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Feasibility Analysis of Spacecraft Design for a Manned Mars Free-Return Mission in 2018

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The Inspiration Mars Foundation (IMF) is committed to sending a two-person American crew, a man and a woman, on a journey to fly around Mars and then return to Earth safely. In 2018, Earth and Mars will share a unique orbit, creating an opportunity to travel to Mars and back in only 501 days. In this paper, we conducted the design of a mission and spacecraft for the Inspiration Mars (IM) mission using the design procedures described in the Human Spaceflight textbook and then evaluated them based on the rules of the Mars Society International Student Design Competition. We especially focused on the habitable volume which is an important system driver for designing spacecraft. The Initial Mass in Low Earth Orbit (IMLEO) for this mission was estimated using heuristics and relationships derived from the rocket equation while varying the habitable volume per crewmember from 6 m^3 to 18 m^3 (24 m^3 to 72 m^3 total pressurized volume). As a new spacecraft design, a Crew Reentry Vehicle (CRV) and a Mars Transit Habitat (MTH) with 72 m^3 pressurized volume and hybrid Life Support System (LSS) remained by comparing the Life Cycle Cost (LCC)s and failure rates. The crew and the cargo can be launched by three Falcon Heavy launches.

Nomenclature

4BMS	=	Four Bed Molecular Sieve
AMCM	=	Advanced Mission Cost Model
B	=	Hardware generation (1 for new design, 2 for second generation)
CER	=	Cost Estimating Relationship
CM	=	Crewmember
CRV	=	Crew Reentry Vehicle
CS	=	Crew supply
D	=	Difficulty (0 for average, 2.5 for very difficult, -2.5 for very easy)
DPC	=	Development and Production Cost
DRM	=	Design Reference Mission
ECLSS	=	Environmental Control and Life Support System
ESMD	=	Exploration Systems Mission Directorate
f_i	=	Failure rate of subsystem i
HTV	=	H-II Transfer Vehicle
i	=	Number of subsystem
IM	=	Inspiration Mars
IMF	=	Inspiration Mars Foundation
IMLEO	=	Initial Mass in Low Earth Orbit
IOC	=	Initial Operation Capability
ISS	=	International Space Station
I_{sp}	=	Specific impulse

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LCC	=	Life Cycle Cost
LSS	=	Life Support System
LiOH	=	Lithium hydroxide
M	=	System dry mass
m_{bo}	=	Burn-out mass
MDRS	=	Mars Desert Research Station
MF	=	Multifiltration
MOCM	=	Mission Operations Cost Model
m_{prop}	=	Propellant mass
MTH	=	Mars Transit Habitat
N	=	Maximum number of subsystem
NASA	=	National Aeronautics and Space Administration
NEA	=	Near Earth Asteroid
OGA	=	Oxygen Generation Assembly
Q	=	Total quantity of development and production
R	=	Number of redundancy of subsystem
S	=	Type of mission (2.13 for human habitat, 2.27 for human reentry)
SLS	=	Space Launch System
SPWE	=	Solid Polymer Water Electrolysis
STEM	=	Science, Technology, Engineering and Math
TCCS	=	Trace Contaminant Control Systems
TMI	=	Trans-Mars Injection
VCD	=	Vapor Compression Distillation
Δv	=	Velocity increment

I. Introduction

A private organization called the Inspiration Mars Foundation (IMF) announced on February 27, 2013 that it is planning a manned Mars flyby called the Inspiration Mars (IM) mission[1]. The IMF is committed to sending a two-person American crew, a man and a woman, on a journey to fly around Mars at a distance of 160 km and then safely return to Earth. In 2018, the planets will literally align, offering a unique orbit opportunity to travel to Mars and back in only 501 days. The mission will provide a platform for unprecedented science, engineering, and educational opportunities, using state-of-the-art technologies derived from NASA and the International Space Station (ISS).

A feasibility analysis has been published by the IMF[2, 3]. In addition, the Mars Society is holding an international student design competition for the IM mission[4]. Designs in the competition will be evaluated using a scoring system made up of a 100 points maximum score broken down into 30 points for cost, 30 points for technical quality of design, 20 points for operational simplicity and 20 points for schedule.

As of January 2014, 38 teams representing 56 universities in 15 countries had registered for the student competition[5]. This competition will inspire and motivate youth through the promotion of the science, technology, engineering and math (STEM) educational program. For example, in the Space Habitat Design course taught at the University of Colorado Boulder, in the fall semester of 2013, the contest was used to motivate the students who conducted group work to design new mission concepts and architecture. They used *Human Spaceflight: Mission Analysis and Design*[6] as a reference.

Various studies have been published on manned Mars missions, such as NASA Case Studies, 90-Day Study, Synthesis Group, NASA Mars Design Reference Mission (DRM) version 1.0 - 5.0, Exploration Systems Mission Directorate (ESMD), and Mars Direct[7]. In such studies, the plans included landing and conducting activity on the Martian surface with large-scale exploration systems. Compared to those missions, the unique characteristics of the IM mission are that a crew leaves Earth, flies by Mars, and returns to Earth using a fast free-return trajectory of 501 days without needing any additional maneuvers after the Trans-Mars Injection (TMI). Since the crew will not land on Mars, the system and operations are relatively simple. As the first of its kind, the mission's challenge is to achieve this human deep space flight with current existing technology. However, the crew cannot easily abort the mission after the spacecraft leaves Earth since it is on a free return trajectory. In addition, the crew cannot use resources such as carbon dioxide available on Mars since they will not land. Therefore, the Environmental Control and Life Support System (ECLSS) for this mission must be ultra-reliable, and able to operate reliably for over 500 days.

