International Student Design Competition for Inspiration Mars
Mission Report Summary (Team Kanau)

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Abstract

There is strong potential for a near-future, human mission to Mars, with Inspiration Mars and the Mars Society in particular advocating for a two-person, manned flyby mission within the coming decade. This document presents team Kanau's (Kanau, the Japanese word for "collaboration" and "synergism", was chosen to symbolize our team members hailing from different universities of US and Japan) response to the challenge posed by the Mars Society for student groups to plan a flyby of Mars in the year 2018 by a male and female astronaut pair. Building upon analyses published by Inspiration Mars, team Kanau investigates in more detail and presents novel solutions for various aspects of the mission architecture, including: spacecraft design, crew life support, launch vehicle selection, the flyby trajectory, Earth capture and re-entry. While we address many key technical aspects, we give particular attention to ensuring the physical, mental, and psychological health of the two astronauts for the duration of the journey. Specifically, regenerative air scrubbers, the combination of pre-packaged and grown food, as well as 3-D printing technology support the human crew during the 501-day flight. A novel combination of a Falcon Heavy launch to LEO and a ULA ACES-enabled Trans-Mars Injection places the crewed spacecraft on a free-return trajectory, while aerocapture upon Earth return enables reduction in the required size of the re-entry heat shield. The technologies and design concepts that we propose, while requiring further development prior to the stated 2018 mission opportunity, are powerful enabling factors for the crewed Mars flyby as well as other manned deep space missions.

1 Mission Objectives

Our mission objective is to design a Mars flyby mission for two crewmembers in 2018 that is both safe and cost efficient. Any Mars-bound manned spacecraft will be too large to launch in one attempt, so in order to meet these objectives we desire a small number of assemblages in outer space.

2 Mission Design

2.1 Mission Design Methods

We carried out the mission design using the process shown in Fig. 1. As a first step, the requirements generation and analysis was performed to find out critical goals and restrictions to be met through the design process. Design options are then generated to satisfy required subsystem functionalities and filtered to arrive at feasible concepts. The subsystem drivers are then used to perform trade studies and select the best design concept as the baseline concept for that subsystem. The baseline architecture is constructed based on these reference concepts.

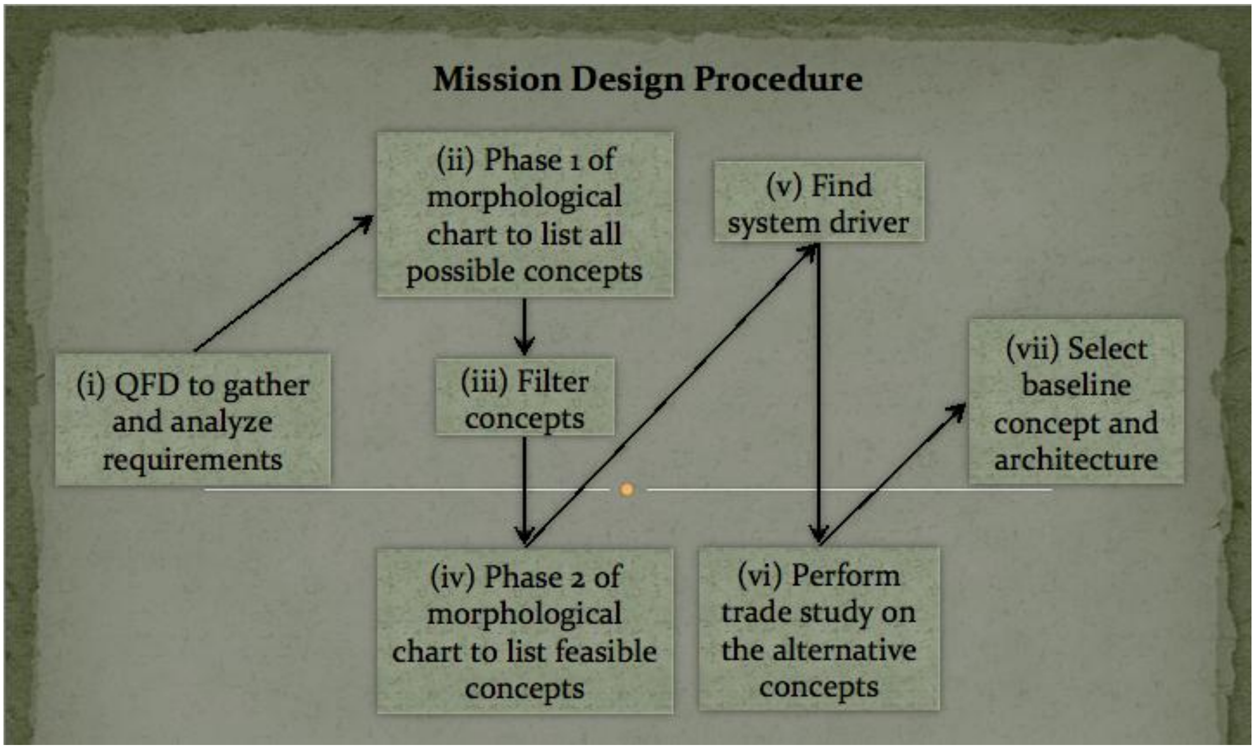

Figure 1 Process for mission design.

2.2 Requirements Generation (The House of Quality)

To gather requirements, we performed quality functional deployment (QFD). In order to translate the needs of the mission into technical characteristics and specifications a "House of Quality"[1] (HoQ) was constructed using the following steps. Portions of the House of Quality appear as figures in the following sections. Please refer to Appendix B for the full House of Quality.

Step 1: Mission Requirements

The team gathered the mission requirements, as shown in Fig. 2, from the design competition website as well as using an affinity diagram.

WHAT	Mission Segments	Affordable
		Timely Launches
	Performance	Adequate Life Support Elements
		Radiation Minimizing Elements
		Required Thrust
		Low Re-entry Velocity
		Easy to Assemble Vehicle Stack in Space
	Maintenence	Repairable
		Reliable
	Communication	Re-establish Link
		Maintain Link
	Additional Constraints	Political Support
		Re-entry Heat Rate Limit
		Vehicle Mass Limit
		Stress-Free Environment

Figure 2 Whats of House of Quality.

Step 2: Regulatory Requirements

The team documented implicit requirements that are dictated by regulatory standards that the mission must adhere to.

Step 3: Requirements Importance Ratings

On a scale from 1 - 10, the importance of each requirement is rated as shown in Fig. 3. The team balanced the stated goals of the competition with our engineering judgment and experience to arrive at these relative rankings. These numbers will be used later in the relationship matrix.

Mission Segments	Affordable	10
	Timely Launches	10
Performance	Adequate Life Support Elements	10
	Radiation Minimizing Elements	8
	Required Thrust	10
	Low Re-entry Velocity	9
	Easy to Assemble Vehicle Stack in Space	7
Maintenence	Repairable	9
	Reliable	7
Communication	Re-establish Link	9
	Maintain Link	9
Additional Constraints	Political Support	10
	Re-entry Heat Rate Limit	9
	Vehicle Mass Limit	8
	Stress-Free Environment	7

Figure 3 Importance Ratings of Requirements.

Step 4: Comparison with the benchmark
The requirements were compared to other mission architectures in order to assess the validity of our design concepts. Since the Inspiration Mars mission design concept is only proposed mission comparable to the competition objectives, team Kanau adapts their standards as a preliminary benchmark for design decisions.

Step 5: Technical Descriptors
The technical descriptors of the mission that can be measured and benchmarked against the competitors are shown in Fig. 4.

HOW
Mission Cost
Mission Safety
Mission Complexity
Technology Readiness Level
Mission Duration
Number of Launches
Re-entry Velocity
Radiation Exposure
Throw Mass Capacity
Psychological Stress

Figure 4 Hows of House of Quality.

Step 6: Direction of Improvement

The desired direction of movement for our design relative to the baseline Inspiration Mars architecture for each descriptor is shown in Fig. 5.

HOW									
Mission Cost	Mission Safety	Mission Complexity	Technology Readiness Level	Mission Duration	Number of Launches	Re-entry Velocity	Radiation Exposure	Throw Mass to Mars	Psychological Stress
billion $	*probability*	#	*year*	*days*	#	*m/s*	*sieverts*	*mT*	*suds*
↓	↑	↓	↑	↓	↓	↓	↓	↑	↓

Figure 5 Direction of improvement over Inspiration Mars baseline.

Step 7: Relationship Matrix

The connection between needs for the mission and the ability to meet those needs was determined using a relationship matrix. Relationships were kept numeric and assigned values equal to 3, 6 or 9 as shown in Fig. 6.

Kanau Inspiration Mars Mission Analysis

Relationship Attributes	
3	Weak
6	Moderate
9	Strong

					HOW										BENCHMARKS
					Mission Cost	Mission Safety	Mission Complexity	Technology Readiness Level	Mission Duration	Number of Launches	Re-entry Velocity	Radiation Exposure	Throw Mass Capacity	Psychological Stress	Inspiration Mars
Units					*billion $*	*probability*	*#*	*year*	*days*	*#*	*m/s*	*sieverts*	*mT*	*suds*	
Relative Importance					↓	↑	↓	↑	↓	↓	↓	↓	↑	↓	1
WHAT	Mission Segments	Affordable	10	7.58	9			6	3	6			6		10
		Timely Launches	10	7.58			9	9		9			6		10
	Performance	Adequate Life Support Elements	10	7.58	9	9			6			9			9
		Radiation Minimizing Elements	8	6.06	9	9		6							7
		Required Thrust	10	7.58	9			6		9			9		10
		Low Re-entry Velocity	9	6.82		6		6			9				9
		Easy to Assemble Vehicle Stack in Space	7	5.30			9			9					7
	Maintenence	Repairable	9	6.82	3	6	3								7
		Reliable	7	5.30	9	9	6			3	3	3			6
	Communication	Re-establish Link	9	6.82	3	6									6
		Maintain Link	9	6.82	3	6									7
	Additional Constraints	Political Support	10	7.58	9	6	9	9	3						10
		Re-entry Heat Rate Limit	9	6.82		9		6			9				9
		Vehicle Mass Limit	8	6.06	3				3	6			9		8
		Stress-Free Environment	7	5.30				6	9					9	6

Figure 6 Relationship matrix.

Step 8: Target and Threshold Values for Technical Descriptors

Target values for the technical descriptors, acting as a baseline to compare against, are tabulated in Fig. 7, along with the limits for each technical descriptor.

HOWS-	Mission Cost	Mission Safety	Mission Complexity	Technology Readiness Level	Mission Duration	Number of Launches	Re-entry Velocity	Radiation Exposure	Throw Mass Capacity	Psychological Stress
Targets	1.5	0.99	3	2015	501	3	12.8	0.65	15	3
Threshold	2	0.98	4	2017	600	4	14.18	1	10	7

Figure 7 Target and threshold values for hows of the House of Quality.

Step 9: Correlation Matrix

Team members examined how the technical descriptors impact each other and documented strong relationships between them as shown in Fig. 8.

HOW	Mission Cost	Mission Safety	Mission Complexity	Technology Readiness Level	Mission Duration	Number of Launches	Re-entry Velocity	Radiation Exposure	Throw Mass to Mars	Psychological Stress	BENCHMARKS: Inspiration Mars	
Mission Cost												
Mission Safety												
Mission Complexity	+	+									++	Strong Positive Relationship
Technology Readiness Level	+	+									+	Positive Relationship
Mission Duration	-	+		-							--	Strong Negative Relationship
Number of Launches	++		+	-							-	Negative Relationship
Re-entry Velocity		+	-	-								
Radiation Exposure		++		-	++							
Throw Mass Capacity			-	++		--						
Psychological Stress				-	++							

Kanau Inspiration Mars Mission Analysis

Figure 8 Correlation matrix.

Step 10: Absolute and Relative Importance

The absolute importance for each technical descriptor is the product of the cell value and the importance rating. Numbers were then added up in their respective columns to determine the importance for each technical descriptor. The relative importance of each technical descriptor was then calculated, with the results shown in Fig. 9. From these absolute importance values, the most crucial technical aspects that drive the mission architecture are then ranked as shown in Fig. 10.

HOWS-	Mission Cost	Mission Safety	Mission Complexity	Technology Readiness Level	Mission Duration	Number of Launches	Re-entry Velocity	Radiation Exposure	Throw Mass Capacity	Psychological Stress
Absolute	600	582	312	498	207	372	183	111	282	63
Relative	0.1869	0.1813	0.0972	0.1551	0.0645	0.1159	0.0570	0.0346	0.0879	0.0196
Targets	1.5	0.99	3	2015	501	3	12.8	0.65	15	3
Threshold	2	0.98	4	2017	600	4	14.18	1	10	7
Rank	1	2	5	3	7	4	8	9	6	10

Figure 9 Absolute and relative importance of hows of the House of Quality.

Rank	Hows
1	Mission Cost
2	Mission Safety
3	Technology Readiness Level
4	Number of Launches
5	Mission Complexity
6	Throw Mass Capacity
7	Mission Duration
8	Re-entry Velocity
9	Radiation Exposure
10	Psychological Stress

Figure 10 Ranking of the hows of House of Quality.

2.3 Concept Generation

2.3.1 Morphological Chart

The subsystems, with their respective required functionalities, needed for the mission are found based upon the HoQ. The options to achieve these functionalities were then found from literature surveys and reviews of previous missions, and the best options were chosen using a morphological chart, i.e., a design matrix, as shown in Fig. 11. We have chosen to exclude infeasible and complex design options, thus reducing the number of design alternatives displayed.

MORPHOLOGICAL CHART

Subsystem	Function	Option 1	Option 2	Option 3	Option 4
Propulsion	Produce required thrust	Chemical propulsion	Nuclear propulsion	Electric propulsion	
	Complete flyby mission	High thrust full propulsion	Free return trajectory	Low thrust full propulsion	Hybrid chemical / electric propulsion
Launch	Trans-Mars Injection	Space launch system	Falcon Heavy	Ariane 5 ME	Delta IV-H / ACES
	Launch to LEO	Space launch system	Falcon Heavy	Ariane 5 ME	Delta IV-H / ACES
	Carry sufficient cargo	2 launches	3 launches		
	Carry crew	1 launch			
Environment control and life support system	Provide habitable space per crew member	5.10 cubic meters	9.91 cubic meters	18.41 cubic meters	
	Oxygen generation	Electrolysis	Storage	Bioregenrative	Hybrid
	Carbon dioxide removal	Sabatier	LiOH canister	Bioregenrative	Hybrid
	Water management	Recyclable	Storage and dump		
	Food	Storage	Grown		
Spacecraft interior design	Simulate comfortable, natural environment	Audio and video devices (multi-channel speakers, video projector and screen)	Audio and video devices (speakers, PC and a monitor)		
Medical healthcare facilities	Reduce exposure to radiation	Shielding by water	Sleeping bag using nano form	Nano form materials	Use consumables as shielding
Electrical power system	Source	Fixed Photovoltaic	Gimbaled Photovoltaic	RTG's	
	Conversion	Direct DC-DC	Thermoelectric		
	Storage	Regenerative Fuel Cell	Si-Zn Battery	Ni-Hyd Battery	Li-Ion Battery
	Distribution	Regulated DC	Unregulated DC	DC-AC	
	Backup	Fuel Cell	Si-Zn Battery	Ni-Hyd Battery	Li-Ion Battery
Thermal control system	Maintain cabin temperature	Active	Passive	Active, Passive backup	
	Maintain exterior temperature	PTC	Active (Radiator)		
	Maintain Photovoltaic temperature (if PV present)	Local	Distributed		
Command and data handling	Determine spacecraft position (contigency navigation)	Star tracker	Inertial measurement unit	Deep Space Atomic Clock	
	Maintain link	Directed radio	Laser	Omnidirectional radio	
Attitude dynamics and control system	Maintain and change spacecraft orientation	Monopropellant	Reaction wheels	Control moment gyro	Spin stabilization
Entry, descent and landing system	Enter Earth's atmosphere	Direct entry	Aerobraking	Aerocapture	
	Use vehicle's aerodynamic properties to decelerate	Ballistic entry	Lifting entry		
	Able to withstand 2000 W/cm^2 of heat rate	PICA-X	SLA	AVCOAT	SIRCA
	Return crew	Dragon	Orion		

Figure 11 Morphological chart.

2.4 Concept Selection

We evaluated each of the design alternatives using QFD to help in the feasibility analysis. For each subsystem, the driver and baseline design were identified based on trade studies. The baseline architecture for the entire mission was then constructed using the best concept (indicated by green text and red cell borders in Fig. 11) from each subsystem.

