power equation is given by:

$$P_f \cong \kappa \cdot W_p + \lambda \cdot V_p + \mu \cdot P_p + \nu \cdot N_c + o \cdot E_c \tag{11}$$

where the corresponding terms are defined as follows:

$\mu \cdot P_p =$ power required for supporting the payload and payload support systems

$\nu \cdot N_c = P_c =$ power required to support crew related subsystems and facility support systems

$\kappa \cdot W_p = \lambda \cdot V_p = o \cdot E_c = 0$.

The first (payload weight) and second (payload volume) terms are not needed since the third term, represented by the payload power, gives the direct relation for the payload. The last term is applicable if the power subsystem design strategy is to treat power as a commodity to be resupplied at various time intervals. An example may be fuel cells, non-rechargeable batteries, or short-lived nuclear power sources. For this model we will assume conventional photovoltaic or solar dynamic systems are used and that sufficient design margin is used to compensate for power system efficiency degradation over time.

The above equation coefficients can be shown to relate to actual spacecraft power budgets. The above equations form the basis for the engineering concept model. Details of this model and its validation are beyond the scope of this paper but can be found in reference [4].[3] Internal to the model, the coefficients of the governing equations can be related to 14 key system parameters that are unique to each system. These parameters are presented in Table 1 along with their typical range for several human spaceflight designs.

Internal Model Variable	Typical Range	Average Value	Physical Meaning
ν, kW/person	0.5 to 3.4	2.771	Crew Specific Power
ζ	0.19 to 0.36	0.301	Bus to Payload Power Ratio
α_1	1.3 to 2.2	1.5	Powerless Bus to Payload Weight Ratio
δ_{cs} , kg/person	1573 to 18662	11281	Crew System Specific Weight
χ , kg/kW	75 to 317	190.69	Power System Specific Weight
ε_a, kg/person-day	2.6 to 32.2	9.48	Crew Member Consumption Rate
ε_p, kg/day	5.6 to 80.0	29.29	Propellant Consumption Rate
ρ_p, kg/cu. m	69.3 to 461.5	296.1	User Payload Density
ρ_{cs}, kg/cu. m	64.6 to 317.9	229.2	Crew System Density
ρ_b, kg/cu. m	25.8 to 202.2	115.5	Bus Density
η_h, cu. m/person	1.1 to 120	66.1	Crew Habitation Specific Volume
η_{cg}, cu. m/person	0.03 to 0.4	0.2	Crew Gear Specific Volume
ι_a, cu.m/person-day	0.007 to 0.040	0.020	Crew Member Volumetric Consumption Rate
ρ_{pc}, kg/cu. m	70 to 1268	721	Propellant Density

Table 1. Key System Parameters Internal to Engineering Model

COST MODEL

A cost model was created to complement the engineering model. For a commercial development cost tends to be the bottom line and therefore should be addressed. Various cost factors that result from space facility designs and an estimation of rough order of magnitude cost are included in this cost model. The model consists of two main sections: required investments and revenues. The required investment areas addressed include the space segment, launch vehicles, operations, and logistics. The revenues considered include crew and user payload related revenues.

Space Segment Cost

The space segment is modeled using the product of four variables. The space segment cost factor (Scf) is the price per kg of facility on orbit. This value typically varies from 58 to 148 $K/kg for unmanned spacecraft.[4] For manned space programs (Note that these values are for government run programs) the range is 38 to 157 $K/kg and the mean is 104 $K/kg. The program cost is normalized over the number of manned vehicles produced. The low number in the Skylab program is likely due to less research and development required since it was derived from the Apollo program.

The research, test, development, and engineering (RTD&E) cost factor (Rcf) is used to compensate for new development cost. RTD&E cost tends to be about three times that of the theoretical first unit (TFU) cost. For manned systems this would make the Scf range from 22 to 52 $K/kg for the TFU if you assume all of the programs were pure RTD&E cost (not including Skylab). Assuming this range for the TFU, then the Rcf should be 3 for new development programs, 1 for a program based on existing hardware, or somewhere in between if there is partial development required.