The whole system needs to be developed and launched by the end of 2017 to complete this mission. Therefore, the limited time between now and 2017 presented a crucial factor to consider in designing the mission. In this paper, we conducted the design of the mission and spacecraft for the IM mission using the design procedures described in the *Human Spaceflight* textbook and then evaluated them based on the rules of the Mars Society International Student Design Competition.

II. Mission goals and requirements

A. Mission statement

As stated in the IM mission statement, this project aims to execute a human flyby mission to Mars in late 2017. The IMF is committed to sending a two-person American crew, a man and a woman, on a journey to fly around Mars at a distance of 160 km and then safely return to Earth[1]. Eight goals and four requirements for the mission are described in the technical summary report published by the IMF[3]. Regarding the mission design specifically, three goals and three requirements are studied here. The three goals are to 1) demonstrate the feasibility of human missions to Mars, 2) address the technical risks involved with human deep space exploration, and 3) conduct research on human physiology during deep space travel. The three requirements are to 1) implement the earliest practical IM fast free-return trajectory, 2) use two crewmembers consisting of one man and one woman, and 3) return the crew safely back to Earth.

B. Mission requirements and constraints

The mission requirements and constraints governing the spacecraft design derived from these three goals and three requirements are shown in Table 1. The functional requirements consist of a two-person crew size made up of a man and a woman, a mission duration of at least 501 days, a destination to reach Mars at a distance of 160 km, the operating environment is at zero-g with deep space extreme temperature and radiation, a trajectory with a fast return trajectory, and vehicle needs for reentry and transfer.

Table 1. Top level mission requirements and constraints

Factors	Mission requirements and constraints
Functional requirements	
Performance :	
Crew size	2-person – a man and a woman
Mission duration	At least 501 days
Destination	Fly around Mars at a distance of 160 km
Operating environment	Zero-g in deep space, extreme deep space temperatures and radiation
Trajectories	Fast free return trajectory
Vehicle	Crew Reentry Vehicle (CRV) and Mars Transit Habitat (MTH)
Operational requirements	
Duration:	
Vehicles	At least 501 days
ECLSS	Storage or recycling
Habitable volume	More than 6 m^3/CM and less than 18 m^3/CM
Logistics:	Limited to current inventory of launch assets
Mission operations:	Minimize complexity and cost
Command, Control, and Communications:	Communications delay between Mars and Earth
Constraints	
Cost	Minimize
Schedule	Must launch at launch window between Dec. 24, 2017 and Jan. 4, 2018
Development	Current existing vehicle, technology, and operation

The operational requirements consist of Crew Reentry Vehicle (CRV) and Mars Transit Habitat (MTH) that are operational for at least 501 days, an ECLSS that includes both storage and recycling options, an optimal habitable volume requirement between 6 m^3 /Crewmember (CM) and 18 m^3 / CM, logistics limited to the current inventory of launch assets, the mission operation needs of minimum complexity and cost, and a communication system with a delay between Earth and Mars. The constraints of cost and the 2017 launch window restrict development to currently existing vehicles, technology, and operations.

C. Top-level mission concept

The IM mission concept has been exhibited by the IMF[1-3]. The sequence includes a two person crew consisting of a man and a woman, vehicles, and a launch system carrying the crew and the logistics for 501 days for flying to Mars by the TMI, doing a flyby around Mars, and returning to Earth using a fast free-return trajectory. This mission architecture consists of crew, trajectory, space vehicle, launch system, and mission operation elements.

D. Trajectory and launch window

This mission utilizes what is known as a “fast free-return trajectory”[2]. This particular trajectory, shown in Fig. 1, occurs only once every 15 years with the next opportunity arising in late 2017. The total travel time is 501 days.

There is a fixed and finite time window in which to execute the TMI burn for the IM mission trajectory. Due to rocket use restrictions for the good of completing the mission, the TMI window opens when the C3 (minimum energy requirement to accomplish the mission for a given payload mass) drops below 40.5 km^2/s^2. Due to the limitations of the current heat shield materials, the window closes when the expected Earth reentry speed at the end of the mission exceeds 14.2 km/s. This results in a TMI window from December 24, 2017 to January 4, 2018[2].

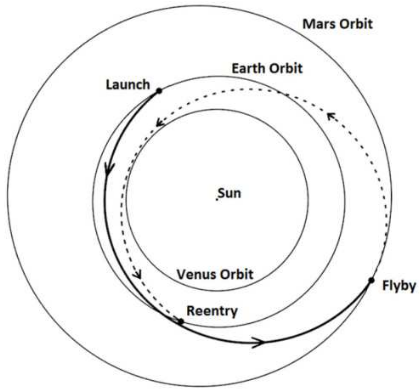


Figure 1. Inspiration Mars Trajectory

III. Mission architecture

A. Mission elements and trade tree

A space mission architecture is the mission concept plus a definition of each of the elements of the mission[8]. Four of the mission elements, consisting of the mission concept, development strategy for CRV, habitable volume, and selection of the ECLSS type, have been chosen from the requirements and constraints in Table 1 as important mission architecture elements for making trade trees to assist with designing this mission architecture. There are two options for the mission concept consisting of multiple and single launch methods, two options for the development strategy for CRV, two options for the MTH habitable volume consisting of 6 m^3/ CM and 18 m^3/CM, and 3 options for selection of the ECLSS type consisting of a storage, recycling, or hybrid (storage and recycling) type.

The trade tree made up by a combination of the four mission elements is shown in Table 2. Table 3 shows possible space launch system candidates for 2017. Trade tree options 7-12 in Table 2, which are a combination of multiple launch and commercial based development, such as Dragon, are studied in this section because there is no existing rocket with an ability to launch a 130-ton payload in a single launch. The SLS Block I rocket development will not be ready in time for the launch in 2017, and CRV based on Orion is too large for 2 crewmembers. The launch costs of Falcon 9 for crew and Falcon Heavy for cargo are the lowest in Table 3.