3 Concept of Operation

The entire mission was divided into nine phases as shown in Fig. 12 and given below:

I. LOX propellant tank launch
II. LOX/LH2 propellant tank launch
III. Crew vehicle launch

IV. Crew rendezvous and dock
V. Earth departure and Trans-Mars Injection (TMI)
VI. Mars flyby encounter
VII. Earth return cruise and approach
VIII. Jettison habitat and service module
IX. Aerocapature, entry, descent and landing, and crew recovery

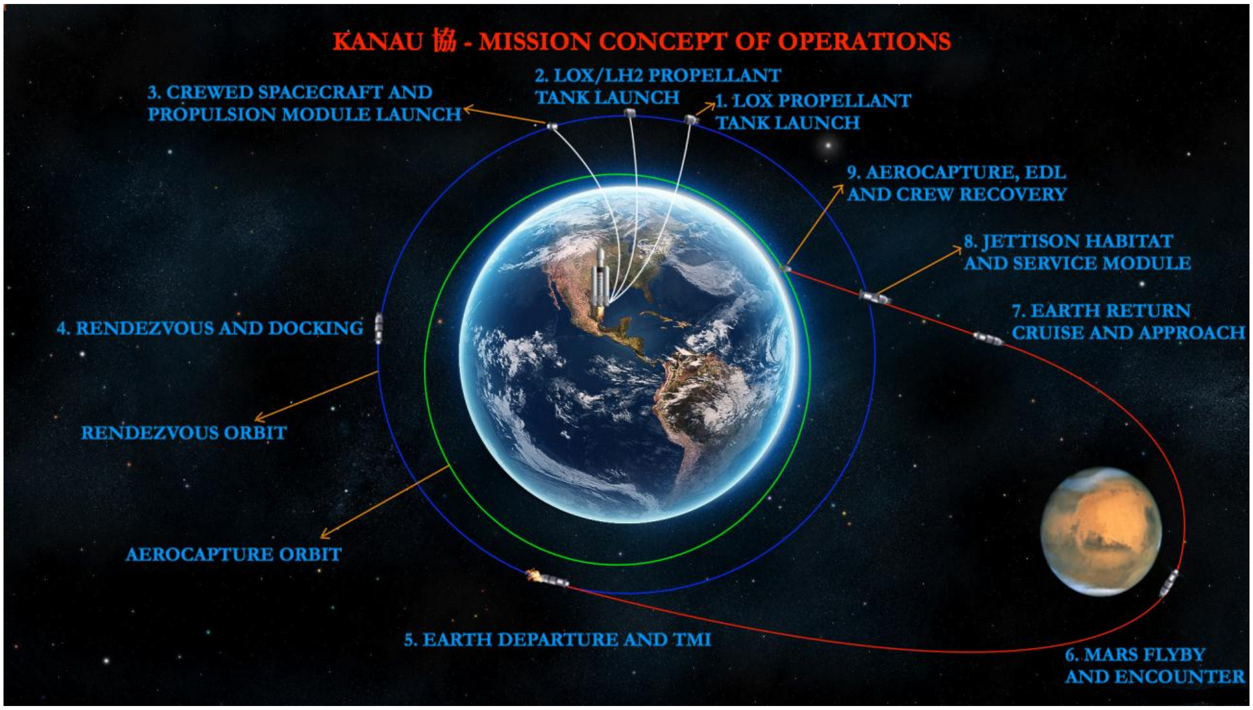

Figure 12 Concept of operations.

An artist's rendition of the vehicle stack adopted by team Kanau is shown in Fig. 13. Note that the illustrated vehicle stack includes the TMI booster stage, the solar panels for electrical power generation, the pressurized crew compartment as well as the Earth re-entry capsule.

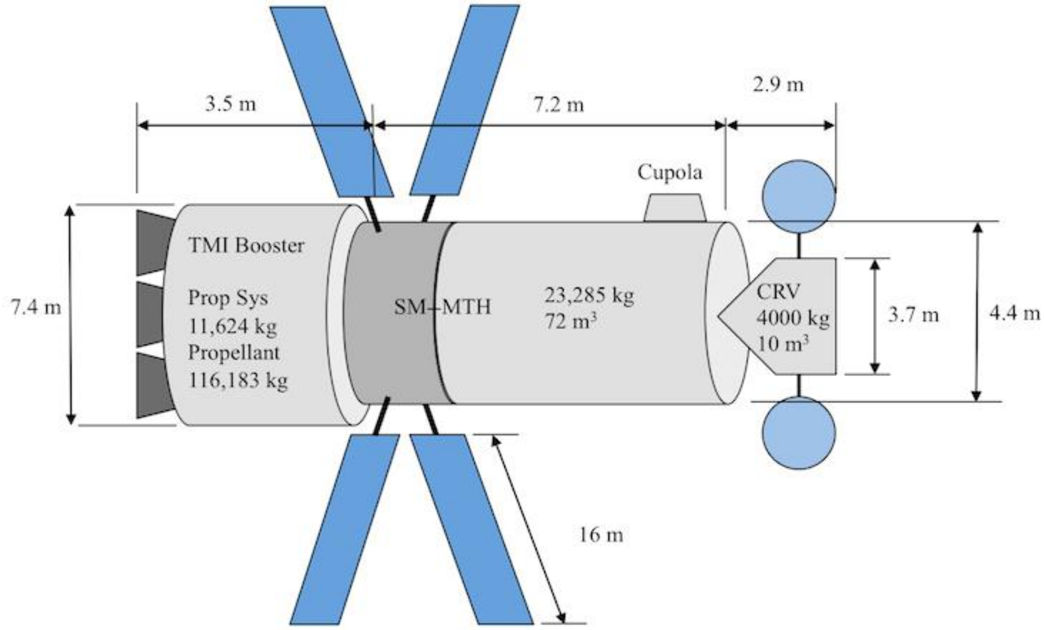

Figure 13 Vehicle stack before TMI.

4 Mission Design and Launch Vehicle Selection

4.1 Overview

As a baseline mission design, we recreate the free-return trajectories identified by Inspiration Mars (IM) in their mission concept documents. We also seek to explore novel mission architectures that could potentially reduce the needed number of launches, increase operational safety margins, and further enhance the baseline scenario. The following sections contain our recreation of the IM baseline as well as an assessment of launch requirements to satisfy the mass/budget specifications of team Kanau.

4.2 Interplanetary Ballistic Free-Return Trajectory

The potential for a fast free-return flyby of Mars by a spacecraft was first identified by Patel[2] in 1998; the original Inspiration Mars concept uses these free-returns to enable what could be the first manned mission to visit Mars.[3,4] For our design concept, we select as our baseline trajectory a January 2018 departure free-return with the flyby occurring in August 2018. After a chemical boost stage from LEO to escape the vicinity of Earth, the manned spacecraft coasts to the Mars close approach, uses Mars for a gravity assist maneuver, and returns to Earth in May 2019 to complete a 501-day journey. The magnitude of the Trans-Mars Injection (TMI) maneuver is 4.86 km/s to depart from the staging LEO altitude of 200 km, where the propulsion system and propellant required for implementing this maneuver must be delivered to LEO in addition to the crewed spacecraft. The critical epochs of the free-return trajectory, constructed using Lambert arcs with the Sun as a point mass and the positions and velocities of Earth and Mars from the Jet Propulsion Laboratory's (JPL's) HORIZONS system, are contained in Table 1 while the calculated hyperbolic velocities are reflected in Table 2.

Table 1 Critical epochs of the Earth-Mars-Earth free-return trajectory.

Leg	DEPART		ARRIVE		Flight Time (Days)
1	Earth	2018 January 04 07:10:33.6 UT	Mars	2018 August 20 07:49:43.7 UT	228.0271999998950
2	Mars	2018 August 20 07:49:43.7 UT	Earth	2019 May 20 20:57:41.8 UT	273.5471999999136
				Total Duration	501.5743999998086

Table 2 Velocities and periapse altitudes of the departure, fly-by, and return hyperbolic trajectories.

Leg	DEPART				ARRIVE			
	Vinf (km/s)	Alt Peri (km)	V Peri (km/s)	C3 (km^2/s^2)	Vinf (km/s)	Alt Peri (km)	V Peri (km/s)	C3 (km^2/s^2)
1	6.230	200 km	12.649	38.811	5.389	239.2	6.501	29.038
2	5.389	239.2	6.501	29.038	8.865	--	14.201	78.596

The interplanetary cruise, illustrated in Fig. 14, is a relatively well-known design that requires only minimal course corrections to address statistical maneuver and navigation errors. The outbound arc to Mars is shown in black while the return leg is dashed; the paths of Earth and Mars are shown in blue and red, respectively. The Earth departure and staging orbit, the close approach to Mars, as well as the Earth return and re-entry are illustrated in Figures 15-17, respectively, where the point of closest approach at Mars is 239.2 km above its surface. These trajectory arcs are computed using a patched conic model; however the Inspiration Mars reports detail similar results generated in a full ephemeris force model.

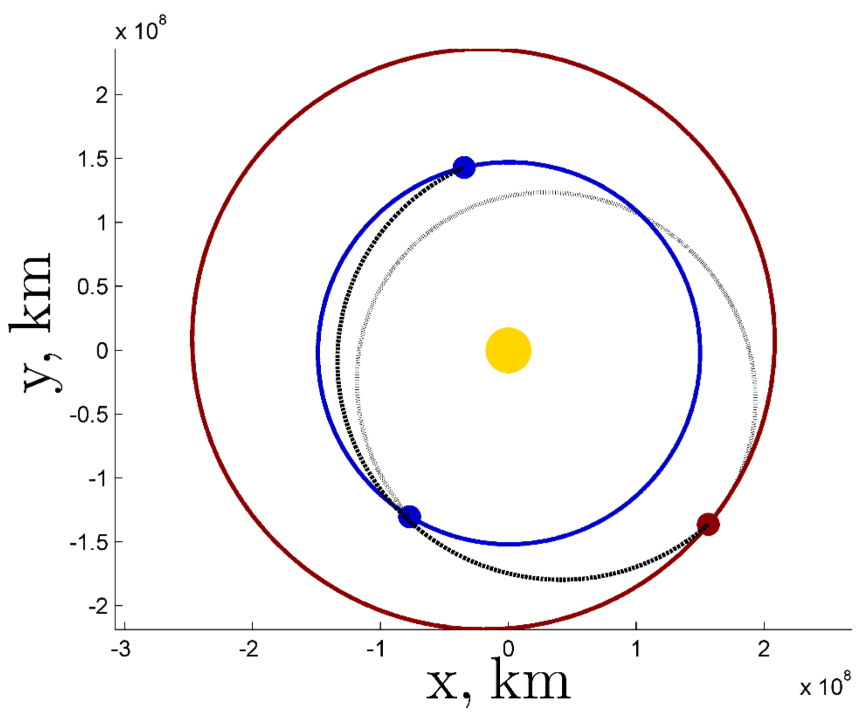

Figure 14 Ballistic free-return trajectory with Earth departure in January 2018, Mars flyby in August 2018, and Earth return in May 2019.

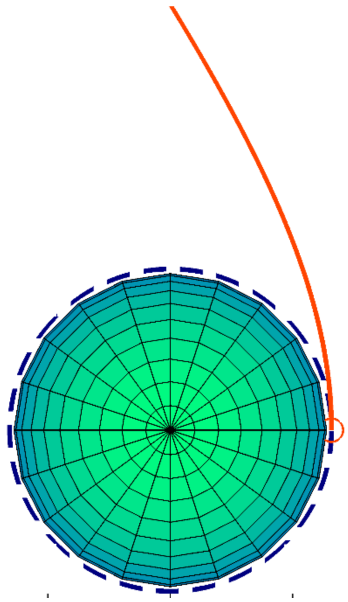

Figure 15 Earth departure and staging orbits.

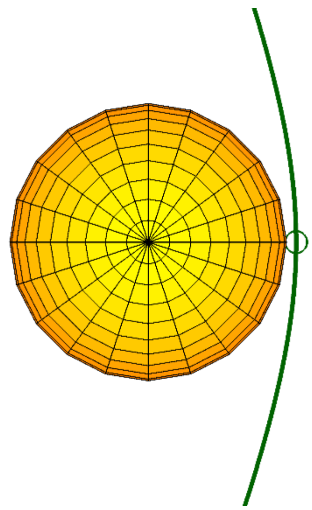

Figure 16 Mars close approach.

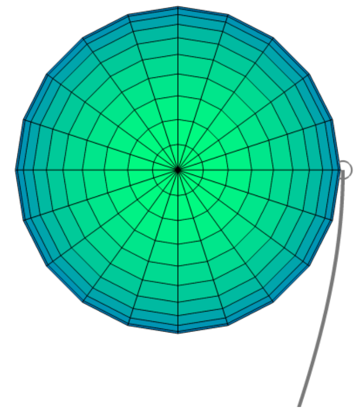

Figure 17 Earth return and re-entry.

4.3 Earth Launch to LEO

4.3.1 Launch Site

The selection of an appropriate launch site has significant implications for the achievable insertion orbits as well as the launch schedule. For example, the latitude of a launch site determines the minimum inclination that is achievable for the insertion orbit. In Table 1 are the launch sites we have considered in our investigation. Of the four launch sites; Cape Canaveral has the closest latitude (28.5°) to the axial tilt of the Earth with respect to the solar system ecliptic (23.6°). Thus, with proper launch timing, the launch payload can be inserted into an orbit appropriate for Earth departure onto a Mars-bound trajectory. While Cape Canaveral provides needed access to our desired staging and departure orbits, inclement weather is not an infrequent local occurrence, necessitating the scheduling of sufficiently wide launch windows to mitigate potential weather disruptions. However, in a best-case scenario, wide launch windows can provide additional on-orbit time for system checks prior to the desired TMI of January 4, 2018.

Table 3 Potential launch sites.

Launch Site	Country	Latitude (deg.)
Cape Canaveral	USA	28.5 N
Vandenberg	USA	34.8 N
Kourou	French Guiana	5.2 N
Baikonur	Kazakhstan	46.0 N

4.3.2 Launch Mass to LEO

The total mass of the spacecraft after TMI, that is the mass of the crew, capsule, and consumables, drives the sizing of the required propellant mass and associated propulsion system. We size the propulsion system in a three-step process:

1. Derive, via the Ideal Rocket Equation $\frac{\Delta V}{v_e} = \ln\frac{m_{sc}+m_{pi}}{m_{sc}}$, the required propellant mass (m_{pi}) to insert the spacecraft mass (m_{sc}) onto the Earth departure trajectory.
2. Size the propulsion system by applying a 15% fraction to the currently computed propellant mass, $m_{ps} = 0.15m_{pi}$.
3. Find the true propellant mass, m_p, from the Ideal Rocket Equation using the combined mass of the spacecraft and the propulsion system, $\frac{\Delta V}{v_e} = \ln\frac{m_{sc}+m_{ps}+m_p}{m_{sc}+m_{ps}}$.

Table 4 Launch vehicle and upper stage trade analysis, for spacecraft with and without 10% growth margin.

	Scenario 1 - Falcon Heavy		Scenario 2 - ACES / Falcon Heavy		Scenario 3 - SLS		Scenario 4 - Delta IV-H / ACES		Scenario 5 - Ariane 5 ME	
	No margin	Margin	No margin	Margin	No margin	Margin	No margin	Margin	No margin	Margin
Mass spacecraft [msc] (mT)	32.767	36.044	32.767	36.044	32.767	36.044	32.767	36.044	32.767	36.044
Isp (sec)	340	340	450	450	450	450	450	450	450	450
Propulsion system mass [mps] (mT)	16.229	17.852	9.886	10.875	9.886	10.875	9.886	10.875	9.886	10.875
Total propellant mass [mp] (mT)	161.780	177.959	85.791	94.371	85.791	94.371	85.791	94.371	85.791	94.371
Burnout mass after TMI [mbo=msc+mps] (mT)	48.996	53.896	42.653	46.919	42.653	46.919	42.653	46.919	42.653	46.919
Mass to LEO, before TMI [msc+mps+mp] (mT)	210.776	231.855	128.444	141.290	128.444	141.290	128.444	141.290	128.444	141.290
# Launches	**3.977**	**4.375**	**2.423**	**2.666**	**1.889**	**2.078**	**5.585**	**6.143**	**5.097**	**5.607**
Total cost for launches + TMI propulsion (million $)	**556**	**693**	**423**	**425**	**1,018**	**1,520**	**858**	**1,000**	**588**	**589**
mT to LEO per launch	53	53	53	53	68	68	23	23	25.2	25.2

The resulting summed mass of the crewed spacecraft as well as the propulsion system and propellant for the TMI is the total mass that must be delivered to the 200 km LEO staging orbit. We investigate several launch scenarios with margined and un-margined spacecraft masses as well as various launch vehicle options with distinct upper stage specific impulses. We consider four launch systems from different companies and agencies as well as one case where we consider the combination of a SpaceX launch system (Falcon Heavy) with a United Launch

Alliance (ULA) upper stage (ACES). Note that of the five launch vehicles considered, all require further technology development and demonstration flights prior to the launch date of January 2018. The total cost for each scenario is determined by adding the stated launch cost to LEO from the launch service providers to an estimated cost of the TMI propulsion based upon the dollar per mass ratio for the respective launch system upper stage.