The space segment cost (Sc) can now be defined by the following equation:

$$S_c = S_{cf} \cdot P_{cf} \cdot R_{cf} \cdot W_f \qquad (12)$$

Launch Vehicle Cost

The launch cost factor (Lcf) can be estimated using historical data and planned cost goals for future developments. Launch vehicle costs[5] for several competing launch systems range from 4.4 to 57.4 $K/kg. The average cost is around 15.2 $K/kg.

An insurance cost factor (Icf) is used to account for insurance cost related to launch. Typically for commercial launches, insurance runs about one third of the launch cost. The Icf would therefore be a value of around 1.33.

The launch cost (Lc) for delivering the facility to orbit can now be defined by the following equation:

$$L_c = L_{cf} \cdot I_{cf} \cdot W_f \qquad (13)$$

Ground Operations and Support

The cost for the ground equipment is typically much smaller than the cost needed for the space segment and launch. But the operations for the ground stations becomes significant over time and should be considered in the cash flow calculations to counter the yearly revenues. Operations, mission, and program support costs for the Skylab program[6] average (over 4 years) $31.6 M in 1970's dollars. This cost is roughly $83 M in 1997. The International Space Station program has $13 B in its operations budget over 10 years for an average of $1.3 B per year.[7] This figure likely includes logistics costs for delivery of consumables and maintenance costs to upkeep the facility over its ten-year design life. It may also include the RDT&E for future payloads, experiments, and support. For the purposes of this ROM cost model, a figure of $80M per year is used for yearly operations and support costs (Yosc). A ten-year operational period (Ny) is assumed for life cycle costing purposes.

Logistics:

To account for the delivery of people, payloads, consumables, and products to and from the facility, a yearly logistics cost is calculated based on weight delivered and launch cost. A logistics crew specific weight (δ_{cs}) is defined as the equipment weight needed for crew support during the trip to and from orbit. This value should be no more than the crew specific weight for space facilities due to much shorter duration on orbit. The value for crew specific weight can be estimated from previous manned missions (1500 kg/person for Mercury to 11030 kg/person for Apollo). The Apollo crew specific weight is high due to the stressing requirements to go to the moon. The Space Shuttle crew specific weight is high due to its design to accommodate a heavy lift payload.

For this cost model a nominal value of 2000 kg/person is assumed for logistics crew specific weight, which is just higher than a Gemini capsule. Equation 14 defines the yearly crew logistics weight (Wcl) including consumables. This model assumes resupply intervals to be that of the endurance interval, Ec.

$$W_{cl} = 365 \cdot \left((\delta_{cs} + \delta_{crew} + \delta_{cg}) \cdot \frac{N_c}{E_c} + \varepsilon \right) \tag{14}$$

Here δ_{crew} and δ_{cg} are the crew system specific weights for the crew itself and their gear, respectively. ε is the consumable consumption rate for the entire facility.

For user payload logistics, a yearly turnover fraction (Tf) is defined as that fraction of total payload weight that is replaced during the year. This term is more useful for permanent facilities. If the payload requires a certain amount of production materials delivered, then a materials weight fraction (Mf) is used. The Mf is defined as that fraction of equivalent payload weight is required per year for payload production needs. The value for Mf is highly dependent upon the payload mission. For a tourism mission it might be zero and for a materials processing facility it could be more than 100%. The value for Mf is also mission dependent and very much market driven. If payloads have a nominal life of 5 years, then the turnover rate would be 100% in ten years. Therefore the yearly turnover rate would average 10%. The yearly user payloads logistics weight (Wupl) is then defined by the following:

$$W_{upl} = W_p \cdot \left(M_f + T_f\right) \tag{15}$$

To account for maintenance materials required for the facility, a maintenance materials weight fraction (Mmf) is established. This term like the previous term is also more useful for permanent facilities. The Mmf is that fraction of the facility weight that is required to be replaced each year. A nominal value of Mmf = 0.01 is assumed for this model. The maintenance materials yearly delivery weight (Wmm) is therefore:

$$W_{mm} = W_f \cdot M_{mf} \tag{16}$$

The total yearly logistics weight is the sum of equations 14, 15, and 16 as follows:

$$W_l = W_{cl} + W_{upl} + W_{mm} \tag{17}$$

The yearly logistics cost is similar to equation 13 but based on logistics weight:

$$L_{gc} = L_{cf} \cdot I_{cf} \cdot W_l \tag{18}$$

The total life cycle operations and support cost (Osc) including ground and logistics is then:

$$O_{sc} = N_y \cdot \left(Y_{osc} + L_{gc}\right) \tag{19}$$

The total investment required over the facility life is then:

$$TI = S_c + L_c + O_{sc} \tag{20}$$

Table 2. Some Key Parameters for the Transit Surface Habitat

Internal Model Variable	Typical Range	Average Value	Transit Surface Habitat	Physical Meaning
ν, kW/person	0.5 to 3.4	2.771	3.18	Crew Specific Power
ζ	0.19 to 0.36	0.301	0.49	Bus to Payload Power Ratio
α_1	1.3 to 2.2	1.5	1.0	Powerless Bus to Payload Weight Ratio
δ_{cs}, kg/person	1573 to 18662	11281	3833	Crew System Specific Weight
χ, kg/kW	75 to 317	190.69	99.13	Power System Specific Weight
ε_a, kg/person-day	2.6 to 32.2	9.48	10.2	Crew Member Consumption Rate

APPLICATION TO HUMAN MARS MISSIONS

This model can be applied to most vehicles that uses humans as payloads. The initial application for this model was space stations. An evaluation of NASA's X-15 high speed test vehicle was conducted using this model. Some minor modification was required. The concept model was within 22% of the actual weight, which is acceptable for concept level designs.

For the NASA Mars Design Reference Mission (DRM), this model can be applied to the transfer habitat, surface habitats, ascent/descent vehicles, and the Earth Return Vehicle (ERV). By using the weight and power budget data in reference 7, Table 2 shows some of the key system parameters for the Transit Surface Habitat design. These system parameters were found using the engineering model described above. Of the six parameters calculated, two are outside of the typical range for typical human space system designs. The first, the bus to payload power ratio, is somewhat higher than the typical range. This could mean that the bus power was overestimated, the power required for the payloads was underestimated, or some of both. The second, the powerless bus to payload weight ratio is somewhat lower than the typical range for typical human space system designs. This could mean that the powerless bus weight was underestimated, the payload weight was overestimated, or some of both. The powerless bus weight is the weight of the bus less that of the power subsystem.

Comparing the NASA DRM to Apollo

As one looks to the one big past human exploration mission to another celestial body, Apollo, it is enlightening to see the enormity of the mission. Table 3 compares the net mass of the two missions at various stages. To keep the missions normalized for consistency, each one assumes 12 flights to the destination and back.

Table 3. Mass Comparison between the Mars DRM and Apollo

Mass Characterized (metric tons)	NASA Mars DRM	Apollo	Multiplication Factor
Total System	62340 (est.)	17684	3.5
Delivered to LEO	2790	817	3.4
Delivery to Body Orbit	1254	274	4.6
Delivery to Body Surface	560	41	13.7
Delivered from Body Surface	14	14	1
Returned to Earth	28	35	0.8

One may ask why there is such a big difference in these masses when compared. The return mass to earth is more for the Apollo missions since there were men returned to earth with each mission. For the Mars DRM only three of the twelve missions are piloted for a total of 18 people as opposed to Apollo returning 36 people (that is, if we had really done 12 missions). If we were to normalize by number of people then the multiplication factors would be even higher than shown in the table.