Table 2. Trade tree for designing mission architecture

Mission concept	Development strategy for CRV	Habitable volume, m^3/CM	ECLSS Type	Launch for crew	Launch for cargo	Option number
Multiple Launch	Orion (Government)	18	Storage	Delta IV Heavy	SLS Block I	1
			Recycling			2
			Hybrid			3
		6	Storage			4
			Recycling			5
			Hybrid			6
	Dragon (Commercial)	18	Storage	Falcon 9 or Falcon Heavy	Falcon Heavy	7
			Recycling			8
			Hybrid			9
		6	Storage			10
			Recycling			11
			Hybrid			12
Single Launch	Orion (Government)	18	Storage	SLS Block II	-	13
			Recycling			14
			Hybrid			15
		6	Storage			16
			Recycling			17
			Hybrid			18
	Dragon (Commercial)	18	Storage			19
			Recycling			20
			Hybrid			21
		6	Storage			22
			Recycling			23
			Hybrid			24

Table 3. Comparison of space launch systems

	Falcon 9 [9]	Falcon Heavy [10]	SLS Block I [11] SLS Block II [11]	Delta IV Heavy [12]
Payload to LEO, kg	13,150	53,000	77,000 143,000	23,000
Payload to TMI, kg	TBD	13,200	20,200 TBD	TBD
Launch Cost, M$	56.5	77.1	500	140
Fairing Diameter, m	5.2	5.2	TBD 8.4	5.0
First operation, year	2010	2014	2017 TBD	2004

B. Habitable volume and spacecraft mass

Vehicle mass

The initial vehicle mass is estimated based on the pressurized volume required by the crew. The pressurized volume per crewmember as a function of the mission duration and performance level required by the crew has been defined[13] and will be discussed in a later section of this paper. The first calculation regarding the spacecraft mass can be estimated using the following equation, which is based on historical data for human spacecraft[14]. This value represents the expected mass at the end of the maneuver, which is called the burn-out mass, or m_{bo}.

$$m_{bo} = 592 \text{ x } (\textit{number of crew} \text{ x } \textit{mission duration} \text{ [day] x } \textit{pressurized volume} \text{ [m}^3\text{]})^{0.346}$$

For example, if the pressurized volume per crewmember is assigned as 36 m^3, the m_{bo} is as follows:

$$m_{bo} = 592 \text{ x}(2 \text{ x } 501 \text{ x } 36)^{0.346} = 22{,}341 \text{ kg}$$

The spacecraft mass is estimated at 18,146 kg by subtracting the reduced mass of 4,195 kg, (19 % of the burn-out mass m_{bo})[15], from 22,341 kg.

Propellant mass

The most massive component of the spacecraft is the propellant, which depends on the vehicle mass, the propulsion system efficiency, and the size of the propulsive maneuvers, represented by a velocity increment. The propellant mass, m_{prop}, is described using the vehicle mass following the maneuver, m_{bo}, the required velocity increment, Δv, and the propulsion system's specific impulse, I_{sp}. Here, the values have been plugged into the form for a rocket equation:

$$m_{prop} = m_{bo}\,(e^{(\Delta v/Isp\cdot g)} - 1)$$

where g is the acceleration due to gravity, 9.8 m/s^2. The Δv and I_{sp} were estimated[2] as the Δv being 4882 m/s, and the I_{sp} being 340 s when the two values are assigned, the m_{pop} is 60,393 kg as follows.

$$m_{prop} = 18{,}146\,(e^{(4{,}882/340\cdot 9.8)} - 1) = 60{,}393\text{kg}$$

Applying the 15 % rule-of-thumb[15] yields a mass of 9,059 kg for the propulsion system. Adding this 9,059 kg, consumable mass of 8,812, and crew of 220 kg to the original m_{bo} estimate of 18,146 kg gives a total of 36,237 kg. When the above equation is repeated, the new total mass is estimated to be 120,603kg.

$$m_{prop} = 36{,}237\,(e^{(4{,}882/340\cdot 9.8)} - 1) = 120{,}603\text{ kg}$$

When the propellant of 120,603 kg and propellant for attitude control of 816 kg is added to the dry mass 36,237 kg, the final Initial Mass in Low Earth Orbit (IMLEO) is 157,656 kg.

C. ECLSS type and failure rate

The size of the mission payload is defined by the crew input and output and crew supply (CS) shown in Tables 4 and 5. Both the input and output of each crewmember is defined at 5.01 kg/CM-day with shower, dishwashing, and hygiene water not included. The logistics for the crew's daily life needs is defined 6.26 kg/CM-day as shown in Table 5, which is estimated using the current operation data of the ISS [16-18]. The final crew supply, 6.89 kg/CM-day, includes a 10 % margin.