4.3.3 Selection of Launch Architecture and Timeline

Of the launch scenarios considered, we select as our primary option the Falcon Heavy launch to LEO with the ACES upper stage. We choose this scenario based upon expected launch cost as well as estimated technology readiness level; while this configuration will require collaboration between competing launch service providers, the years leading up to the 2018 mission opportunity can be used to finalize technical and organizational details. The Falcon Heavy is expected to first launch in 2014, leaving several years to the human rating of the system, while the ACES system is built upon proven ULA hardware, whereas competing options such as SLS Block I (Block II will not be developed in time for 2018 launch opportunity) and Ariane 5 are not expected to have demonstration flights until 2017 at the earliest, leaving a slim margin for further testing of the system. The Falcon Heavy by itself is another cost effective option, however it requires significantly more launches and therefore adds to the complexity of the mission. While the Delta IV-H has a proven record of successful launches and the ACES configuration is designed to replace the current upper stages for this launch vehicle, the Delta IV-H is also more expensive than the Falcon Heavy. While we select the Falcon Heavy / ACES configuration as our primary launch scenario, we retain the other competing options as viable contingency options.

For both the margined and un-margined spacecraft, our baseline scenario requires a total of 3 launches, however only the final launch carrying the crew must occur within a short time frame before the nominal TMI. On the other hand, the cryogenic nature of the LOX / LH2 propellant that the ACES stage uses means that boil-off is a concern while the propellant is in the staging orbit.[5] Therefore, even though on-orbit cryogenic storage must be used for the fuel and oxidizer, launches of the storage tanks should not occur too early before the launch of the crewed capsule. Since boil-off of liquid hydrogen occurs at a faster rate than LOX, the LH2 tanks must be placed on the second propellant launch. We therefore propose the launch timeline in Table 5, where weather events at the launch site that could unduly affect the mission timeline are also accommodated for by scheduling a two-week window between each propellant launch. A shorter launch window is used for the crew in order to reduce the crew time in the staging orbit prior to TMI. Note that the last launch opportunity for the crewed launch is constrained to terminate one day prior to the nominal TMI so that on-orbit systems checks may be performed prior to TMI.

Table 5 Launch timeline.

Launch	Payload to LEO	Window
1	LOX propellant tank	Nov. 26th - Dec. 10th, 2017
2	LOX / LH2 propellant tanks	Dec. 10th – Dec. 24th, 2017
3	Crewed spacecraft + propulsion module	Dec. 24th, 2017 – Jan. 3rd, 2018

5 Aerocapture

To reduce the re-entry velocity the options of aerobraking and aerocapture were investigated. Table 1 in appendix A shows the comparison between both these aero-assist methods. Although aerobraking has been performed four times[6] in the past, it is not suitable for human missions and hence aerocapture was selected to reduce the re-entry velocity.

The General Mission Analysis Tool[7] (GMAT) developed by NASA Goddard was used to perform aerocapture analysis for this mission. The idea of aerocapture is to utilize Earth's atmosphere to provide an effective reduction in velocity such that the spacecraft shifts from a high-energy hyperbolic orbit to a low-energy elliptical orbit. This process is illustrated in Fig. 18. Based on various perigee altitudes, Earth-capture ellipses of different eccentricities are obtained as shown in Fig. 19.

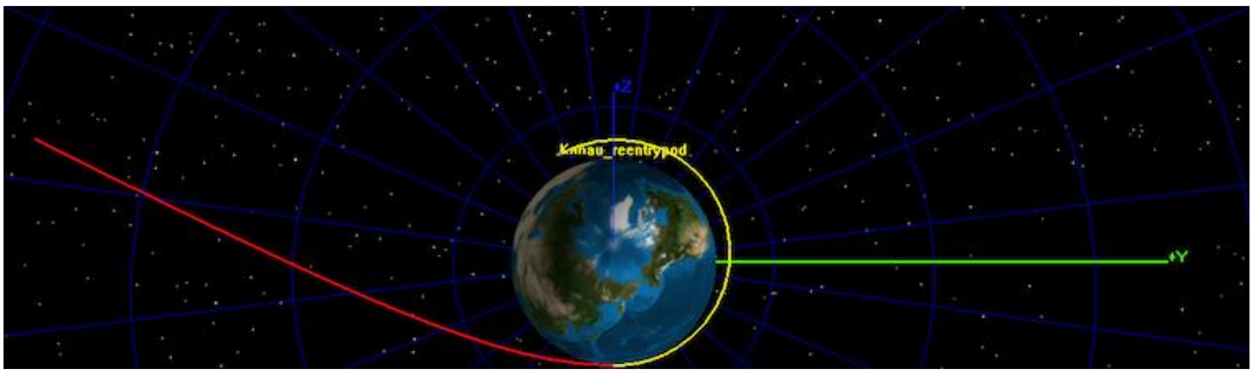

Figure 18 Transition from hyperbolic velocity to elliptical capture velocity.

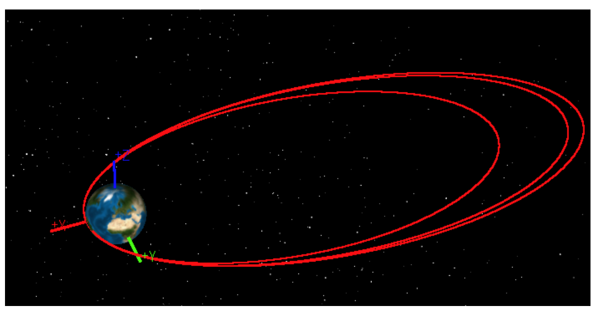

Figure 19 Various Earth-capture ellipses after performing aerocapture.

5.1 Sample Analysis

We analyzed a sample case, using data shown in Table 6, to demonstrate the usefulness of aerocapture for this mission. The relevant ballistic parameters of Space X's Dragon spacecraft were used as defined in Fig. 2 of appendix B. The MISE90 atmospheric model was used for this analysis (an error resulted while using the Jacchia-Roberts atmospheric model for Earth as shown in Fig. 3 of appendix B, thus this model could not be used for comparison). Our objective was to find the minimum possible reduction in velocity due to Earth's atmosphere and, thus, the eccentricity of elliptical Earth-capture orbit was deliberately kept high for this case. Due to the inability to access advanced software, heat analysis was not performed, therefore team Kanau suggests further heat analysis for determining both heat rate and heat load on the re-entry vehicle. Assuming a corridor width requirement of 0.7 degrees, vehicles with an L/D of 0.4-0.5 are found to be satisfactory for arrival speeds up to 14.5 km/s and g-loads lesser than 5 gs.[8] The current L/D ratio for Dragon is 0.18[9] , so we suggest that improvements be made to the current technology to increase this value to 0.4 for better results using aerocapture.

Table 6 Aerocapture sample analysis data.

Parameter	Value
Altitude at maximum velocity (km)	123.9468
Maximum velocity during re-entry (km/s)	14.1326
Perigee altitude (km)	56.3637

5.2 Results

The Altitude vs. Periapse Velocity plot results for the sample case is shown in Fig. 20. The results obtained are shown in Table 7. The run time for simulation was observed to be 4.4 seconds.

Table 7 Results for aerocapture sample analysis.

Parameter	Value
Velocity at ellipse perigee (km/s)	11.1029
Reduction in velocity (km/s)	3.0297
Initial eccentricity of capture orbit	0.99

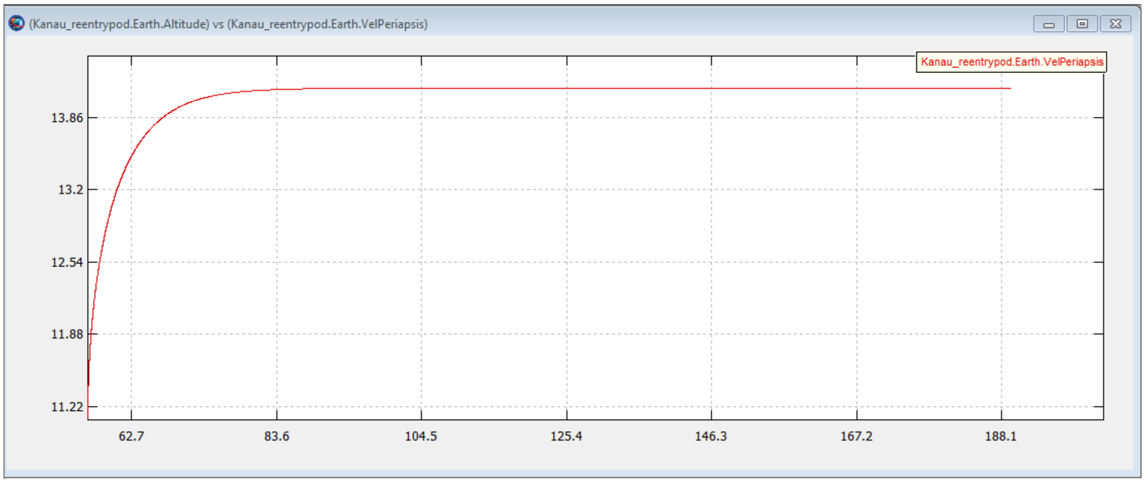

Figure 20 Results: Altitude vs. Periapse Velocity plot for a sample analysis.

6 Key subsystem architectures

6.1 Environment Control and Life Support System (ECLSS)

6.1.1 Essential Subsystems of ECLSS

We conducted the feasibility study for designing the Environment Control and Life Support System (ECLSS) with data and using our own simulator, SImulator for Closed Life and Ecology (SICLE).[15]The ECLSS system has four essential subsystems: Atmosphere Management, Water Management, Waste Management, and Food Supply. Each subsystem has the functions as shown in Fig. 21. We classify the key functions which must be handled uniquely for the Mars manned mission as red boxes while other functions which can be transferred from current International Space Station (ISS) technology as blue boxes.

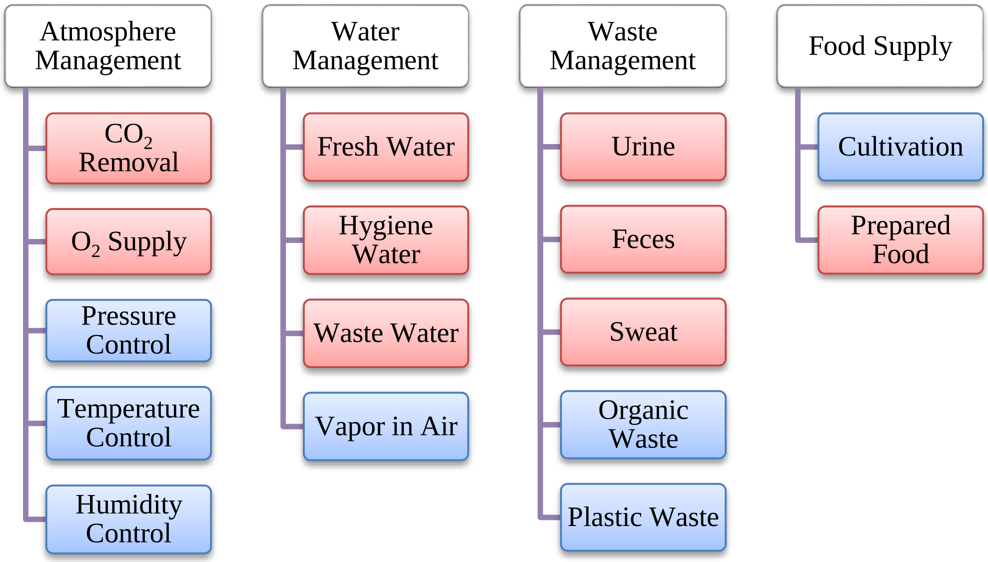

Figure 21 Functions for each ECLSS management subsystem.

6.1.2 Equivalent System Mass (ESM) and Tradeoffs

6.1.2.1 ESM for Mars Mission

In order to analyze various parameters of the ECLSS, the Equivalent System Mass (ESM) is used to perform comparisions. ESM is a method to conduct trade studies by converting diverse elements of a system such as volume and power to equivalent mass values so that comparison of diverse systems becomes easier.[10] The typical ESM equation is:

$$\mathrm{ESM} = \mathrm{M} + (\mathrm{V} \cdot \mathrm{V_{eq}}) + (\mathrm{P} \cdot \mathrm{P_{eq}}) + (\mathrm{C} \cdot \mathrm{C_{eq}}) + (\mathrm{CT} \cdot \mathrm{D} \cdot \mathrm{CT_{eq}}) \tag{1}$$

where

M = actual mass of the system [kg]
V = total pressurized volume of the system [m^3]
V_{eq} = mass equivalency factor for the pressurized volume infrastructure [kg/m^3]
P = total power requirement of the system [kW]
P_{eq} = mass equivalency factor for the power generation infrastructure [kg/kW]
C = total cooling requirement of the system [kW]
C_{eq} = mass equivalency factor for the cooling infrastructure [kg/kW]
CT = total crew time requirement of the system [CM-h/y]
D = duration of the mission segment of interest [y]
CT_{eq} = mass equivalency factor for the crew time support [kg/CM-h]

Since crew time is not as critical for our scenario when compared to the ISS missions for less workload, we neglect the CT term in the equation for our analysis. Other equivalency factors that we use are V_{eq} = 215.5 kg/m^3 (shielded), P_{eq} = 237 kg/kW, C_{eq} = 60 kg/kW.[11]

Table 8 shows the resources that humans require for survival and utilization, whereas Table 9 shows substances that humans produce in their day to day life. Based on these values, the following sections investigate ESM for 501 days of the Mars mission.

Table 8 Material consumption of human crew (kg/Crew Member-day).

	Minimum	Nominal	Maximum	Comments
Water				
Drinking		2.00[12]		
Food supply		0.50		
Hygiene (oral, hand, face)		4.45[12]		
Shower		2.72[12]		
Laundry		12.47[12]		
Urinal flush		0.49[12]		
Food				
Food	0.54[11]	0.617[11]	0.66[11]	
Packaging	0.08[11]	0.09[11]	0.10[11]	15% of food
Air				
Oxygen	0.385[11]	0.835[11]	1.852[11]	

Table 9 Material production of human crew (kg/CM-day).

	Minimum	Nominal	Maximum	Comments
Water				
Urine water		1.886[11]		
Fecal water		0.091[11]		
Respiration water	0.803[11]	0.885[11]	0.975[11]	
Perspiration water	0.036[11]	0.699[11]	1.973[11]	

Gray water		20.14		sum of utilized water
Solid Waste (dry basis)				
Fecal waste		0.032[11]		
Perspiration waste		0.018[11]		
Air				
Carbon dioxide	0.466[11]	0.998[11]	2.241[11]	

6.1.2.2 ECLSS currently used in the ISS

On board the International Space Station (ISS), water is recycled, oxygen is generated by electrolysis (Eq. 2) and carbon dioxide is reduced by a Sabatier system (Eq. 3).

$$2H_2O \rightarrow 2H_2 + O_2 \tag{2}$$

$$CO_2 + 4H_2 \rightarrow CH_4 + 2H_2O \tag{3}$$

In order to produce 0.835 kg of oxygen per person per day, theoretically, 1.88 kg of water is needed. If the hydrogen required to be combined with carbon dioxide in a Sabatier reaction is supplied from water electrolysis, only 1.15 kg of carbon dioxide can be reduced, while two persons create about 2.0 kg of carbon dioxide per day. The rest of the carbon dioxide must be dumped out. Another concern is that water electrolysis consumes about 0.94 kg of water per day over the amount of water produced by the Sabatier reaction. Thus this closed loop system requires 471 kg of water to be carried in the spacecraft to maintain oxygen for 501 days. Note that the ESM for the ISS systems is 10739.01 kg. Data for this subsystem is shown in Table 2 of appendix A.

6.1.2.3 Bio-regenerative Systems

Technologies used onboard the ISS for generating oxygen and removing carbon dioxide are based on separate subsystems such that it becomes difficult to sustain the appropriate balance of atmospheric components without much loss of intermediate products such as hydrogen and water. Another possible option for atmosphere recycling is using bio-regenerative systems. Common bio-regenerative ECLSS uses plants but their maintenance is difficult and requires a large amount of cultivation area. Therefore we suggest the use of an algal system for the mission. One cyanobacteria, Spirulina, is known for its high rate of photosynthesis and, furthermore, its activity can be controlled by temperature (it is most active at 30 degrees Celsius). A study by Minoo and Bernhard[13] shows that 200 L of Chlorella, an algal species, can supply enough oxygen to support one person. Fig. 22 shows one of the design concepts of an air revitalization system using algae or cyanobacteria. The prime concern with bio-regenerative subsystem is that currently there is no such system that has been tested in a spaceflight-worthy operational phase.