The mass delivered from the body surfaces matches but the Apollo missions would have lifted 24 persons from the surface in 12 missions. The difference in mass per person is likely due to a more efficient packing factor for the Mars DRM. The Apollo LEM was designed to lift two at a time while the Mars Ascent Vehicle is designed to haul six. The efficiency occurs since the MAV needs less support system mass per person.

The mass delivered to the body surface differs by more than an order of magnitude. Perhaps this can be accounted for by the length of stay required on the surface and the desired redundancy needed for such a remote location. On the other hand, a closer look should be performed to see if an unreasonable amount of material is being transported to the surface. The mass delivered to Mars orbit should be somewhat greater than the Apollo missions since the transfer vehicle and Earth Return Vehicle require a larger delta-V for TEI as compared to a trip from the moon.

The total system mass and the mass delivered to LEO have about the same multiplication factor of 3.5. This is just the difference in mass between going to Mars (using the NASA DRM methods and technologies) as compared to the moon (using Apollo's methods and technologies). Note that the DRM total system mass is estimated and based upon using a system similar to the Saturn V for delivery to a LEO parking orbit. This is somewhat of a disturbing figure to most aerospace engineers who realize that cost can be linked to around 80% of the system mass. For the Apollo scaled comparison realize that only 6 missions actually landed on the moon and returned to the Earth but 32 units were produced. In estimating mission cost it should be somewhat less than Apollo if we account for the learning curve factor of human spaceflight knowledge over the last 40 years. Perhaps we would also have less test flights prior to an actual landing mission.

For a rough cost estimate it should be more accurate to use a specific cost number for the International Space Station (ISS) of $66,000 per kg (Apollo was $157,000 per kg). Using the mass delivered to LEO for the DRM, the estimated cost is $165.5 B (FY 97 dollars). This amount is very close to the Apollo program cost of $166.8 B in FY97 dollars. So the paramount question is: “Can we afford to do another Apollo-sized program?”

REDUCING THE COST TO GO TO MARS

The following modifications could be made to the NASA Mars DRM in order to save a significant amount of infrastructure cost.

A. Use of existing transportation systems. To avoid the excessive cost of creating a new launch vehicle system, the approach assumed is to use existing transportation systems to deliver modules into a low Earth orbit (LEO) prior to Trans-Mars injection (TMI). This approach allows existing docking and maneuvering techniques to be used as well as the existing ISS infrastructure should there be integration or start-up problems. Such problems are best dealt with prior to TMI, when there is a better chance to recover from serious problems.
   During return trips the intermediate destination is in LEO. Payloads can be retrieved in LEO by the Space Shuttle, then subsequently returned to earth.
B. Reusable transfer vehicle designs. The transfer vehicle is designed such that it can be used more than once. This lends itself to producing less hardware overall that would be more reliable than a single trip vehicle. It also would be a huge step in creating a permanent infrastructure for Earth-Mars transfers.
C. Mars Orbiting Station (MOS). A space station in low Mars orbit (LMO) is proposed to allow for staging materials between the surface and LMO as well as providing human missions an extra safe haven in case of catastrophic failures in other systems or destructive weather on the surface. As the ISS provides a life support node in LEO, the MOS provides one in LMO.
D. Change Basic Requirements. If we examine the impact of varying the payload requirements, this can provide a natural way to choose payload amount in a cost-constrained program. The key system parameters for the DRM transfer vehicle were placed in the engineering model described above. By varying the amount of payload, both human and space hardware, the resulting transfer vehicle mass is shown in Table 4. For 1800 kg of space hardware payload removed results in a transfer vehicle mass savings of 4230 kg. For each person removed from the mission results in a mass savings of 17656 kg.

E. Reuse of Surface Habitats. As time goes on, it is evident that the surface habitats on Mars for the DRM grows as more people arrive. The number of inhabitants remains the same. This gives the last crew the most comfortable amount of space as compared to the previous crews. If one were to normalize the mass per person over all missions using the most austere environment of the first mission, then the overall amount of mass needed to be delivered to the surface should decrease substantially.