Table 4. Crewmember input and output

Input	kg/CM-day	Output	kg/CM-day
Oxygen	0.84	Carbon dioxide	0.84
Drinking and food preparation water	2.38	Respiration and perspiration condensate	2.38
Urine flush water	0.5	Used urine flush water	0.5
Wash water	1.29	Used wash water	1.29
Total supplies	5.01	Total output	5.01
Shower	N/A	Used shower water	N/A
Dish washing	N/A	Used dish washing water	N/A
Hygiene water	N/A	Used hygiene water	N/A

Table 5. Current ISS crew supply

Crew supply	kg/CM-day	m^3/CM-day
Food	2.51	0.00757 [19]
Crew supply	1.19	0.00486 [19]
Maintenance	2.56	TBD
Total	6.26	0.0124
Total x 1.1	6.89	0.0137

The failure rates for the storage LSS and the recycling LSS will be reviewed before the design review of the CRV and the MTH. We use the simplified system failure rate estimation method for system redundancy and spare parts introduced by Jones, 2012[20]. The individual component failure rates in this paper were also determined based on the LSS reliability values and assumptions from Jones, 2012[20]. The probability of failure, Pr(fail), versus the duration for two storage LSSs and five multiple redundant recycling LSSs is shown in Table 6. The Pr(fail) of a series system consisting of N subsystems, such as Four Bed Molecular Sieve (4BMS), Sabatier, Solid Polymer Water Electrolysis (SPWE), Multifiltration (MF), and Vapor Compression Distillation (VCD), are calculated as follows:

$\Pr(\text{fail}) = \Sigma f_i^R$, i = 1,2,3・・・N

i: Number of subsystem i

f_i: Failure rate of subsystem

R: Number of redundancies of subsystem

The failure rate of the recycling LSS with 10% spares is estimated to be one tenth the failure rate in the reference[20]. On the other hand, the failure rate of the storage LSS is estimated by Yakut and Barker, 1968, in the reference[20]. The probability of failure for the storage LSS with 10% spares is calculated by the binomial distribution model. The Pr(fail)s for required, storage, storage and 10% spares, recycling and 10% spares, and multiple redundant recycling and 10% spares are shown in Table 6 and Fig. 2. The storage for LSS and the five-time redundant recycling and 10% spares for the LSS are able to meet the required failure rate, 0.002, for a 500-day mission.

Table 6. Storage and recycling LSS failure rates versus duration

Day	Pr(fail) required for 500-day mission	Pr(fail) storage	Pr(fail) storage and 10% spares	Pr(fail) recycling and 10% spares	Pr(fail) dual redundant recycling and 10% spares	Pr(fail) triple redundant recycling and 10% spares	Pr(fail) four redundant recycling and 10% spares	Pr(fail) five redundant recycling and 10% spares
1	2.E-03	3.72E-05	5.82E-11	1.59E-03	5.39E-07	1.92E-10	7.16E-14	2.77E-17
10	2.E-03	3.72E-04	5.82E-09	1.59E-02	5.39E-05	1.92E-07	7.16E-10	2.77E-12
100	2.E-03	3.72E-03	5.81E-07	1.59E-01	5.39E-03	1.92E-04	7.16E-06	2.77E-07
500	2.E-03	1.86E-02	1.45E-05	7.96E-01	1.35E-01	2.40E-02	4.48E-03	8.64E-04
1000	2.E-03	3.72E-02	5.77E-05	1.59E+00	5.39E-01	1.92E-01	7.16E-02	2.77E-02

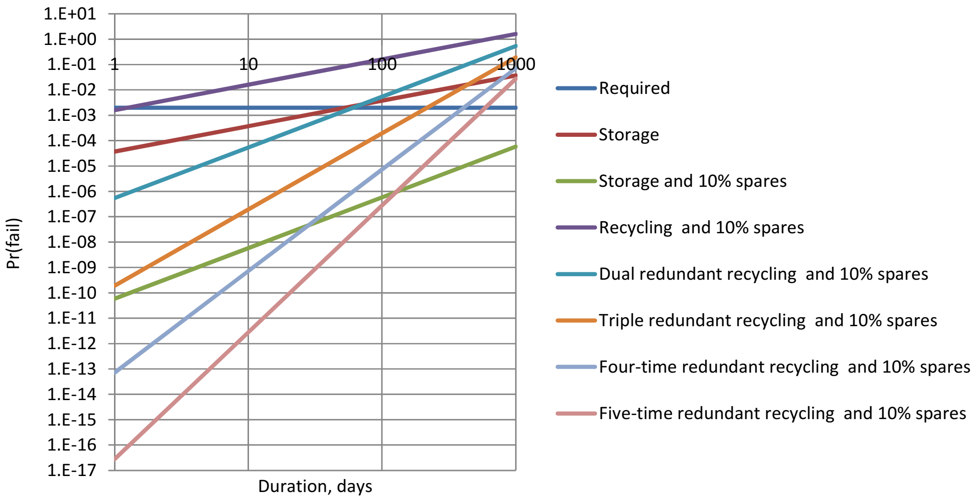


Figure 2. Storage and recycling LSS's Pr(fail)

Fig. 3 shows the recycling LSS's Pr(fail) versus the recycling rate. The recycling rate is the ratio of LSS operation days to 500 days. The Pr(fail) for single, dual, triple, four-time, and five-time redundant recycling and 10% spares LSS are able to meet the required failure rate within 1, 60, 218, 408, and 591 days, respectively. Therefore, we will compare storage, hybrid-3 (triple redundant recycling and 10 % spares with 282-day partial storage), hybrid-4 (four-time redundant recycling and 10 % spares with 92-day partial storage), and recycling (five-time redundant recycling and 10 % spares with partial 50-day storage) LSSs regarding their mass and volume. According to the recycling Pr(fail) versus recycling rate in Fig. 3, the recycling LSS corresponds to the hybrid-5 LSS. Here, we think the hybrid-4 LSS has an advantage over the recycling LSS because of its combination of different technologies in the redundancies, such as the storage LSS and the recycling LSS being used in parallel, which may decrease the failure rate and common cause failure. In addition, partial storage also provides a useful dynamic buffer for down time and repairs[21].