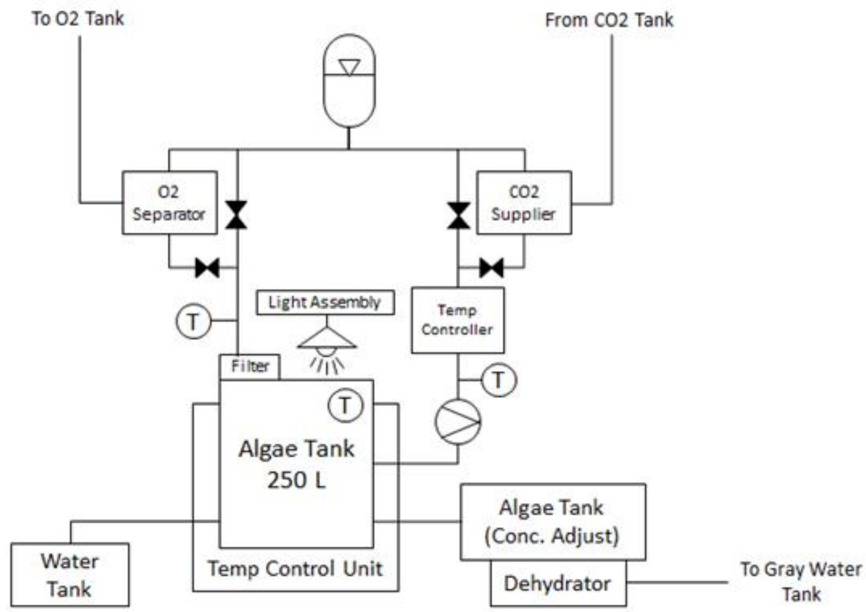

Figure 22 A concept of an Algal air revitalization system.

6.1.2.4 Non-regenerative Systems

Without utilizing the ISS water recycle system or atmosphere controlling systems, the quantity of water and oxygen that must be brought along, and the amount of carbon dioxide that would be absorbed needs to be determined. For the mission duration of 501 days, an ESM for a non-regenerative system is found to be 10739.01 kg. Data for this subsystem is shown in Table 3 of appendix A.

6.1.3 Selection of ECLSS for Mars Mission

To create highly reliable ECLSS such that two persons can safely return Earth after a 501 day journey, at least triple redundancy is required for each ECLSS management design. Although the most reliable selection is taking non-regenerative systems (storage), it will occupy large mass and volume leading to higher launch costs as calculations show in previous sections. In order to make the mission meaningful for future space exploration, relatively new and state-of-art systems should be integrated for technological demonstration with non-generative systems as back up. This hybrid life support system could be a combination of recycling technologies onboard the ISS with double redundancy and storage for half of the mission duration. The total main ECLSS mass becomes 2,304 kg as a result. Data for subsystem is shown in Table 4 of appendix A.

6.1.4 Food Management

Growing many crops or vegetables would be difficult within a small habitable area. Also, it would be an inefficient use of space for a crew size of two for the mission duration, so food supplies should be precooked foods such that it is ready to eat by just reheating or adding water. However, team Kanau suggests cultivation of some vegetables like lettuce that are easy to cultivate and require a small amount of space. This will not only provide a limited amount fresh food but also assist in the psychological health of the crewmembers as they tend to its growth and nurturing. Many types of home cultivation kits are available as shown in Fig. 23.

Figure 23 Lettuce home cultivation kit[14].

6.1.5 Simulation of ECLSS using our own simulator, SICLE

Some members of team Kanau have been developing a new program for ECLSS simulation, named the SImulator for Closed Life and Ecology (SICLE).[15] Two main advantages of SICLE are its user-friendliness and its ability to apply new models and functionalities. Users can easily design and follow their own system designs graphically by utilizing SICLE's GUI as shown in Fig. 24. Furthermore, it is able to analyze both closed and open loop systems. SICLE will be open to the public in the near future. Simulations and analysis of this mission using SICLE is ongoing.

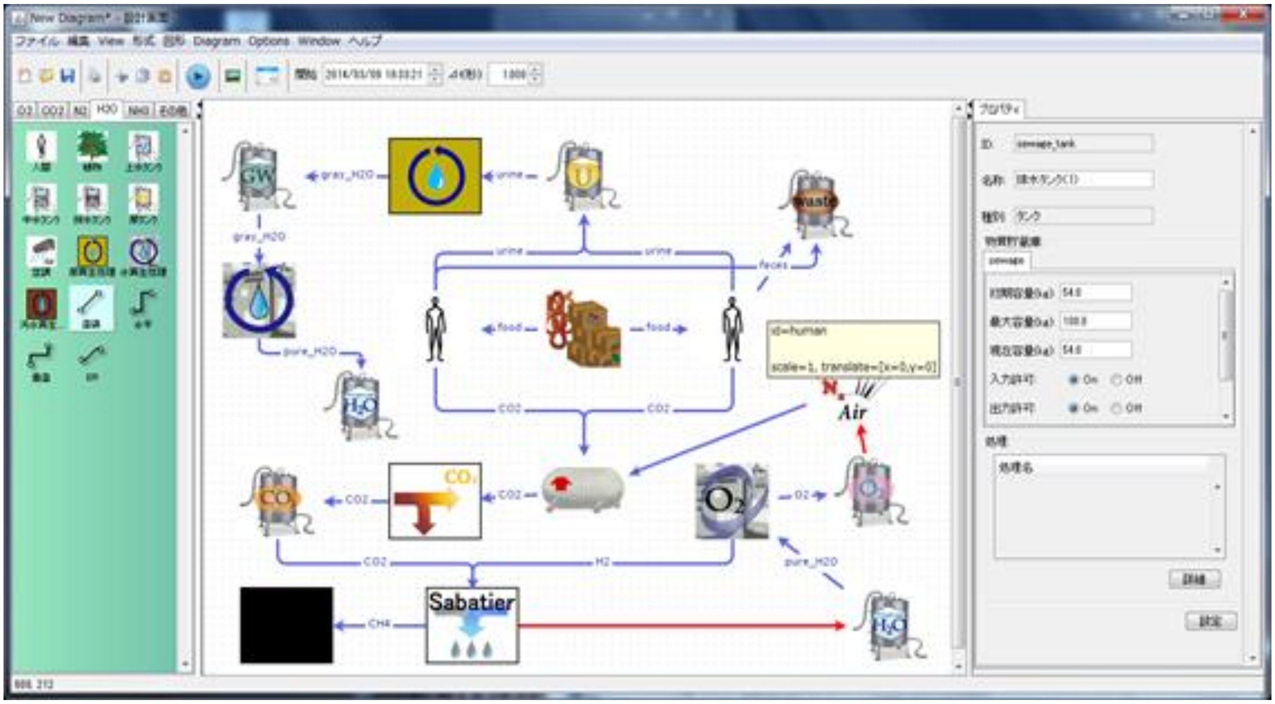

Figure 24 Sample image of SImulator for Closed Life and Ecology, SICLE.

6.2 Kanau Spacecraft's Interior Design

6.2.1 Requirements for Interior Design

The crews' physical and mental health must be maintained during the deep space mission in order to ensure a safe return to Earth. NASA's Space Flight Human System Standards, NASA-STD-3001, Volume 1 and Volume 2 address the human needs for space flight. One requirement is that the interior environment of spacecraft need to be designed to "support human perceptual and cognitive capabilities to meet system performance requirements".[16] Schlacht IL (2006) suggested that the interior of the spacecraft in a long duration mission should meet the following eight requirements- (1) safety, (2) visibility, (3) flexibility, (4) variation, (5) intuitive and friendly, (6) customization, (7) visual stimuli, and (8) Earthly stimuli.[17]

6.2.2 Baseline Architecture

Team Kanau proposes an interior design incorporating five components as shown in Fig. 25.[18,19]

Figure 25 Aspects of interior design.

6.2.3 Details and supporting data

6.2.3.1 Interior design of spacecraft

All potential dangers should be removed to ensure (1) safety during the mission, (2) visibility when crew participates in some activities, (3) flexibility to adapt to various situations, (4) variability because monotony of visual stimuli leads to strong discomfort, which is certified by past space missions like Skylab Space Station 4,[20] and (5) intuitive and friendly design**.**

6.2.3.2 Sleeping bag

Customization of spacecraft is needed because interviews of astronauts suggest privacy is the most important factor for crew.[21] The crewmembers of Inspiration Mars mission will be a long-time married couple and hence a sleeping bag for the two people should suffice. Physical intimacy is one important factor to maintain the healthy relationship of a married couple. Therefore, such a sleeping bag and placement of any cameras will be important considerations. Furthermore, the use of nano materials could reduce radiation exposure during sleeping, thus the sleeping area could also be a shelter in the case of a high radiation event.[18]

6.2.3.3 Lighting changes

Visual stimuli produced by interior light has the potential to affect crewmembers' psychological condition. For example, a new lighting system that produces artificial rainbow lights can be created by slowly rotating a transparent ring and a spot light as shown in Fig. 26.[19] Such an artificial light recreates natural color variations during the course of a whole day mimicking biological sunlight effects.[20]

Figure 26 Artificial rainbow generated by spotlight.

6.2.3.4 Sound Environment and Feeling

Earthly stimuli are also very important since humans that are confined in a small spacecraft for long durations are placed under great psychological stress. One effective method to avoid mental disorders is to lead life as if it is being lived on Earth. To mimic Earth's environment, soundscape design was considered for the Kanau spacecraft. For example, Nature sounds such as sound of a stream and a bird's twittering can be used as a sound therapy. [27] Such nature sounds are helpful for relaxation.

6.2.3.5 Plant Cultivation Box

To recall nature on Earth, the spacecraft can be equipped with plant cultivation boxes. These plant boxes invoke positive stimulation. After enjoying cultivation of the plants, crewmembers can eat them for additional variety in their diet. Cultivating plants in a spacecraft is a difficult task but imitations of wood, grain or artificial flowers can also help in mental relaxation of crewmembers.[19]

6.3 Facilities for crew physical health

6.3.1 Requirements for crew physical health

This system needs to provide measures to meet crew bone, muscle, sensory-motor, and cardiovascular standards defined in NASA-STD-3001, Volume 1. Measures shall maintain in-flight skeletal muscle strength at or above 80 percent of baseline values and bone mass consistent with requirements for a safe return to Earth's gravity.[22] The exercise and muscle data gathered from nine crewmembers while on the ISS for 6 months clearly support the notion that changes to the exercise prescription are necessary to protect skeletal muscle for long-duration space missions.[23]

6.3.2 Baseline Architecture

Facilities for crew physical health are composed of three components as shown in Fig. 27.

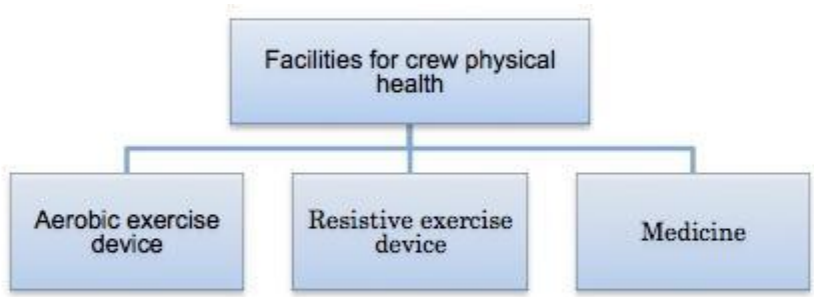

Figure 27 Outline of facilities for crew physical health.

6.3.3 Details and supporting data

6.3.3.1 Aerobic and resistive exercise devices [24]

U.S. crewmembers are required to complete a 2.5-hour bout of combined aerobic and resistance exercise on 6 of 7 days during the mission. On board the ISS, approximately 1.5 hours were devoted to resistive exercise on the interim resistive exercise device (iRED) and 1 hour was devoted to either the Treadmill with Vibration Isolation System (TVIS) or the Cycle Ergometer with Vibration Isolation System (CEVIS) or a combination of the two. We suggest implementation of exercise devices such as those in use on the ISS, whilst also being open to new developments in astronaut exercise regimes.

6.3.3.2 Medicine

Medicine can be used to prevent the loss of bone mass and formation of kidney stones during long stays in space with a bisphosphonate formula used in the treatment of osteoporosis. Such medicines have been experimentally used by the ISS crewmembers and can be explored for this Mars mission.[25]

6.4 Facilities for Radiation Protection

6.4.1 Requirements for Radiation Protection

Crew occupational exposure to ionizing radiation should be managed through system design and the application of appropriate countermeasures.[26] Countermeasure for cosmic radiation is one of the most critical issues for long duration, deep space missions. Radiation may cause cancer and other types of tissue damage to the crew. In addition, the crew must evacuate to a shelter for several hours to one day during the event of a solar storm.[27]

6.4.2 Baseline Architecture

Facilities for radiation protection comprise of two components as shown in Fig. 28.

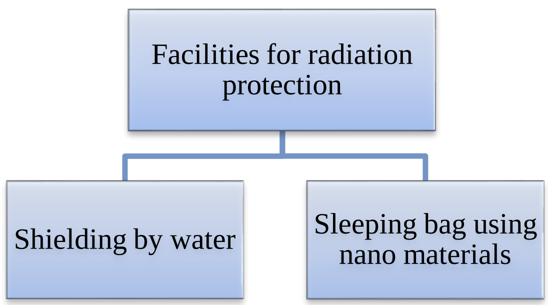

Figure 28 Outline of facilities for radiation protection.

6.4.3 Details and Supporting Data

6.4.3.1 Shielding Water

Using the consumable water for the additional purpose of radiation shielding offers potential mass conservation advantages. It is recognized that many schemes may impose requirements that exceed practical water mass allowances. Water tank locations will entail strategic planning implications as well.[27] In addition, installing water shielding material on a stack board consisting of hygienic wipes and towels inside the spacecraft will reduce the space radiation dose for crews.[28]

6.4.3.2 Sleeping Bag using Nano materials for Radiation Reduction[27]

Humans need to sleep for several hours a day. The use of Nano materials that reduce radiation during this period of extended inactivity is recommended. Additionally, the sleeping room could be a shelter, so that the crew may feel comfortable within the environment. It demonstrated that materials like nano foams may be designed to tolerate radiation exposure.[29] Such materials would offer significant advantages for space applications providing major cut offs in mass while offering multiple structural applications.

6.5 Command & Data Handling Subsystem[30]

The Command and Data Handling (C&DH) subsystem, essentially the brain of the spacecraft, performs the following:

- manages all forms of data on the spacecraft
- carries out commands sent from Earth
- prepares data for transmission to Earth
- manages collection of solar power and charging of the batteries
- collects and processes information about all subsystems and payloads
- keeps and distributes the spacecraft time
- carries out commanded maneuvers
- autonomously monitors and responds to a wide range of onboard problems that might occur.

The key parts of this system are:

- Space Flight Computer: Consists of next generation of space-qualified processors. The team suggests use of RAD5500 PowerPC processor that is 10 times faster than RAD750 processor.[31]
- Flight Software: The Flight Software is an integral part of the Space Flight Computer, and includes many applications like Fault Protection running on top of an operating system.
- Solid State Recorder: The primary storage for science instrument data onboard the spacecraft. The science data is stored on this recorder until it is ready for transmission to Earth, and then is overwritten with new science data.

6.5.1 Requirements[30]

- Perform all functions requested of a command and control module in a complex spacecraft.
- Provide continuous audio and video link with the ground station.
- Provide internet-style connection.

6.5.2 Baseline Design[30]

Several distributed units called Control and Data Management Units (CDMU) are used to implement DCDMS (Distributed Control and Data Management Systems). The baseline concept consists of modular functions listed in Table 10.

Table 10 Modular functions and their configuration for DCDMS.

Modular function	**Configuration**
Processor module	Includes digital interfaces.
Telemetry transfer frame generator	Directly interfaced with transponders.
Reconfiguration module	Two modules always powered, one of which is a master clock and the other acts as a backup processor or spare.
Distributed memory module	Contains VRAM and NVRAM modules. NVRAM acts as a safeguard memory.

The crewmembers should have available mass memory of the order of several terabytes, including dedicated memory for personal use.

6.5.3 Budgets[30]

The expected C&DH mass and power budget for this mission has been evaluated as shown in Table 11.

Table 11 C&DH budgets.

Property	Type of module	Value
C&DH + harness mass	Inhabited	650 Kg
Percentage of C&DH + harness mass	Unmanned	5%
Power, per module	Manned	2200 W
Power, per module	Unmanned	400 W

6.6 Communications

The overall communications must support the full scope of Mars flyby mission, including launch, Earth orbital operations, trans-Mars injection (TMI), Earth-Mars cruise, Earth return, and Earth arrival. Meeting these mission phases would require the combined capabilities of the Space Network for initial near-Earth support, the Deep Space Network (DSN), and dedicated Mars network assets as shown in Table 12.

Table 12 Communication for a Mars flyby mission.