F. Use A Commercial Approach. The cost of government programs almost always can be done more cheaply through the commercial sector. The last 40 years of human spaceflight endeavors performed by various governments have certainly reduced the risk to the point that companies can start to imagine how human spaceflight could be performed with no, little, or reduced government support.

At first, some of these architecture changes may appear costly, but with more investigation they might actually provide a net cost savings while significantly improving overall system operational redundancy.

Table 4. Modeled Transfer Vehicle Mass by Varying Payload Amount

Number of People	Payload Mass, kg	Transfer Vehicle Mass, kg
6	1800	110166
6	0	105936
4	0	70624
3	0	52928
2	0	35312
1	0	17656

THE IMPACT OF CHANGING PAYLOAD REQUIREMENTS

To see how payload requirements affect the Mars DRM let's first examine what type of payload drives the architecture. For the DRM 14.72 metric tons of science equipment are planned for delivery to the Martian surface. There is about 560 metric tons total planned for delivery to the Mars Surface. So only about 3% of the mass is science payload. Some of the mass delivered would be to supply power and consumables for the science payloads. But it is safe to say that the Mars DRM is driven by the human payload.

The mass savings for the transfer vehicle is realized by reducing the number of people on the mission described above in Table 4. The impact on the entire mission can be approximated by dividing the total DRM system mass (over 12 launches) by the number of people delivered

(18). This number is 3463 metric tons per person. Note that for Apollo, 491 metric tons were needed per person on the 6 landing missions showing seven times the mass is needed to support people on Mars (using the NASA DRM) as opposed to going to the moon for much shorter periods (3463/491 ~ 7).

KEY PARAMETERS FOR SURFACE ARCHITECTURE OVER TIME

By examining the difference between two key model parameters for the Mars DRM over time, we can see that there is a drastic change in the amount of equipment available to the second and third landing groups as compared to the first. Looking at the crew system specific weight, this increases drastically from 11533 kg/person to 48166 kg/person between the 1st and 2nd crews. Similarly the powerless bus to payload ratio decreases from 1.33 to 0.46. These both indicate that a large increase in the amount of equipment available to support the later crews. This may just be the plan for providing redundancy over time and the price paid for expanding the exploration frontier.

Another way to look at this difference in key system parameters would be to realize that overall system mass could be saved by maintaining roughly the same amount of equipment for each crew. Yes, this is an opportunity to save overall system mass. This would certainly not be true if we were planning to keep personnel there for even longer stay periods (i.e. 1200 days) requiring 2 crews to be supported at the same time. Then again, that is not the plan for the DRM.

CONCLUSIONS

This paper has shown the basic governing equations for both an engineering and cost model that can be used for human-tended space facilities and vehicles. Although the cost model used is rough and not absolute, it can show the relative effect of various payloads and facility designs. The power of combining both engineering and cost models can effectively show the impact of payload requirements on cost. Alternatively, this combination can show the amount of payloads you can support given a limit on the initial investment amount. Such trades are critical during the concept design phase for space facilities. For a commercially developed and operated facility such trades are mandatory in providing a business case based upon sound engineering and cost data. To learn more about the modeling techniques involved in this paper, please contact Dr. Charles Reynerson at creyners@alum.mit.edu.

Through the above analysis it was found that the DRM system mass, and therefore cost, could be reduced significantly by changing requirements such as reducing the number of crewmembers and the amount of surface science payloads. The effect of reducing the number of crew members by far outweighs the effect of reducing science payloads. Other potential ways to reduce cost include reuse of transportation systems and surface habitats and taking a commercial approach rather than an inflated government approach. This may be crucial information in selling the price tag to all countries destined to be involved in this endeavor. If the mission is not sellable due to the price tag, this paper has shown that key mission requirements can be traded with desired cost.

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[2] Reynerson, Chapter 4

[3] Reynerson, Chapter 4.

[4] Wertz, SMAD, pp 735, table 20-14.

[5] Space Business News, February 5, 1997, pp 3.

[6] Ezell, pp 62-69.

[7] International Space Station Fact Book.