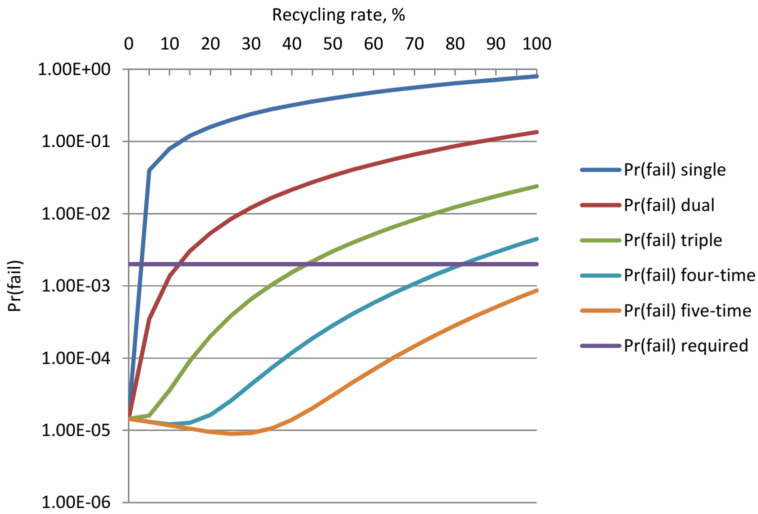


Figure 3. Recycling LSS's Pr(fail) versus recycling rate

The three LSS options for the storage, recycling, and hybrid-4 will be reviewed. The LSS subsystem mass and volume for the recycling and hybrid-4 are estimated in Table 7 based on LSS subsystem specifications[22].

Table 8 shows masses for the storage, recycling, and hybrid-4 LSS operation, which are estimated based on crewmember input and output as shown in Table 4. 501-day, 50-day (500 days x 0.1), and 92-day (500 days - 408 days) masses such as oxygen, water, nitrogen, and LiOH stored for storage, recycling, and hybrid LSSs are shown, respectively. The storage masses include a 10 % margin for safety. The tankages of oxygen, water, and nitrogen are used as coefficients 0.364 kg/kg, 0.2 kg/kg, and 0.556 kg/kg, respectively[23]. The oxygen and nitrogen leak rate are set at 0.0005 %/m^3 [24].

Table 7. LSS subsystem mass

Subsystem	Mass, kg/CM	Volume, m^3/CM	Recycling			Hybrid-4		
			Redundancy	Total mass, kg	Total volume, m^3	Redundancy	Total mass, kg	Total volume, m^3
4BMS	30	0.15	5	300	1.5	4	240	1.2
TCCS	20	0.15	5	200	1.5	4	160	1.2
OGA	35	0.03	5	350	0.3	4	280	0.24
Sabatier	38	0.07	5	380	0.7	4	304	0.56
VCD	25	0.1	5	250	1	4	200	0.8
MF	10	0.04	5	100	0.4	4	80	0.32
Spares	-	-	-	158	0.27	-	126	0.22
Total	-	-	-	1,738	5.7	-	1,390	4.5

Table 8. Mass for LSS operation

Subsystem	Mass	Volume	Storage		Recycling		Hybrid-4	
	kg/CM-day	m^3/CM-day	kg for 501 days	m^3 for 501 days	kg for 50 days	m^3 for 50 days	kg for 92 days	m^3 for 92 days
Oxygen	0.84	0.002772	842	2.7775	84	0.2772	155	0.5100
Oxygen tankage	0.31	0.001009	306	1.0110	31	0.1009	56	0.1857
Water	5.31	0.005310	5,321	5.3206	531	0.5310	977	0.9770
Water tankage	1.06	0.001060	1,062	1.0621	106	0.1060	195	0.1950
LiOH and packaging	1.75	0.005000	1,754	5.0100	175	0.5000	322	0.9200
Oxygen leak	0.0088	0.000029	4.42	0.0146	4.42	0.0146	4.42	0.0146
Oxygen leak tankage	0.0032	0.000011	1.61	0.0053	1.61	0.0053	1.61	0.0053
Nitrogen leak	0.0353	0.000116	17.68	0.0583	17.68	0.0583	17.68	0.0583
Nitrogen leak tankage	0.0196	0.000065	9.83	0.0324	9.83	0.0324	9.83	0.0324
Total	-	-	9,318	15.3	960	1.6	1,738	2.9
Total x 1.1	-	-	10,250	16.8	1,056	1.8	1,912	3.2

D. Evaluation of mission architecture using life cycle cost

Life Cycle Cost (LCC)s for the storage, recycling, and hybrid-4 LSSs are estimated. LCC includes all costs incurred during the development, launch, and operations of the space mission. The development and production cost can be estimated early in the conceptual design using the Advanced Mission Cost Model (AMCM) developed at Johnson Space Center[25,26]. This model consists of a single Cost Estimating Relationship (CER) relating a set of independent variables to the total system cost in FY 1999 dollars. The equation, derived using regression analysis, is

$$\text{System Cost} = 5.65 \times 10^{-4} \times Q^{0.5941} \times (M/0.4536)^{0.6604} \times 80.599^{S} \times (3.8085 \times 10^{-55(1/IOC\text{-}1900)}) \times B^{-0.3553} \times 1.5691^{D}$$

where Q is the total quantity of development and production, M is the system dry mass, S specifies the type of mission (2.13 for human habitat, 2.27 for human reentry), IOC (Initial Operation Capability) is the first year of system operation, B is the hardware generation (1 for new design, 2 for second generation), and D is estimated difficulty (0 for average, 2.5 for very difficult, -2.5 for very easy).

The results of the development and production cost and the failure rate for the storage, recycling, and hybrid-4 LSSs are shown in Table 9. The development and production costs are $125M for storage, $821M for recycling, and $767M for hybrid. The cost of storage LSS is the lowest among three.