Mission phase	Network	Services	Bands utilized
Launch through TMI	NASA Space Network	Tracking and Data Relay Satellite System (TDRSS)	S-band and Ka-band
Earth-Mars-Earth cruise	NASA DSN		X-band for basic telemetry link and Ka-band or laser for high-rate link

6.6.1 Requirements[30]

- Support Tracking, Telemetry and Command (TT&C) communications during all mission phases and at any attitude.
- Two-way ranging and Doppler capabilities during all mission phases.
- Support a maximum range of 2.7 A.U, which is the maximum distance between Earth and Mars.
- Selectable telecommand (TC) and telemetry (TM) data rates.
- Optimized data rates based on realistic assumption of on-board equipment and ground segment availability.
- Range of data rate requirements is shown in Table 13. Determine maximum data rate based on cost, complexity and technology readiness level (TRL).

Table 13 Data rate requirements for TV.

	Uplink	Downlink
Maximum Data Rate (overall, kbps)	11280	9232
Average Data Rate (overall, kbps)	3484	1436
Minimum Data Rate (overall, kbps)	160	160

6.6.2 Band and Frequency Design[30]

Laser communications have been used only for downlink while Ka-band has been used for uplink because using laser as an uplink is too expensive using current technology. Additionally, Ka-band data rates are higher for uplink than for downlink, mainly because of the higher transmitted power by the ground station (G/S). For contingencies, X-band has been used because it has less weather dependence than Ka-band and hence higher availability.

For the TV-relay satellite link, X-band has been chosen, since the pointing requirement is lower than Ka-band and it has a high enough data rate. The bands and frequencies used are consistent with the Space Frequency Coordination Group (SFCG).[32] Table 14 summarizes this design.

Table 14 TV antennas in a nutshell.

Kind of antenna	Quantity	Band	Gain (dBi)	Minimum required pointing precision	Size	Radiated power (W)	Data rate Uplink	Data rate Downlink	Steering mechanism (hemispherical)	Comments
Telescope	1	Optical		2μrad	30.5 cm	5	No uplink	10 Mbps	180°	LASER link, only used for downlink
Dish antenna	1	Ka-Band	59.1	0.01 deg	3m	65	1.8 Mbps	1.5 Mbps	180°	Main link.
Medium gain antenna (MGA) Patch	2	X-band	18	20 deg	8.2 x 8.2 x 2 cm	65	22 Kbps	460 bps	180°	Intelligence to point to the Earth in a contingency case, even with loss of TV attitude
Dish antenna	1	X-band	30	2.25 deg	45 cm	65	30 Mbps	30 Mbps	180°	Link with relay satellite

6.6.3 Ground station assumptions[30]

G/S with Ka-band and X-band capability and 70 m of antenna diameter are used. Their characteristics are described in Table 5 in appendix A.

6.6.4 TV contingency communications

Possible contingency scenarios have been discussed in Table 15.

Table 15 Contingencies and proposed solutions.

Contingency scenario	Proposed solution
Laser downlink cannot be used.	Use Ka-band 3 m antenna.
High gain antenna cannot be used because of loss in attitude.	Use two MGAs with an intelligent steering mechanism so that it will be pointing one of the antennas to the Earth.

6.6.5 Suggested Options

1) Ka+ band

Pros

- Can be used for both Mars and near-Earth missions.
- Has improved linking capacity than Ka band.

Cons

- No technological development thus far.
- Atmospheric and rain attenuation is higher than Ka- band.

2) Laser link coding

Pros

- Has higher net data rate.

Cons

- Four times higher bit rate is required after coding than before coding. Technology does not exist for such a high bit rate.

6.6.6 Budget

Table 16 illustrates the communications budget for this mission.

Table 16 TV communications budget.

Unit	**Number of units**	**Unit mass (kg)**	**Total Mass (kg)**	**Power (W)**
Optical transmitter	2	20.0	40	150.0
Optical transmitter device (telescope)	1	25.0	25	
Ka-band transponder	2	6.5	13	160.0
Ka-band antenna (3m)	1	35.3	35.3	
X-band transponder	2	6.5	13	100.0
MGA (X-band), patch	2	0.6	1.2	
UHF patch antenna	1	1.0	1	
UHF transceiver	2	2.5	5	16.5
X-band dish antenna (0.45 m)	1	1.0	1	
Harness			21	
Total:			155.5	426.5

6.7 Power Systems

6.7.1 Source Selection

Long duration spaceflight currently presents only two options for power supplies: Photovoltaic and nuclear. While nuclear offers several advantages in terms of power density and reliability, no nuclear system has yet been developed for extended human spaceflight. In addition, photovoltaics have demonstrated flight heritage on the ISS, providing lifetime and power levels in excess of those required for the IM mission.[33]

Based on these considerations, and given that the abbreviated schedule allows for no time to develop novel nuclear space systems, we judge photovoltaics as the superior choice for powering the IM spacecraft systems.

6.7.2 Power System Design Factors

The requirements to support a crew for mission duration >1 year suggests a useful benchmark of comparison for designing the IM vehicle power systems is the ISS. Design of the power system shall thus be based on the established heritage of the ISS whenever possible. However, there are five critical considerations for the design of the IM spacecraft that distinguish it from the ISS as shown in Table 17:

Table 17 Critical considerations for power system.

	ISS	**IM**
Crew/Spacecraft size	3-6 crew, 900 m^3	2 crew, 82 m^3
Eclipse frequency/duration	45 minutes eclipse, Eclipse every 1.5 hours	No periodic eclipse, .85 hours eclipse at Mars encounter
Available insolation	Constant, ~1344 W/m^2	Variable, 525-2600 W/m^2
Spacecraft orientation	Fixed Geocentric	Free
Required array lifetime	>7 years in LEO environment	<1 year in LEO, 1.4 years in interplanetary

The above factors will have effects on the design of the power system as given below:

- Smaller crew and vehicle size will translate to smaller power and redundancy requirements.
- Uninterrupted solar power will be available for the entire mission, with only a single eclipse occurring during Mars flyby, drastically reducing the charge/discharge cycle requirements on the batteries.
- Solar insolation will not be constant throughout the mission, varying from 1340 W/m^2 at Earth departure, 525 W/m^2 at Mars flyby, and 2600 W/m^2 at perihelion. Power collection and storage systems must be designed to accommodate this wide range of insolation.
- The spacecraft is under no obligation to maintain a fixed orientation with respect to Earth, allowing for the option of fixed or single axis steered arrays.
- Exposure to greater solar proton flux in interplanetary space will result in increased performance degradation of photovoltaics. However, shorter mission length, and the prevalence of high insolation towards the end of the mission may mean that more rapid performance degradation is acceptable.

6.7.3 Power Budget

The total vehicle power budget is given in Table 18.

Table 18 Preliminary vehicle power budget.

Power Budget (W)[34,35]	Nominal		Transient Peak		Emergency	
ECLSS	**3689**	**63%**	**7882**	**72%**	**1179**	**47%**
Air	1870	32%	2626	24%	60	5%
Water	193	3%	529	5%	0	0%
Food	39	1%	1860	17%	0	0%
Cabin Thermal	99	2%	300	3%	99	8%
Waste	7	0%	174	2%	0	0%
Other	331	6%	823	8%	0	0%
Avionics	**1150**	**20%**	**1570**	**14%**	**1020**	**41%**
GD&C	1000	17%	1200	11%	1000	40%
Comm	150	3%	370	3%	20	1%
TCS	**1000**	**17%**	**1500**	**14%**	**300**	**12%**
Total	**5839**		**10952**		**2499**	

6.7.4 Power System Design Decisions

6.7.4.1 PV-type:

Based on a tradeoff analysis, emphasizing flight heritage and reliability, the best photovoltaic type was identified as the conventional Si cell, which has demonstrated long duration success on board the ISS.[33,36]

6.7.4.2 PV-sizing and number:

The power requirements in Table 18 were used, along with the specifications of common Si PV-cells to generate total PV-panel area of 64 m^2. Assuming the arrays remain unshaded, and optimally oriented with respect to the sun, this area will generate 6500 W at minimum insolation (525 W/m^2). This is sufficient for emergency and nominal operations, and 60% of the transient peak load.

However, at maximum insolation, this system is over designed, placing additional loads on the shunt regulator and thermal control systems. This problem can be avoided by distributing the PV area among multiple arrays, and stowing certain arrays when excess insolation is available. Distributing the 64 m^2 among four 16 m^2 arrays will allow sufficient power to be generated with a single array at perihelion, and two arrays at Earth departure and return. This has the added benefit of increased system redundancy in the later phases of the mission, when breakdowns and failures are most likely to occur.

6.7.4.3 Battery Selection:

The selection of a NiMH battery chemistry for the ISS is driven by their high energy density and charge cycle lifetime.[33,36] However, Li batteries offer superior energy density, and are superior to NiMH in virtually every respect except for charge cycle lifetime. The lack of regular eclipses in the IM mission means the batteries are unlikely to be extensively cycled, and superior energy density becomes a decisive advantage. Li-Ion or Li-Pol chemistry batteries are identified to be the best alternative battery chemistry for powering the IM spacecraft systems. Based on the required power levels, and the duration of an expected emergency and eclipse, four 8-cell 28V Li-Ion batteries are identified as sufficient to provide the current, power and energy requirements for the IM spacecraft.

6.7.4.4 PV-configuration:

Fig. 29 shows the various PV-geometries that were considered. Alternatives 5 and 6 were identified as optimal based on the level of redundancy, and flexibility under variable insolation. This geometry also does not *require* the arrays to be fixed, as the spacecraft would assume their optimal power orientation naturally, to minimize the load on the thermal control systems (see section on thermal design).

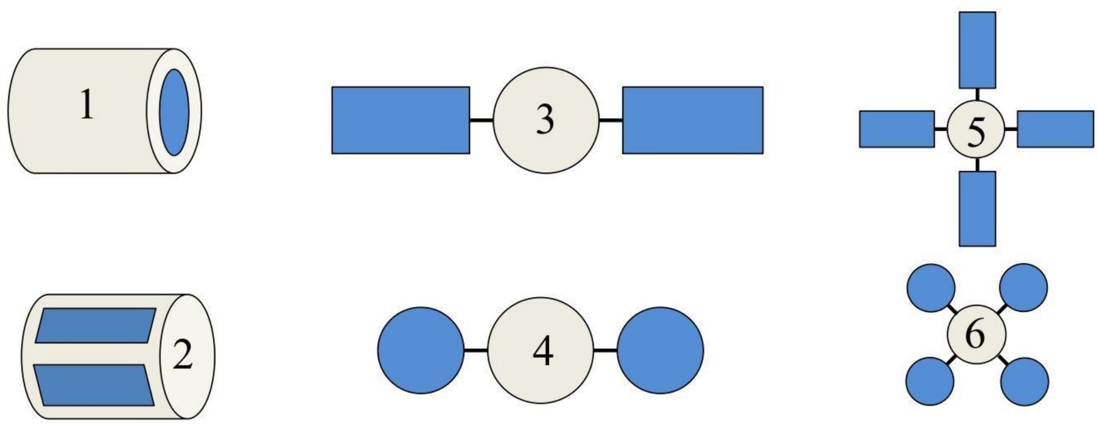

Figure 29 Various PV geometries.

6.7.4.5 Power System Architecture:

The required solar array size and required power reserves implied including 4 of each element. In keeping with conventions established in Shuttle/Station operations, these systems were designed to incorporate two independent DC power buses. The top-level system architecture is shown in Fig. 30.

In this architecture, each photovoltaic is isolated via a combination switching regulator/charge controller. These regulators monitor PV and battery voltage, and regulate the charge cycle and voltage of each battery. Switches 1-4 permit regulated battery power to be directed to either DC bus 1 or 2 as required, with nominal operation dedicating two batteries per bus. When the load from a given battery is not required, charging takes place. When charging is complete, excess power is rejected via a shunt to the heat exchanger.

Nominal operation calls for direct power supplied to buses 1 and 2 from photovoltaics 1 and 4, with batteries 1 and 4 used to provide transient loads as required. Photovoltaics 3 and 4 are used in this mode to maintain charge of batteries 3 and 4, with surplus power dissipated via the heat exchanger.

This architecture is designed to be highly flexible and provide complete double fault redundancy. During low insolation portions of the mission, a single component may fail and nominal power levels can still be maintained, while the fault of any two components will still permit emergency power to be available.

During high insolation portions of the mission, the entire spacecraft can be powered from a single photovoltaic/battery, with transient power provided by a second battery is required. Any two components may fail and nominal power will still be maintained.

In the event of an entire battery system failure, emergency power is available anytime in the mission via regulator 5, which provides the option of supplying unregulated power from the solar arrays. Such a contingency would not permit transient loads, and should be considered as a backup system in the event of catastrophic battery damage.

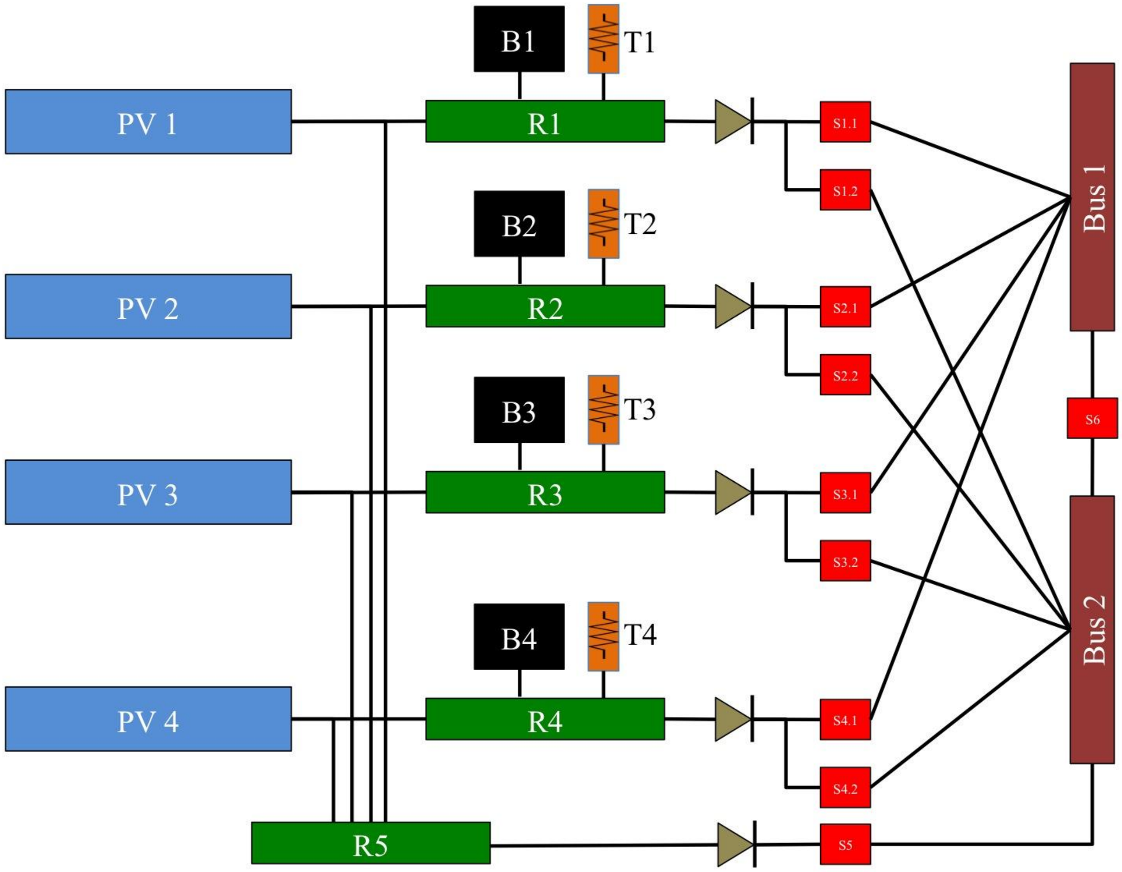

Figure 30 Top-level system architecture.

PV: Photovoltaic Array
R: Switching Regulator/Charge controller
B: Li-Ion Battery
T: Shunt Regulator to heat exchanger
S: Switch
Bus: 28V DC Bus.

6.8 Thermal Control Systems

Due to the relatively constant heat generation of the spacecraft, but the highly variable solar insolation, completely passive thermal management is not possible. Thus, the spacecraft will have two methods of thermal management:

6.8.1 Active Thermal Control:

This is normal mode of operation. Spacecraft maintains "sun astern" orientation, with the axis of the spacecraft pointed towards the sun, ensuring maximum power generation with minimal PV slewing, and presenting minimum illuminated surface for minimum thermal load. As on the ISS, heat is transferred from all systems, including the solar arrays via ammonia and water based coolant loops, where it is rejected through heat exchanger unit(s) on the shaded "front" end of habitat.[37] Precision modeling of this system is not possible at this point in the design, as it is extremely sensitive to the size of the spacecraft, and the insulation of the pressure hull. A rough first order model of the heat absorbed by the solar arrays and generated by the avionics, suggest the heat generated by the spacecraft could be rejected by a radiator with area 10 m^2 at an operating temperature of only 100 °C. Future work will focus on a refined thermal model of the spacecraft, and verifying that this approach will be sufficient.

6.8.2 Passive Thermal Control:

Passive thermal control is used for short periods, during times when the spacecraft cannot hold attitude with respect to the sun (course corrections maneuver, solar proton events, Mars flyby etc.) or when insufficient power is available for ATC. Photovoltaics are stowed and power is supplied from the batteries to minimize thermal load on the spacecraft. A slow rotation is adopted to distribute the absorbed heat evenly across the exterior of the spacecraft. Heat rejection in PTC occurs principally through radiation from the shaded side of spacecraft. Detailed analysis of this technique is impossible without further understanding of the vehicle size and systems. Future work will focus on the design of the pressure hull and the cabin insulation to ensure that a safe interior temperature can be maintained for at least two hours in the event of a complete TCS failure. The working of ATC and PTC is shown in Fig. 31.

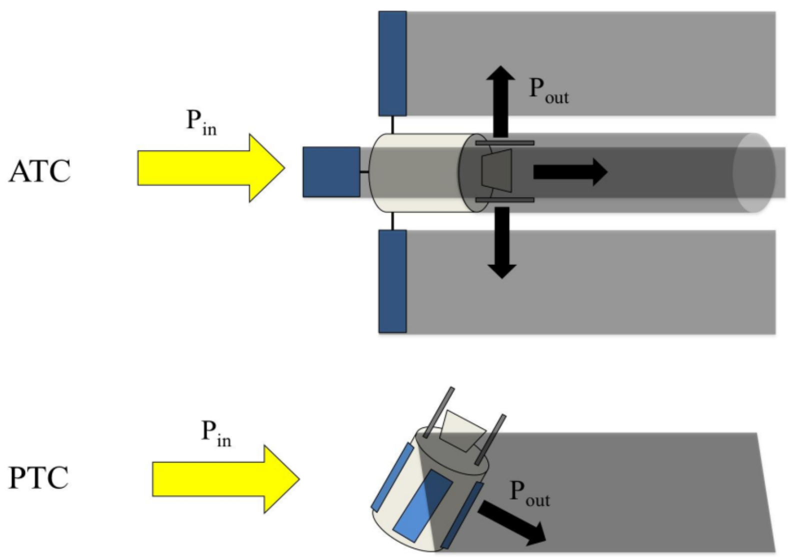

Figure 31 Active and passive thermal control processes.

6.9 Payload mission

Scientific experiments are also important for this mission to make it more meaningful. We propose two kinds of experiments ideal for this long mission.

6.9.1 Baseline Architecture

There are four kinds of experiments in our proposal as shown in Fig. 32.

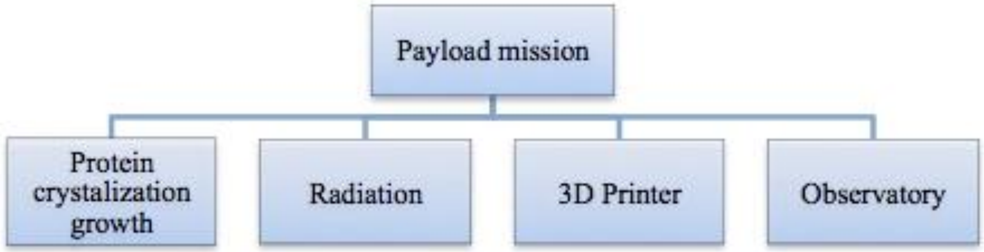

Figure 32 Examples of payload mission.

6.9.2 Details and Supporting Data

6.9.2.1 Protein Crystal Growth Experiment

The microgravity environment, in which neither thermal convection nor sedimentation occurs, is ideal for growing high-quality protein crystals as demonstrated on the ISS. There are many kinds of proteins with various functions.[38,39,40] High quality protein structural information plays a key role in understanding the biological structure-function relationships and in the development of new pharmaceuticals.[41] Thus, the astronaut crew can perform experiments similar to those conducted on the ISS, but in a different environment for longer time.

6.9.2.2 Radiation

Measuring radiation data on spacecraft for long mission will be good reference data for future manned missions. The data analysis for human spaceflight has been limited over the decades. However, several research experiments to monitor the radiation dose have been performed at the ISS.[42] Team Kanau suggests carrying both PS-TEPC and RRMD3 to measure radiological dosage, where this information can be used to plan further manned mission to Mars.

6.9.2.3 3D Printer

Since this deep space mission will require the crew and vehicle to be entirely self-sufficient, an on-board method to repair equipment is necessary. A 3D printer will be ideal for making necessary parts and tools on the spacecraft due to the limited quantity of spare parts and tools. NASA is planning to send first 3D printer to space in 2014, thus increasing the TRL of this technology to a level suitable for inclusion in the 2018 flyby mission.[43]

6.9.2.4 Observatory

During this long mission, the crew will need mental refreshment. Observation outside the windows using telescope will be fascinating activity for crew in the spacecraft and thus should be included for both scientific observation and health reasons.

7 Safety Analysis and Design

7.1 Safety Requirements[44]

The following were defined for determining safety requirements.

7.1.1 Mission Success: to perform a flyby of two crewmembers around Mars and return them safely to Earth.
7.1.2 Safety Goal: to identify all possible safety hazards, to eliminate/control them to an acceptable level during all the phases of the mission.
7.1.3 Probabilistic Goals: to have risk requirement of around 0.5% for human mission to Mars.

Table 19 describes the safety requirements for the Inspiration Mars mission.

Table 19 Safety requirements for Inspiration Mars mission.

Failure Category	Definition	Failure Tolerance Level
Catastrophic	Disabling or fatal personnel injury, loss of elements of vehicle stack or major ground facility.	Double-failure tolerant
Critical	Non-disabling personnel injury, major occupational illness, loss of elements not in critical path.	Single-failure tolerant
Marginal	Damage to emergency system, minor personnel injury or occupational illness.	

Other concepts required are given below:

- **The Fail Op/Fail Op Concept**: The system or function to which this concept is applied maintains functionality after the first and second failure.
- **The Fail Op/Fail Safe Concept**: The "critical" system or function to which this concept is applied maintains functionality after the first failure but not after second.
- **The Fail Safe/Fail Safe Concept**: The functionality is not maintained after any failure but no hazardous consequence occurs after two failures.

7.2 Abort options[45]

Detailed investigations of Martian human mission risks have not yet been performed. Abort options are designed into the mission for as many phases as possible to achieve acceptable risks as shown in Table 20.

Table 20 Abort possibilities.

Phases	Options
Earth Departure	Return to Earth possible
Early Part of Transfer to Mars	No practical abort scheme
Later Part of Transfer to Mars	No practical abort scheme
Transfer to Earth	Continue normal return to Earth

7.3 Risk acceptability[45,46]

For human missions to Mars the estimated risk reduction potential is shown in Fig. 33. Table 21 illustrates the risk acceptability for the mission. Based on these data a risk reduction strategy can be formulated. Large uncertainties exist regarding physiology and psychology of the crew due to the lack of previous experience and information available.

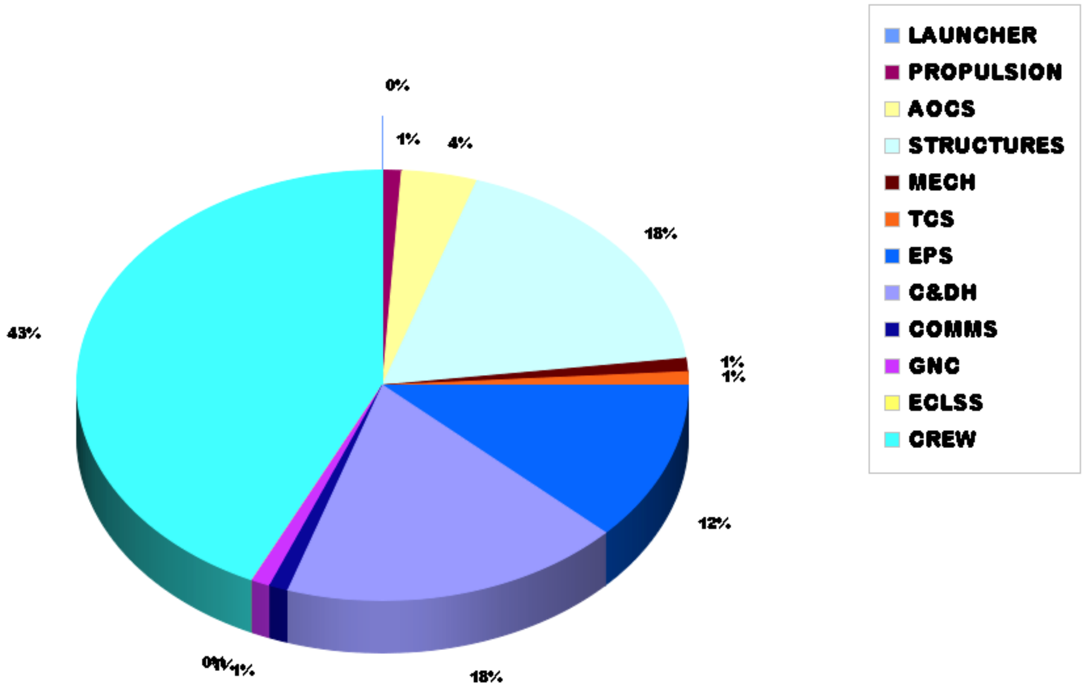

Figure 33 Estimated risk reduction potential per subsystem for human Mars mission.

Table 21 Risk acceptability.

Risk acceptability	Risk domain & scenario	Reason-status
Unacceptable	1. Maximum likelihood with catastrophic consequences: 1. Human factors inadequate to mission. 2. Inadequacy to radiation environment. 3. ECLSS failure. 2. Maximum likelihood with critical consequences: Failures during assembly, integration and verification (AIV) activities	• Numerous critical areas with uncertain environment definition. • Research level only. • New project beyond the state of the art. • High level of autonomy required for operations. • Highly complex program.
Acceptable if reduction impossible	Medium likelihood with critical consequences. Communications loss.	• Qualified technologies but never applied in projects. • Numerous modifications of qualified product.
Acceptable	Others	Defined environmental conditions, qualified products, existing processes & facilities.

7.4 Risk assessment[47]

The team used the open source software OpenFTA[48] to develop risk assessment architecture as shown in Fig. 34 and perform fault tree analysis (FTA) in the following manner:

1) Identification of undesirable situations.
2) Identification of functional level causes for these undesirable situations.
3) Identification of components that cause functional level losses.
4) Determining probability of occurrence of risks.
5) Constructing fault tree and improving the design.

Monte Carlo simulations were performed and the probability for the occurrence of an undesired situation were evaluated. To improve the design and make it safer team suggests the use of reference 49, which is a standard safety document for the ISS operations.

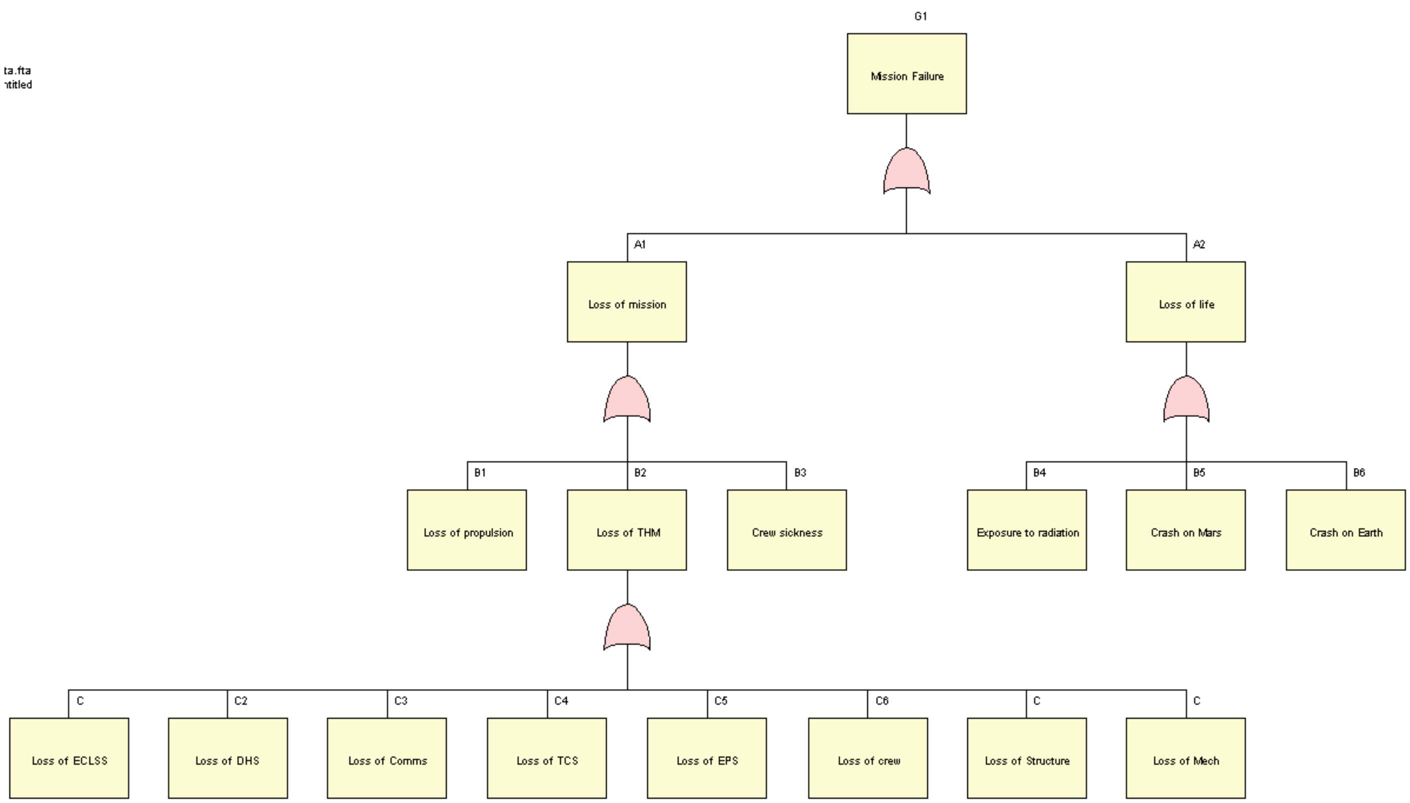

Figure 34 Risk assessment architecture.

8 Crew Selection from American Astronauts

Since the crewmembers will be a married couple, team Kanau constructed Table 22 shown below to list all possible candidates for crewmembers from American astronauts and then select the best option for this mission. Based on the parameters used in this table the team selected Shannon Walker and Andrew Thomas as the crewmembers. While couples with longer marriages will likely have a higher level of bonding, the statement made by Karen Nyberg indicates that astronaut couples that have children will likely refrain from joining this mission.[50]

Table 22 Candidate American astronaut couples list for Inspiration Mars mission.

Astronaut Wife (Status)	Astronaut Husband (Status)	Marriage status	Number of children	Years of marriage
Anna L.Fisher (active)	William F. Fisher (retired)	Divorced	2	-
M.Rhea Seddon (retired)	Robert L. Gibson (retired)	Married	3	33
Sally K. Ride (deceased)	Steven Hawley (retired)	Divorced	0	5
N.Jan Davis (retired)	Mark C. Lee (retired)	Divorced	-	-
Linda M.Godwin (retired)	Steven R.Nagel (retired)	Married	2	-
Tamara E.Jernigan (retired)	Peter J. Wisoff (retired)	Married	-	-
Bonnie J. Dunbar (retired)	Ronald M. Sega (retired)	Divorced	0	-
Shannon Walker (active)	Andrew S. Thomas (active)	Married	0	9
K. Megan McArthur (active)	Robert L. Behnken (active)	Married	1	-
Karen L. Nyberg (active)	Douglas G. Hurley (active)	Married	1	-

9 Conclusion

Team Kanau's architecture for a two-person flyby mission of Mars in the year 2018 presents opportunities for the incorporation of many intriguing technologies and techniques for the 501 day crewed journey. We propose astronauts Shannon Walker and Andrew S. Thomas, a married couple of 9 years, as the crew of the spacecraft. Three Falcon Heavy launches will deliver the crewed capsule, along with the required ACES propulsion stages for the Trans-Mars Injection, to a LEO staging orbit; this launch program relies only upon current or near-term technologies from SpaceX and United Launch Alliance, and so reduces the risk of delays to the schedule and offers significant cost savings compared to other available launch systems. Aerocapture upon return to the Earth will mitigate high re-entry velocities and enable the use of a SpaceX Dragon capsule for the crew re-entry and descent to the surface of the Earth.