Table 9. Development and production cost for storage, recycling, and hybrid-4 LSSs

LSS type	Subsystem	Q	M	S	IOC	B	D	$M
Storage LSS	Oxygen tank	80	4.76	2.13	2017	20	-1	31
	Water tank	80	14.84	2.13	2017	20	-1	66
	LiOH canister	552	0.70	2.13	2017	20	-1	28
	Total	-	-	-	-	-	-	125
Recycling LSS	4BMS	5	60	2.13	2017	4	1	140
	TCCS	5	40	2.13	2017	4	1	107
	OGA	5	70	2.13	2017	4	1	155
	Sabatier	5	76	2.13	2017	2	1	209
	VCD	5	50	2.13	2017	4	1	124
	MF	5	20	2.13	2017	2	1	87
	Oxygen tank	9	4.76	2.13	2017	20	-1	8
	Water tank	9	14.84	2.13	2017	20	-1	18
	LiOH canister	55	0.70	2.13	2017	20	-1	7
	Total	-	-	-	-	-	-	821
Hybrid-4 LSS	4BMS	4	60	2.13	2017	4	1	122
	TCCS	4	40	2.13	2017	4	1	94
	OGA	4	70	2.13	2017	4	1	136
	Sabatier	4	76	2.13	2017	2	1	183
	VCD	4	50	2.13	2017	4	1	109
	MF	4	20	2.13	2017	2	1	76
	Oxygen tank	16	4.76	2.13	2017	20	-1	12
	Water tank	16	14.84	2.13	2017	20	-1	25
	LiOH canister	102	0.70	2.13	2017	20	-1	10
	Total	-	-	-	-	-	-	767

In Table 10, the launch costs for the storage, recycling, and hybrid-4 LSSs are shown together for comparison. The IMLEOs for three LSSs are calculated based on the burn-out mass, 36,237 kg. The launch costs for the three LSSs are calculated by the IMLEO multiplied by launch cost per kg. The launch cost per kg by the Falcon Heavy is set at $1454/kg [10] in the calculations. The launch costs for the storage, recycling, and hybrid-4 LSSs are $65M, $18M, and $21M, respectively, with the storage LSS cost being the largest among the three.

Table 10. Launch cost

		Initial /final mass ratio	Storage LSS	Recycling LSS	Hybrid-4 LSS
Mars Transit Vehicle, kg	36,237	-	10,250	2,794	3,303
IMLEO, kg	157,656	4.35	44,593	12,156	14,369
Launch cost (1454 $/kg x IMLEO), $M	229	-	65	18	21

Table 11 shows the LCC for the storage, recycling, and hybrid-4 LSSs, which is calculated by adding the development and production, launch, and operation costs. The Johnson Space Center Mission Operations Cost Model (MOCM) estimates the operation cost as a percentage of the total development and production cost of spacecraft. For manned spacecraft, the operation cost per year is 10.9% of the total development and production cost. In this model, the operation cost is 10.9 % of the development and production cost [26]. The LCCs are $209M, $962M, and $903M, respectively.

Table 11. Life Cycle Cost (LCC)

	Storage LSS, $M	Percentages	Recycling LSS, $M	Percentage	Hybrid-4 LSS, $M	Percentages
Development and production cost (DPC)	125	60%	821	85%	767	85%
Launch cost (1454 $/kg x IMLEO)	65	31%	18	2%	21	2%
Operation cost (DPC x 0.109 x 501/365)	19	9%	123	13%	115	13%
Total (LCC)	209	100%	962	100%	903	100%

Finally, the development and production costs for the CRV and the MTH are reviewed based on current existing space systems shown in Table 12. Table 13 shows the development and production costs based on the specifications in Table 12. The cost for the CRV is estimated based on the Orion and Dragon vehicles. The cost for the MTH is estimated based on the Node 3, Cygnus, and HTV. The Table 13 shows the estimated cost of a new spacecraft derived from the existing one for the IM mission. The coefficients, 0.5 and 2.8, coordinate the required pressurized volume for this mission. The system generation (*B*) is 1, 2, 3, 2, and 2, for Orion, Dragon, Node 3, Cygnus, and HTV, respectively. The development difficulty (*D*) is 2, 2, 0, 1, and 0, respectively. Node 3 requires the smallest modification compared to the two cargo vehicles. The cost of the CRV is smaller when based on the small capsule, Dragon, as compared with one based on the large capsule, Orion. As the cost of the MTH is almost the same among the three habitat vehicles, another criterion is required for selecting original spacecraft that is best for reducing cost and time of development and production.

Table 12. Specification of crew vehicle and habitat vehicle

Specification	Orion [27]	Dragon Crew [28]	Node 3 [29] (ISS)	Cygnus [30] (Cargo)	HTV [31] (Cargo)
Number of crew	2 - 4	Max 7	-	-	-
Dry Mass, kg	9,684	4,200	12,471	1,500	10,500
Payload, kg		6,000	-	2,000	6,000
Pressurized Volume, m^3	19.5 (Hab8.9)	10	70	18.9	14
Unpressurized Volume, m^3	-	-	-	-	16
Diameter, m	5.0	3.7	4.4	3.07	4.4
Length, m	3.3	2.9	7.2	3.66	9.8
Operation Term	21 -210 days	Up to 2 years	-	30 days	45 days
Launch System	Delta Heavy (2014) SLS (2017-)	Falcon 9	Space Shuttle	Antares	HIIB
First operation	2019	2010	2010	2013	2010

Table 13. Development and production cost for CRV and MTH

Spacecraft	System	Q	M	S	IOC	B	D	$M
CRV	Orion x 0.5	1	4,842	2.27	2017	1	2	4,634
	Dragon modified	1	2,681	2.27	2017	2	2	1,562
MTH	Node 3 modified	1	12,471	2.13	2017	3	0	1,287
	Cygnus x 2.8	1	5,040	2.13	2017	2	1	1,282
	HTV modified	1	10,500	2.13	2017	2	0	1,327

E. System driver and critical requirement

One of the authors of this paper, Miyajima, re-acknowledged the significance of a habitable volume when designing the habitat facility for a long-term stay during the habitation experiment at the Mars Desert Research Station (MDRS) in 2013. In this mission design, the habitable volume is the system driver when designing the mission architecture and spacecraft. The reference[13] defined the required habitable volume per crewmember as a function of mission duration and performance level: Tolerable 5.10 m^3/CM, Performance 9.91 m^3/CM, and Optimal 18.41 m^3/CM. Design studies, such as the lunar surface system for the lunar surface mission[32] and deep space habitat system for the Near Earth Asteroid (NEA) mission[19] focus on habitable volume.

The IMLEOs for this design, shown in Fig. 4, are calculated using the rocket equations described in the above section III B, habitable volume and spacecraft mass, while varying the habitable volume per crewmember from 6 m^3

to 18 m^3 (total pressurized volume is 24 m^3 to 72 m^3, which is double habitable volume). When the crew and the cargo is launched by three Falcon Heavy launches, the possible IMLEO is 159,000 kg, which corresponds to a volume less than 18 m^3 habitable volume per crewmember (72 m^3 total pressurized volume).

Table 14 shows the details of trade tree options 7-12 previously shown in Table 4, which are a combination of the multiple launch mission concept, commercial based CRV development, habitable volume per crewmember of 18 m^3 or 6 m^3, and an ECLSS type of storage, recycling, or hybrid-4. We assume that the volume for water and oxygen, and nitrogen is not included in the pressurized volume. Comparing the total pressurized volume of 24 m^3 to the volume for the consumable crew supply plus volume for the storage, recycling, and hybrid-4 LSSs of 19.2 m^3, 19.9 m^3, and 19.2 m^3, respectively, we identify they all occupy more than 80 %, 83 %, and 80 % of the total pressurized volume respectively, Meaning there is not enough volume capacity for a crew to live. This indicates that such a spacecraft is not possible.

Although there are no problems with the volume when the habitable volume per crewmember is 18 m^3, the IMLEO at 184,656 kg for the storage LSS exceeds the possible launch payload of 159,000 kg. Therefore, the recycling LSS and the hybrid-4 LSS remain the only options when the habitable volume per crewmember is 18 m^3. We thought the hybrid-4 LSS had an advantage over the recycling LSS previously shown in Section III C because a combination of different technologies in the redundancies, such as the storage LSS and the recycling LSS used in parallel, is able to decrease the failure rate and common cause failure.

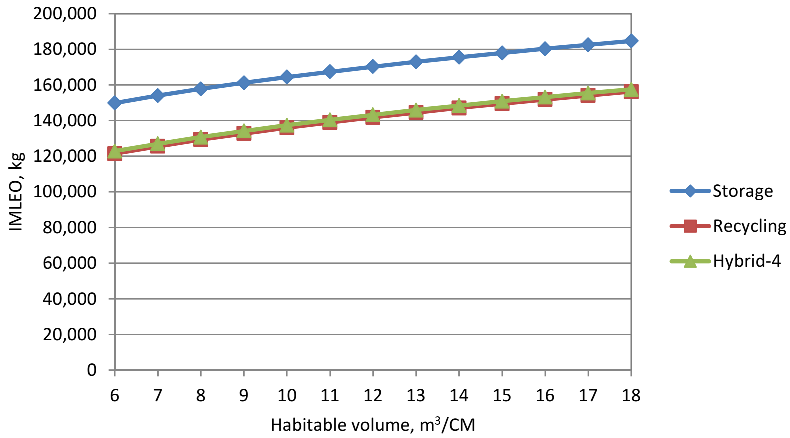


Figure 4. Habitable volume per crewmember vs. IMLEO

Table 14. Comparison of habitable volume, CS volume, and LSS volume

Option	Habitable volume, m^3/CM	ECLSS type	Launch System	Number of launches	Payload to LEO, kg	Required IMLEO, kg	Total pressurized volume, m^3	CS volume*1, m^3	LSS volume*2, m^3	LSS volume*3, m^3	CS and LSS volume*4, m^3	Evaluation
7	18	Storage	Falcon Heavy	3	159,000	184,657	72	13.7	16.8	5.5	19.2	
8		Recycling		3	159,000	156,222	72	13.7	7.5	6.2	19.9	x
9		Hybrid-4		3	159,000	157,656	72	13.7	7.7	5.5	19.2	x
10	6	Storage		3	159,000	149,931	24	13.7	16.8	5.5	19.2	
11		Recycling		3	159,000	121,496	24	13.7	7.5	6.2	19.9	
12		Hybrid-4		3	159,000	122,930	24	13.7	7.7	5.5	19.2	

*1 0.0137 x 2 x 501 (ref. Table 5)
*2 LSS volume for LSS subsystem, LiOH, water, oxygen, and nitrogen (ref. Table 8)
*3 LSS volume for LSS subsystem and LiOH (ref. Table 8)
*4 CS volume + LSS volume for LSS subsystem and LiOH (ref. Table 8)

IV. Baseline concept and architecture

The reference mission and concept of operation for Option 9 is shown in Fig. 5. The layout of the designed spacecraft consisting of a CRV based on the Dragon and an MTH based on Node 3 is shown in Fig. 6. The mass budget and the volume distribution are shown in Table 15 and 16.