We consider several factors to maintain the physical, mental, and emotional well- being of the married astronauts. Adequate space and privacy are provided within the capsule, and a robust communications system is designed to minimize the chance of crew isolation from Earth. We propose a regenerative oxygen and carbon dioxide recycling system based upon the International Space Station with a back-up set of compressed storage tanks for air circulation. This hybrid approach reduces system mass whilst ensuring a continued supply of fresh, breathable air. While most food for the crew will be pre-prepared and packaged, a limited supply of fresh vegetables and seasonings can be grown on-board; in addition to providing variety to the crew diet, the growing and nurturing of the plants will give needed mental stimulation and a psychological connection to Earth for the married couple. Incidental tools and equipment for the crew can be manufactured on-board via the use of 3-D printing, a technology that has been demonstrated in micro-gravity environments upon the ISS. Simulations of the ECLSS are performed using SICLE, the Simulator for Closed Life and Ecology. These novel human factors technologies, along with the proposed launch and Earth return scenarios, are key components that enable deep space human missions, for example the proposed crewed flyby of Mars in 2018, in both the near- and far-term.

10 Kanau Team Workflow, Website and Animation Video

Table 23 Major tasks performed by individual team members.

Team Member	Major tasks
Shota Iino	Project management, mission design, payload mission, Kanau spacecraft's interior design, environment control and life support system, and facilities for crew physical health and radiation protection.
Kshitij Mall	Project management, mission design, launch vehicle selection, concept of operations, aerocapture, command and data handling, communications, safety analysis, crew selection, website development and logo design.
Ayako Ono	Kanau spacecraft's interior design and facilities for radiation protection.

Jeff Stuart	Trajectory design and launch vehicle selection, concept of operations, and aerocapture.
Ashwati Das	Trajectory design and launch vehicle selection, concept of operations, and aerocapture.
Eriko Moriyama	Environment control and life support system.
Takuya Ohgi	Environment control and life support system.
Nick Gillin	Mission animation videos.
Team Member	**Major tasks**
Koki Tanaka	Kanau spacecraft's interior design.
Yuri Aida	Facilities for radiation protection and crew physical health.
Max Fagin	Mission design, power systems, thermal control system, and concept of operations.
Daichi Nakajima	Launch vehicle selection.

Table 24 Major roles of team's advisors.

Advisor	**Major roles**
Dr. H. Miyajima	Mission design, and environment control and life support system.
Dr. M. J. Grant	Selection of launch vehicle, command and data handling, communications, and aerocapture.

Team Website Address: https://sites.google.com/site/occupyplanet4/
Kanau Mission Animation Video: Development of our mission video is ongoing.

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Assistant Professor Kazuhiro Terasawa (Keio University, Japan) – Radiation analysis
Mr. Rizwan Qureshi (NASA Goddard Space Flight Center, USA) – Aerocapture analysis
Mr. Yoshiki Annou (Japanese Mars Society)
Mr. Kiosuke Murakawa (Japanese Mars Society)
Jim D'Entrement (Graduate Student, Purdue University, USA) - Propulsion
Marat Kulakhmetov (Graduate Student, Purdue University, USA) - Propulsion

Dr. Stephen D. Heister (Professor, Purdue University, USA) - Propulsion
Mr. Isaac Cooper (Machinimist, New Zealand) – Animation video

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Appendix A

Table 1 Comparison between aerobraking and aerocapture techniques.

Aerobraking	Aerocapture
1. Performed for around 6 months.	1. Performed for a few minutes.
2. Requires lighter and thinner thermal protection system.	2. Requires heavier and thicker thermal protection system.
3. Demonstrated four times by NASA.[1]	3. Never demonstrated before.
4. Suitable for robotic/cargo missions.	4. Suitable for human space missions.

	Mass (kg)	Volume (m^3)	Power (W)	Cooling (W)	ESM (kg)
Water Management					
Urine / Waste Water Collection System	4.55	0.02	4.00	4.00	10.05
Water Treatment Process	2463.74	4.31	919.74	919.74	3665.71
Urine, Hygiene & Potable Water, & Brine Storage Tankage	181.57	0.47	17.80	17.80	288.14
Process Controller	36.11	0.08	156.18	156.18	99.74
Water Quality Monitoring	14.07	0.04	4.72	4.72	24.09
Product Water Delivery System	51.73	0.12	3.44	3.44	78.61
Potable Water Storage	595.54	0.44	20.74	20.74	696.52
Air Management					
Oxygen Generation	378.86	1.00	3288.88	1801.94	1481.94
Gaseous Trace Contaminant Control	85.81	0.40	194.35	194.35	229.73
Carbon Dioxide Removal	179.14	0.42	536.06	536.06	428.86
Carbon Dioxide Reduction	143.53	0.19	148.59	148.59	228.61
Atmosphere Composition Monitoring Assembly	54.30	0.09	103.50	103.50	104.43
Common Cabin Air Assembly	118.08	0.50	530.52	530.52	383.39
Atmosphere Circulation	9.80	0.02	61.00	61.00	32.23
Resupply Water for Electrolysis	471.04	0.47	0.00	0.00	572.55
Waste Management					
Solid Waste Treatment (Tankage)	345.60	9.60	0.00	0.00	2414.40
Total	5133.47	18.17	5989.52	4502.58	10739.01

Table 2 ESM breakdown for bio-regenerative systems.

Table 3 ESM breakdown for non-regenerative systems.

	Technology	Mass (kg)	Volume (m^3)	Power (W)	Cooling (W)	ESM (kg)
Water Management						
Urine / Waste Water Collection System	ISS	4.55	0.02	4.00	4.00	10.05
Fresh Water	Storage	22680.27	24.95	0.00	0.00	28056.63
Waste Water Tankage	Storage	50.00	1.00	0.00	0.00	265.50
Air Management						
Oxygen High Pressure Tank at 20.7 MPa	Storage	1141.22	6.82	0.00	0.00	2611.05
Gaseous Trace Contaminant Control	ISS	85.81	0.40	194.35	194.35	229.73
Carbon Dioxide Removal Canister	LiOH	1454.40	3.25	0.00	0.00	2154.56
Atmosphere Composition Monitoring Assembly	ISS	54.30	0.09	103.50	103.50	104.43
Common Cabin Air Assembly	ISS	118.08	0.50	530.52	530.52	383.39
Atmosphere Circulation	ISS	9.80	0.02	61.00	61.00	32.23
Waste Management						
Solid Waste Treatment (Tankage)	Storage	345.60	9.60	0.00	0.00	2414.40
Total		25944.03	46.65	893.37	893.37	36261.97

Table 4 Hybrid life support system.

	Q	Total Mass (kg)
Water Management		
Recycling		
Urine / Waste Water Collection System	2	9.10
Water Treatment Process	2	4927.48
Urine, Hygiene & Potable Water, & Brine Storage Tankage	2	363.14
Process Controller	2	72.22
Water Quality Monitoring	2	28.14
Product Water Delivery System	2	103.46
Potable Water Storage	2	1191.08
Storage		
Fresh Water	250	11317.50
Waste Water Tankage	1	20.00
Air Management		
Recycling		
Oxygen Generation	2	757.72
Gaseous Trace Contaminant Control	2	171.62
Carbon Dioxide Removal	2	358.28
Carbon Dioxide Reduction	2	287.06
Atmosphere Composition Monitoring Assembly	2	108.60
Common Cabin Air Assembly	2	236.16
Atmosphere Circulation	2	19.60

		Water Supply for OGA	501	818.70
	Storage			
		Oxygen Tankage (High Pressure)	250	417.50
		Carbon Dioxide Absorber (LiOH canister)	250	725.75
Waste Management				
		Solid Waste Treatment (Tankage)	1	345.60
Food				
	Storage			
		Food	501	661.32
		Food Packaging	501	99.20
Total				23039.22

Table 5 Ground station characteristics for 70 m antenna.

Transmission		Reception	
Frequency band	**EIRP**	**Frequency band**	**Effective G/T, 10o**
7145 – 7190 MHz	89.31 dBW (1995W RF)	8400 - 8450 MHz	42.52 dB/K
34200 –34700 MHz	114.69 dBW (794W RF)	31800 – 32300 MHz	56.71 dB/K

Table 6 Links description.

Link	Ka-band		X-band		UHF		Laser	X-band	
	Uplink	**Downlink**	**Uplink**	**Downlink**	**Uplink**	**Downlink**		**Downlink**	**Uplink**
Frequency	34.5 GHz	32 GHz	7.15 GHz	8.42 GHz	437.1 MHz	401.6 MHz	Wavelength =1064 nm	7.2 GHz	8.45 GHz
Tx power	794 W	65 W	19953 W	65 W	5W	5W	5W	65 W	65 W
Modulation	NRZ/PSK /PM	GMSK. BTb=0.5	NRZ/PSK/PM	GMSK BTb=0.5	PCM-NRZ/BPSK	PCM-NRZ/BPSK	256-PPM	QPSK	QPSK
Coding	Turbo Coding 1/4	Concatenated: Convolutional + RS (255, 223)	Turbo Coding 1/4	Concatenated: Convolutional + RS (255, 223)	Convolutional, rate 1/2	Convolutional, rate 1/2	Reed Solomon (26143, 15685)	Concatenated, Interleaving=5	Concatenated, Interleaving = 5
BER	Negligible	Negligible	BER=10-6	BER=10-6	10-6	10-6	BER=10-6	FER=10-5	FER=10-5
Bit rate (worst case)	1.76 Mbps	1.5 Mbps	22.6 kbps	460 bps	128 kbps	128 kbps	10 Mbps		20 Mbps

Appendix B

HOW	Mission Cost	Mission Safety	Mission Complexity	Technology Readiness Level	Mission Duration	Number of Launches	Re-entry Velocity	Radiation Exposure	Throw Mass to Mars	Psychological Stress
Mission Cost										
Mission Safety										
Mission Complexity	+	+								
Technology Readiness Level	+	+								
Mission Duration	-	+		-						
Number of Launches	++		+	-						
Re-entry Velocity		+	-	-						
Radiation Exposure		++		-	++					
Throw Mass Capacity			-	++		--				
Psychological Stress				-	++					

++	Strong Positive Relationship
+	Positive Relationship
--	Strong Negative Relationship
-	Negative Relationship

Kanau Inspiration Mars Mission Analysis

WHAT				HOW: Mission Cost	Mission Safety	Mission Complexity	Technology Readiness Level	Mission Duration	Number of Launches	Re-entry Velocity	Radiation Exposure	Throw Mass to Mars	Psychological Stress	BENCHMARKS: Inspiration Mars
Units				billion $	probability	#	year	days	#	m/s	sieverts	mT	suds	
Relative Importance				↓	↑	↓	↑	↓	↓	↓	↓	↑	↓	↓
Mission Segments	Affordable	10	7.58	9			6	3	6			6		10
	Timely Launches	10	7.58			9	9		9			6		10
Performance	Adequate Life Support Elements	10	7.58	9	9			6			9			9
	Radiation Minimizing Elements	8	6.06	9	9		6							7
	Required Thrust	10	7.58	9			6		9			9		10
	Low Re-entry Velocity	9	6.82		6		6			9				9
	Easy to Assemble Vehicle Stack in Space	7	5.30			9			9					7
Maintenence	Repairable	9	6.82	3	6	3								7
	Reliable	7	5.30	9	9	6			3	3	3			6
Communication	Re-establish Link	9	6.82	3	6									6
	Maintain Link	9	6.82	3	6									7
Additional Constraints	Political Support	10	7.58	9	6	9	9	3						10
	Re-entry Heat Rate Limit	9	6.82		9		6			9				9
	Vehicle Mass Limit	8	6.06	3				3	6			9		8
	Stress-Free Environment	7	5.30				6	9					9	6
	Absolute			600	582	312	498	207	372	183	111	282	63	
	Relative			0.1869	0.1813	0.0972	0.1551	0.0645	0.1159	0.0570	0.0346	0.0879	0.0196	
	Targets			1.5	0.99	3	2015	501	3	12.8	0.65	15	3	
	Threshold			2	0.98	4	2017	600	4	14.18	1	10	7	
	Rank			1	2	5	3	7	4	8	9	6	10	

Relationship Attributes	
3	Weak
6	Moderate
9	Strong

Figure 1 HoQ.

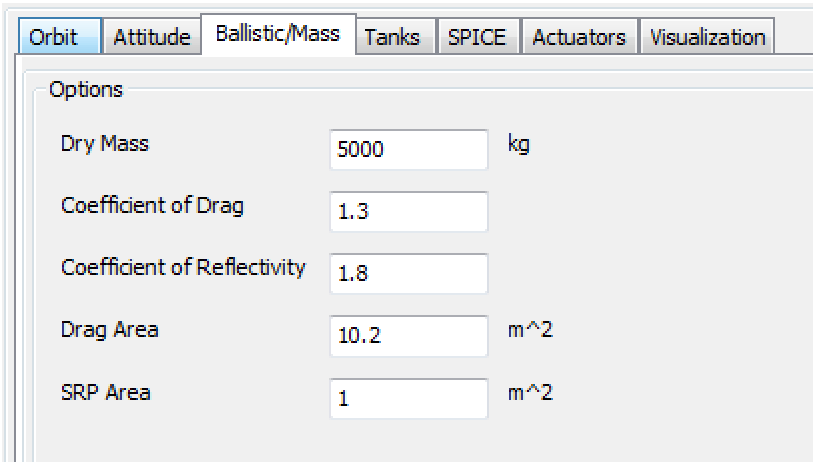

Figure 2 Ballistic properties of Dragon used for analysis in GMAT.

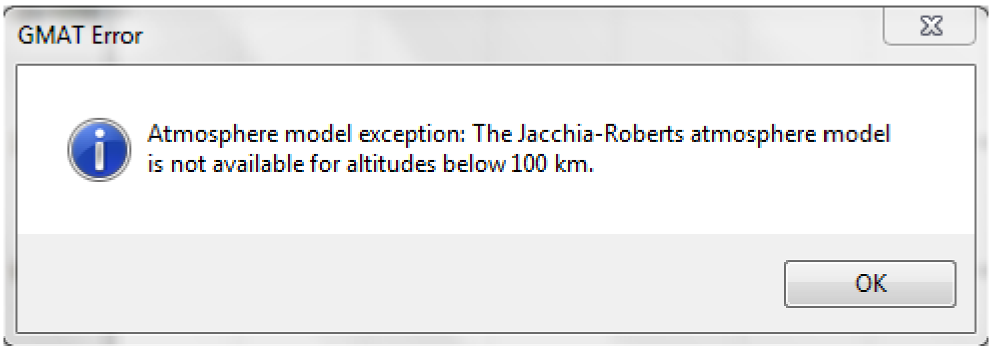

Figure 3 Error using Jacchia-Roberts atmosphere model.

Kanau_reentrypod.Earth.Altitude	Kanau_reentrypod.Earth.VelPeriapsis	Kanau_reentrypod.A1Gregorian
56.36377357027595	11.10294077700751	20 May 2019 20:59:35.034
56.36378456807051	11.10675489259942	20 May 2019 20:59:34.986
56.3638129012943	11.11015410737838	20 May 2019 20:59:34.944
56.36385342487483	11.11321956328929	20 May 2019 20:59:34.905
56.36390366154046	11.11606088604598	20 May 2019 20:59:34.869
56.36396466272345	11.11884293016761	20 May 2019 20:59:34.834
56.36403891543068	11.12168039415599	20 May 2019 20:59:34.799
56.36414130500452	11.12500286778107	20 May 2019 20:59:34.757
56.36425556715312	11.12820026282145	20 May 2019 20:59:34.717
56.36439039765355	11.13151719901826	20 May 2019 20:59:34.675
56.36449462677319	11.13383854155983	20 May 2019 20:59:34.646
56.36459725427994	11.13596538006809	20 May 2019 20:59:34.619
56.36469648819184	11.13789923511816	20 May 2019 20:59:34.595
56.3647936005118	11.13969351029988	20 May 2019 20:59:34.573
56.36488720138277	11.14134377415845	20 May 2019 20:59:34.552
56.36497814387258	11.14288202266224	20 May 2019 20:59:34.533
56.36510539354094	11.14494008943346	20 May 2019 20:59:34.507
56.36523298943575	11.14690730185679	20 May 2019 20:59:34.482
56.36535653975989	11.14873182450689	20 May 2019 20:59:34.460
56.36545657685838	11.15015759052447	20 May 2019 20:59:34.442
56.36555686140673	11.15154500507169	20 May 2019 20:59:34.424
56.36566343978393	11.15297742835266	20 May 2019 20:59:34.407
56.36576414997944	11.1542943559987	20 May 2019 20:59:34.390
56.36586187136254	11.15554075891447	20 May 2019 20:59:34.375
56.36596572986946	11.15683391774899	20 May 2019 20:59:34.358
56.36607993748567	11.15822109964667	20 May 2019 20:59:34.341
56.3661943707657	11.1595770316132	20 May 2019 20:59:34.324
56.36635199159628	11.16139333682492	20 May 2019 20:59:34.301
56.36648106119264	11.16283973219195	20 May 2019 20:59:34.283
56.3666031415014	11.16417644708688	20 May 2019 20:59:34.267
56.36671633951846	11.16539040652235	20 May 2019 20:59:34.252
56.36683549064219	11.16664328225297	20 May 2019 20:59:34.236
56.36694710418124	11.16779500268474	20 May 2019 20:59:34.222
56.36707792574998	11.16911948189121	20 May 2019 20:59:34.205
56.36721266426775	11.17045652989659	20 May 2019 20:59:34.189
56.36734603549576	11.17175449508569	20 May 2019 20:59:34.172
56.36747412386376	11.17297846528137	20 May 2019 20:59:34.157
56.36759844703192	11.17414639166177	20 May 2019 20:59:34.143
56.36772455404116	11.17531187410207	20 May 2019 20:59:34.128
56.36788914750196	11.17680546287589	20 May 2019 20:59:34.110
56.36805471350544	11.17827808813746	20 May 2019 20:59:34.091
56.36820842237466	11.17962000672286	20 May 2019 20:59:34.075
56.36836158431379	11.18093425186049	20 May 2019 20:59:34.058
56.36850868296097	11.18217600752419	20 May 2019 20:59:34.043
56.36866387912869	11.18346542175583	20 May 2019 20:59:34.027
56.36880951990679	11.18465699548755	20 May 2019 20:59:34.012
56.36896996665109	11.18594996361366	20 May 2019 20:59:33.996
56.36912216137716	11.18715814105881	20 May 2019 20:59:33.981
56.36923621955339	11.18805236702231	20 May 2019 20:59:33.970
56.36934143090639	11.18886900644419	20 May 2019 20:59:33.960
56.36945183972148	11.18971774842084	20 May 2019 20:59:33.949
56.36956106036632	11.19054929049654	20 May 2019 20:59:33.939
56.36970046892475	11.19159936930785	20 May 2019 20:59:33.926
56.36985941849798	11.19278170932703	20 May 2019 20:59:33.911
56.3700197854987	11.19395907611858	20 May 2019 20:59:33.896
56.37019006433457	11.19519281428254	20 May 2019 20:59:33.881
56.37035269210992	11.19635595264979	20 May 2019 20:59:33.867

Figure 4 Sample results for aerocapture at an altitude of 56.3 km.

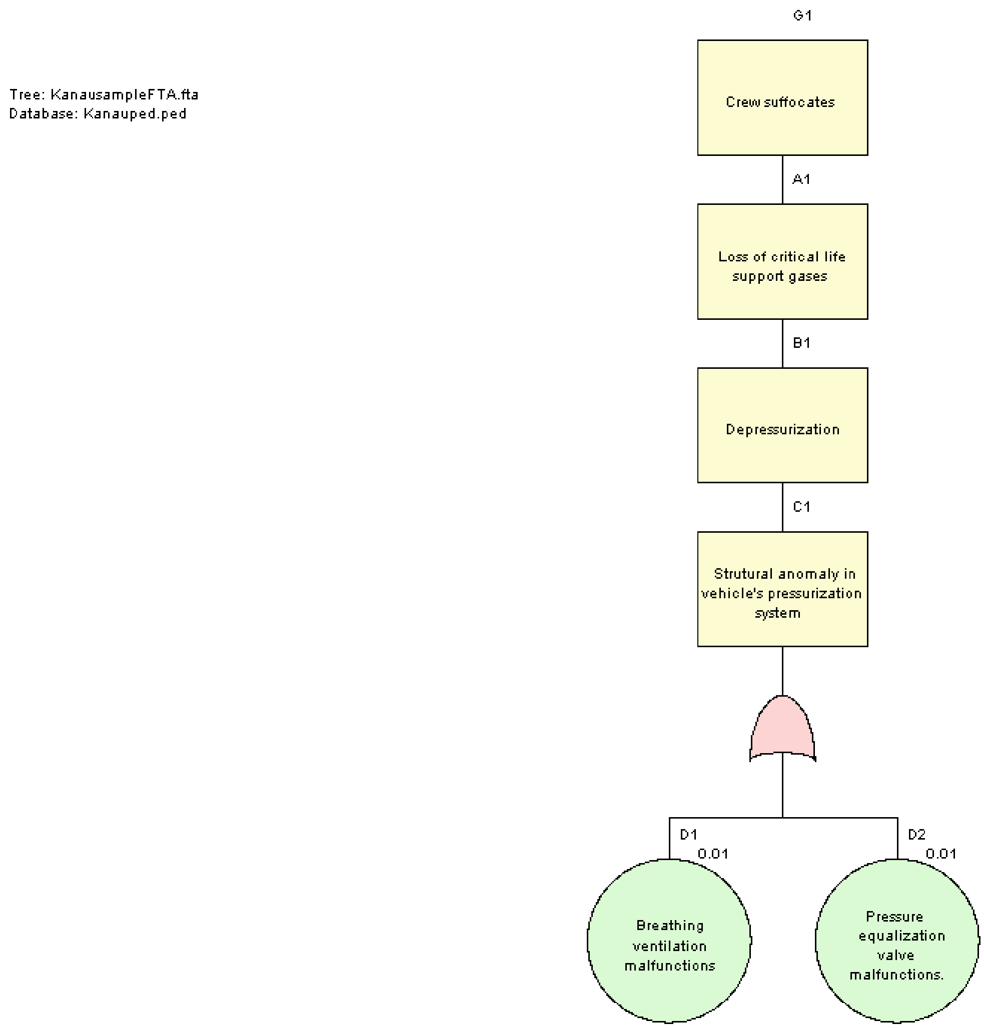

Figure 5 Sample fault tree.

```
Monte Carlo Simulation
======================

Tree   : KanausampleFTA.fta
Time   : Sun Mar 09 19:25:49 2014

Note: Only runs with at least one component failure are simulated

Number of primary events  = 2
Number of tests           = 10000
Unit Time span used       = 1.000000

Number of system failures = 10000

Probability of at least   = 1.990000E-002  ( exact )
one component failure

Probability of top event  = 1.990000E-002  ( +/- 1.990000E-004 )

Rank   Failure mode         Failures  Estimated Probability        Importance

  1    D2                   5017      9.983830E-003 ( +/- 1.409533E-004 )  50.17%
  2    D1                   4941      9.832590E-003 ( +/- 1.398816E-004 )  49.41%
  3    D1 D2                42        8.358000E-005 ( +/- 1.289667E-005 )   0.42%


Compressed:

Rank   Failure mode         Failures  Estimated Probability    Importance

  1    D1                   4983      9.916170E-003 ( +/- 1.404748E-004 )  49.83%
  2    D2                   5059      1.006741E-002 ( +/- 1.415420E-004 )  50.59%


Primary Event Analysis:

 Event          Failure contrib.    Importance

 D1             9.916170E-003           49.83%
 D2             1.006741E-002           50.59%
```

Figure 6 Sample Monte Carlo analysis.

Appendix C

```
%General Mission Analysis Tool(GMAT) Script
%Created: 2014-03-03 02:50:18
%----------------------------------------
%---------- Spacecraft
%----------------------------------------
Create Spacecraft Kanau_reentrypod;
GMAT Kanau_reentrypod.DateFormat = UTCGregorian;
GMAT Kanau_reentrypod.Epoch = '20 May 2019 20:59:00.000';
GMAT Kanau_reentrypod.CoordinateSystem = EarthMJ2000Eq;
GMAT Kanau_reentrypod.DisplayStateType = Keplerian;
GMAT Kanau_reentrypod.SMA = 643449.9999999778;
GMAT Kanau_reentrypod.ECC = 0.9899999999999994;
GMAT Kanau_reentrypod.INC = 0;
GMAT Kanau_reentrypod.RAAN = 0;
GMAT Kanau_reentrypod.AOP = 0;
GMAT Kanau_reentrypod.TA = 0;
GMAT Kanau_reentrypod.DryMass = 5000;
GMAT Kanau_reentrypod.Cd = 1.3;
GMAT Kanau_reentrypod.Cr = 1.8;
GMAT Kanau_reentrypod.DragArea = 10.2;
GMAT Kanau_reentrypod.SRPArea = 1;
GMAT Kanau_reentrypod.NAIFId = -123456789;
GMAT Kanau_reentrypod.NAIFIdReferenceFrame = -123456789;
GMAT Kanau_reentrypod.Id = 'SatId';
GMAT Kanau_reentrypod.Attitude = CoordinateSystemFixed;
GMAT Kanau_reentrypod.ModelFile = '../data/vehicle/models/aura.3ds';
GMAT Kanau_reentrypod.ModelOffsetX = 0;
GMAT Kanau_reentrypod.ModelOffsetY = 0;
GMAT Kanau_reentrypod.ModelOffsetZ = 0;
GMAT Kanau_reentrypod.ModelRotationX = 0;
GMAT Kanau_reentrypod.ModelRotationY = 0;
GMAT Kanau_reentrypod.ModelRotationZ = 0;
GMAT Kanau_reentrypod.ModelScale = 3;
GMAT Kanau_reentrypod.AttitudeDisplayStateType = 'Quaternion';
GMAT Kanau_reentrypod.AttitudeRateDisplayStateType = 'AngularVelocity';
GMAT Kanau_reentrypod.AttitudeCoordinateSystem = EarthMJ2000Eq;
GMAT Kanau_reentrypod.EulerAngleSequence = '321';
%----------------------------------------
%---------- ForceModels
%----------------------------------------
Create ForceModel Kanau_reentry_prop_ForceModel;
GMAT Kanau_reentry_prop_ForceModel.CentralBody = Earth;
GMAT Kanau_reentry_prop_ForceModel.PrimaryBodies = {Earth};
GMAT Kanau_reentry_prop_ForceModel.PointMasses = {Luna, Sun};

GMAT Kanau_reentry_prop_ForceModel.SRP = Off;
GMAT Kanau_reentry_prop_ForceModel.RelativisticCorrection = Off;
GMAT Kanau_reentry_prop_ForceModel.ErrorControl = RSSStep;
GMAT Kanau_reentry_prop_ForceModel.GravityField.Earth.Degree = 10;
GMAT Kanau_reentry_prop_ForceModel.GravityField.Earth.Order = 10;
GMAT Kanau_reentry_prop_ForceModel.GravityField.Earth.PotentialFile = 'JGM2.cof';
GMAT Kanau_reentry_prop_ForceModel.GravityField.Earth.EarthTideModel = 'None';
GMAT Kanau_reentry_prop_ForceModel.Drag.AtmosphereModel = MSISE90;
GMAT Kanau_reentry_prop_ForceModel.Drag.F107 = 150;
```

```
GMAT Kanau_reentry_prop_ForceModel.Drag.F107A = 150;
GMAT Kanau_reentry_prop_ForceModel.Drag.MagneticIndex = 3;
%----------------------------------------
%---------- Propagators
%----------------------------------------
Create Propagator Kanau_reentry_prop;
GMAT Kanau_reentry_prop.FM = Kanau_reentry_prop_ForceModel;
GMAT Kanau_reentry_prop.Type = RungeKutta89;
GMAT Kanau_reentry_prop.InitialStepSize = 60;
GMAT Kanau_reentry_prop.Accuracy = 1e-009;
GMAT Kanau_reentry_prop.MinStep = 0.001;
GMAT Kanau_reentry_prop.MaxStep = 2700;
GMAT Kanau_reentry_prop.MaxStepAttempts = 50;
GMAT Kanau_reentry_prop.StopIfAccuracyIsViolated = true;
%----------------------------------------
%---------- Burns
%----------------------------------------
Create ImpulsiveBurn DefaultIB;
GMAT DefaultIB.CoordinateSystem = Local;
GMAT DefaultIB.Origin = Earth;
GMAT DefaultIB.Axes = VNB;
GMAT DefaultIB.Element1 = 0;
GMAT DefaultIB.Element2 = 0;
GMAT DefaultIB.Element3 = 0;
GMAT DefaultIB.DecrementMass = false;
GMAT DefaultIB.Isp = 300;
GMAT DefaultIB.GravitationalAccel = 9.810000000000001;
%----------------------------------------
%---------- Subscribers
%----------------------------------------
Create OrbitView DefaultOrbitView;
GMAT DefaultOrbitView.SolverIterations = Current;
GMAT DefaultOrbitView.UpperLeft = [ 0.001682085786375105 48.73905996758509 ];
GMAT DefaultOrbitView.Size = [ 0.9747687132043734 0.7698541329011345 ];
GMAT DefaultOrbitView.RelativeZOrder = 82;
GMAT DefaultOrbitView.Maximized = true;
GMAT DefaultOrbitView.Add = {Kanau_reentrypod, Earth};
GMAT DefaultOrbitView.CoordinateSystem = EarthMJ2000Eq;
GMAT DefaultOrbitView.DrawObject = [ true true ];
GMAT DefaultOrbitView.OrbitColor = [ 255 32768 ];
GMAT DefaultOrbitView.TargetColor = [ 8421440 0 ];
GMAT DefaultOrbitView.DataCollectFrequency = 1;
GMAT DefaultOrbitView.UpdatePlotFrequency = 100;
GMAT DefaultOrbitView.NumPointsToRedraw = 0;
GMAT DefaultOrbitView.ShowPlot = true;
GMAT DefaultOrbitView.ViewPointReference = Earth;
GMAT DefaultOrbitView.ViewPointVector = [ -60000 30000 20000 ];
GMAT DefaultOrbitView.ViewDirection = Earth;
GMAT DefaultOrbitView.ViewScaleFactor = 1;
GMAT DefaultOrbitView.ViewUpCoordinateSystem = EarthMJ2000Eq;
GMAT DefaultOrbitView.ViewUpAxis = Z;
GMAT DefaultOrbitView.EclipticPlane = Off;
GMAT DefaultOrbitView.XYPlane = Off;
GMAT DefaultOrbitView.WireFrame = Off;
GMAT DefaultOrbitView.Axes = On;
GMAT DefaultOrbitView.Grid = Off;
```

```
GMAT DefaultOrbitView.SunLine = Off;
GMAT DefaultOrbitView.UseInitialView = On;
GMAT DefaultOrbitView.StarCount = 7000;
GMAT DefaultOrbitView.EnableStars = On;
GMAT DefaultOrbitView.EnableConstellations = Off;
Create GroundTrackPlot DefaultGroundTrackPlot;
GMAT DefaultGroundTrackPlot.SolverIterations = Current;
GMAT DefaultGroundTrackPlot.UpperLeft = [ 0.2926829268292683 48.87196110210697 ];
GMAT DefaultGroundTrackPlot.Size = [ 0.479394449116905 0.3598055105348461 ];
GMAT DefaultGroundTrackPlot.RelativeZOrder = 78;
GMAT DefaultGroundTrackPlot.Maximized = true;
GMAT DefaultGroundTrackPlot.Add = {Kanau_reentrypod, Earth};
GMAT DefaultGroundTrackPlot.DataCollectFrequency = 1;
GMAT DefaultGroundTrackPlot.UpdatePlotFrequency = 50;
GMAT DefaultGroundTrackPlot.NumPointsToRedraw = 0;
GMAT DefaultGroundTrackPlot.ShowPlot = true;
GMAT DefaultGroundTrackPlot.CentralBody = Earth;
GMAT DefaultGroundTrackPlot.TextureMap = '../data/graphics/texture/ModifiedBlueMarble.jpg';
Create ReportFile ReportFile1;
GMAT ReportFile1.SolverIterations = Current;
GMAT ReportFile1.UpperLeft = [ 0 54.57914338919925 ];
GMAT ReportFile1.Size = [ 1.1239092495637 0.9683426443202979 ];
GMAT ReportFile1.RelativeZOrder = 837;
GMAT ReportFile1.Maximized = false;
GMAT ReportFile1.Filename = 'ReportFile.txt';
GMAT ReportFile1.Precision = 16;
GMAT ReportFile1.Add = {Kanau_reentrypod.Earth.Altitude, Kanau_reentrypod.Earth.VelPeriapsis,
    Kanau_reentrypod.A1Gregorian};
GMAT ReportFile1.WriteHeaders = true;
GMAT ReportFile1.LeftJustify = On;
GMAT ReportFile1.ZeroFill = Off;
GMAT ReportFile1.ColumnWidth = 20;
GMAT ReportFile1.WriteReport = true;
Create XYPlot XYPlot1;
GMAT XYPlot1.SolverIterations = Current;
GMAT XYPlot1.UpperLeft = [ 0.2952060555088309 0.06969205834683954 ];
GMAT XYPlot1.Size = [ 0.4945332211942809 0.7990275526742301 ];
GMAT XYPlot1.RelativeZOrder = 90;
GMAT XYPlot1.Maximized = true;
GMAT XYPlot1.XVariable = Kanau_reentrypod.Earth.Altitude;
GMAT XYPlot1.YVariables = {Kanau_reentrypod.Earth.VelPeriapsis};
GMAT XYPlot1.ShowGrid = true;
GMAT XYPlot1.ShowPlot = true;
%----------------------------------------
%---------- Mission Sequence
%----------------------------------------
BeginMissionSequence;
Propagate BackProp Synchronized Kanau_reentry_prop(Kanau_reentrypod) {Kanau_reentrypod.ElapsedSecs = -3600,
    StopTolerance = 0.0001};
```