In this mission concept, first one crew launch and two cargo launches are carried out. Then three space elements such as the CRV, MTH, and TMI booster rendezvous and dock with each other at LEO. This is followed by the docked spacecraft reaching Mars after the TMI burn stage, a flyby on Mars, a return to Earth using the free return trajectory, and a reentry into Earth's atmosphere. The assembly of the space elements can be done using current existing technology and the ISS operation resource, such as the ground system and human resources. Therefore, new investment and training for the mission operation will be at a minimum. Therefore, the total cost, $3,504 M, can be estimated as shown in Table 17 based on the LCC in Table 13.

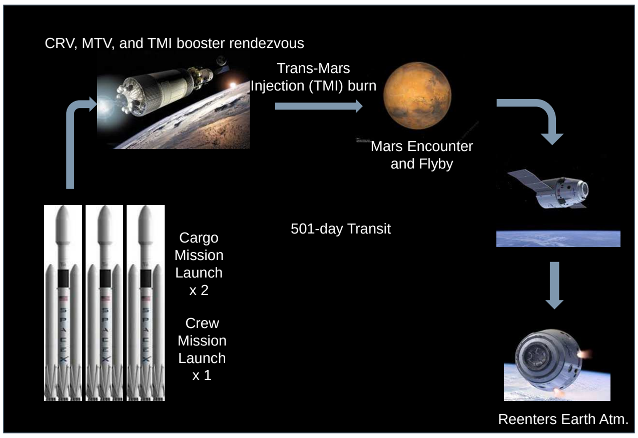
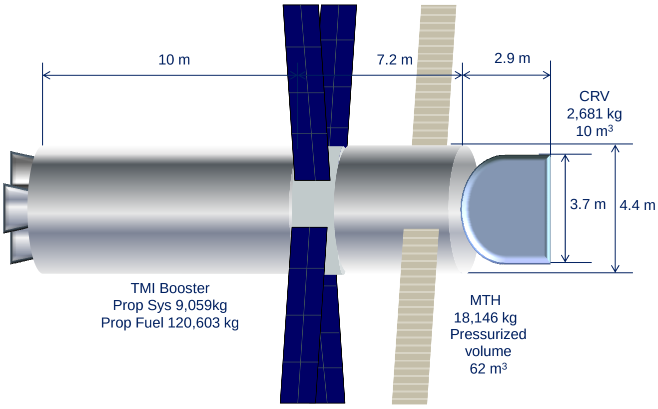


Figure 5. IM Reference Mission/Concept of operation



Crew supply 6,900 kg, 13.7 m^3; Hybrid-4 LSS Logistics 3,284 kg, 7.7m^3

Figure 6. Mass and volume budget of CRV and MTH

Table 15. Mass budget for MTH and CRV

Mass budget	Initial mass, kg
Structure	4,022
Mechanisms	1,564
Thermal protection	1,787
Attitude control	447
Power	2,681
Avionics and control	1,787
ECLSS (Hybrid-4 LSS)	1,390
Crew accommodations	1,787
CRV	2,681
Dry mass without propulsion system	18,146
Propulsion system	9,059
Dry mass	27,205
Crew	220
Crew supply and LSS consumables	8,812
Dry mass + Crew + consumables	36,237
Main propellant	120,603
Propellant for attitude control	816
Gross mass estimate	157,656

Table 16. Volume distribution for habitation in MTH and CRV

Vehicle	Volume	Item	Volume, m^3
MTH 64 m^3	Unpressurized 2 m^3	Water	1.2
		Oxygen	0.7
		Nitrogen	0.1
	Pressurized 72 m^3	Crew Supply	13.7
		LiOH	1
		LSS subsystem (Hybrid-4 LSS)	4.5
		Equipment	11.8
		Habitable	31
CRV 10 m^3		Equipment	5
		Habitable	5

Table 17. Total cost for IM mission

Item	$M
Development and production cost (Dragon modified + Node 3 modified)	2,849
Launch cost, 1454 $/kg x IMLEO	229
Operation cost, DPC x 0.109 x 501days/365says	426
Total	3,504

V. Conclusion

In this paper, we conducted the design of a mission and conceptual spacecraft for the IM mission using the design procedures described in the *Human Spaceflight* textbook and then evaluated them based on the rules of the Mars Society International Student Design Competition. The limited time between now and 2017 is a crucial factor in designing the mission for the IM. Therefore, a feasibility study was conducted on the mission architecture and spacecraft design using current existing technology derived from the ISS.

In this mission concept, the crew and cargo can be launched by three Falcon Heavy launches, the three space elements will rendezvous and dock at LEO, travel to Mars, complete a flyby and return to Earth using the free return trajectory. The assembly of the space elements can be done using current existing technology and the ISS operational concept, thereby keeping new investment and training for the mission operation to a minimum.

We especially focused on habitable volume that is an important system driver for designing spacecraft. The IMLEO for this mission was estimated using relationships based on heuristics and the rocket equations while varying the habitable volume per crewmember from 6 m^3 to 18 m^3 (24 m^3 to 72 m^3 total pressurized volume). As a new spacecraft design, a CRV with 10 m^3 and an MTH with 62 m^3 pressurized volume and hybrid LSS remained by comparing the LCCs and failure rates. The LCCs and failure rates for storage, recycling, and hybrid LSSs were estimated with results indicating that the hybrid LSS is the best ECLSS type for this mission. Therefore, the LSS design will provide ultra-reliable life support that is able to maintain stable operation for more than 500 days.

This mission design also meets the evaluation requirements of the Mars Society International Student Design Competition. The cost is minimized, the technical quality of design is almost ready except for the heat shielding for the CRV and the radiation shielding for the MTH, the operation is relatively simple, and the minimum new development and training required will make the schedule achievable.

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