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

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Abstract—In 1998 Patel et al searched for Earth-Mars free-return trajectories that leave Earth, fly by Mars, and return to Earth without any deterministic maneuvers after Trans-Mars Injection. They found fast trajectory opportunities occurring two times every 15 years with a 1.4-year duration, significantly less than most Mars free return trajectories, which take up to 3.5 years. This paper investigates these fast trajectories. It also determines the launch and life support feasibility of flying such a mission using hardware expected to be available in time for an optimized fast trajectory opportunity in January, 2018.

The authors optimized the original trajectory using patched-conic approximations, and then modeled the trajectory using numerical integration with high fidelity force models and the JPL planetary ephemerides. We calculated an optimum trajectory launching in early January, 2018. At the Mars encounter, the spacecraft will pass within a few hundred kilometers of the surface. We investigated the Earth reentry conditions and developed some aerocapture options to mitigate G-loads on the returning crew. We also describe tradeoffs and studies necessary to develop the Thermal Protection System (TPS).

To size the Environmental Control and Life Support System (ECLSS) we set the initial mission assumption to two crew members for 500 days in a modified SpaceX Dragon class of vehicle. The journey is treated as a high-risk mission, which drives towards reliable—but minimalist—accommodations and provisions. As such, we investigated State Of the Art (SOA) technologies that would meet only basic human needs to support metabolic requirements and limited crew comfort allowances.

We compare a baseline SOA architecture with an advanced architecture. The advanced architecture uses recently developed equipment that has higher efficiencies for water recovery and lighter base mass. They are not currently in operation and therefore present a schedule risk for development and testing.

We also present a notional schedule based on state of the art ECLSS technologies. ECLSS is a systems-integration-intense subsystem, so actual schedule is highly dependent on the vehicle integration schedule and timeline.

The isolated, confined environment psychology aspects of the mission are considered with regard to crew selection, training,

capsule design, the role of mission control / support, and early ground testing. We explore analogues such as Biosphere 2 and long duration spaceflight.

We show that an ECLSS based on SOA technologies is feasible and can be ready for January 2018. A minimalist approach using existing technologies can be safely and robustly realized by utilizing spares and a crew capable of servicing and replacing the equipment.

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I. INTRODUCTION

Government and private organizations are planning manned missions to Mars, including those that orbit Mars and those that land on the Martian surface. To reduce risk these missions are often designed using a "free-return" trajectory. These trajectories leave Earth, perform a close flyby of Mars, and return to Earth without any deterministic maneuvers after the Trans-Mars Injection (TMI) maneuver. As part of preparing for a manned Mars mission, the ability to perform a Mars free-return mission will be necessary. A free return mission itself demonstrates a long duration ECLSS capability, long term human flight operations, and the ability to reenter the Earth's atmosphere at higher velocities than previous missions required. The technologies developed for and demonstrated by a Mars free-return mission are all necessary for Mars orbit and landing trajectories. Therefore, a manned Mars free-return mission is a reasonable precursor to any manned Mars orbital or landing mission. The engineering, organization, and human factors disciplines will learn much from such a mission, and this knowledge will be applicable and necessary for any subsequent manned Mars mission.

Patel, Longuski, and Sims[1] wrote software to automatically search for Earth-Mars free-return trajectories. In their first figure, reproduced here as Fig.1, they found some fast trajectory opportunities that occur approximately two times every 15 years. The Time-of-Flight (TOF) in years is shown along the left, and the launch dates are shown along the bottom. The individual data points on the graph use numbers as symbols to describe the characteristics of the trajectory. On the bottom right of the figure there are a set of trajectories that have a total time-of-flight (TOF) roughly 1.4 years, significantly less than most Mars free return trajectories which can take more than 1.8 years. (Fast opportunities circled in red.) According to their figure, many opportunities occur with a two-year TOF, but most have a TOF from 2.5 to 3.5 years.

These fast free-return trajectories are interesting because they may be viable missions for human spaceflight using current and near-term technology. In fact, Patel et al conclude their work with the sentence "These trajectories may provide a timely opportunity for the first human mission to Mars."

Patel et al used software that uses patched conic approximations, which is a force model that only includes the gravitational effect of one celestial body at a time. Although this is a good approximation for heliocentric trajectories, it is necessary to increase the fidelity of the model to determine the feasibility of a candidate mission design. This report investigates these 1.4 year trajectory opportunities using full numerical integration of multi-body force models, and looks at the feasibility of flying such a mission using hardware that is expected to be available in time for an opportunity in January, 2018.

II. TRAJECTORY DESIGN

To demonstrate the feasibility of these 1.4-year trajectories, we modeled the trajectory by numerical integration with a high fidelity force model using the JPL high-precision planetary ephemerides DE 421 [2]. To do this we used the STK/Astrogator software[3], made by Analytical Graphics, Inc. Using the dates from Patel et al as a first estimate, Astrogator's numerical differential corrector first targeted the B-Plane at Mars, and then the B-Plane at Earth return. After manually adjusting the Earth launch and Mars flyby dates, we were able to create a viable trajectory. This trajectory was not optimized in terms of the Earth departure energy, "C3."

To calculate a trajectory with an optimum C3, we recreated the trajectory using the Mission Analysis Environment (MAnE) software[4] [5] [6] from Space Flight Solutions. This uses a patched conic approximations and enables the operator to choose optimization goals and constraints. MAnE calculated an optimum trajectory using the first estimated dates from the Astrogator runs. This optimum solution is shown in TABLE I and TABLE II.

We then used this patched-conic solution as initial conditions in Astrogator, and targeted using the full force model. The results are shown graphically in Fig.2, and the numerical values are given in TABLE III and TABLE IV. In terms of V-Infinity, these match very well with the patched-conic results from MAnE. After launch, the trajectory (the solid bold line) passes inside the Earth's orbit, then performs the flyby at Mars. This flyby is a gravity assist that causes the return trajectory (dashed line) to have a perihelion close to Venus' orbit, although Venus itself is not close at that time. After perihelion the trajectory continues back to Earth and performs reentry.

The flyby of Mars is shown in Fig.3. The periapsis point occurs on the dark side of Mars at approximately 100 km altitude, although there is some flexibility in this altitude. The figure shows the terminator, and the direction towards the Sun. The velocity at periapsis is about 7.27 km/sec. On this hyperbolic passage, the spacecraft will spend about ten hours within 100,000 km of Mars.

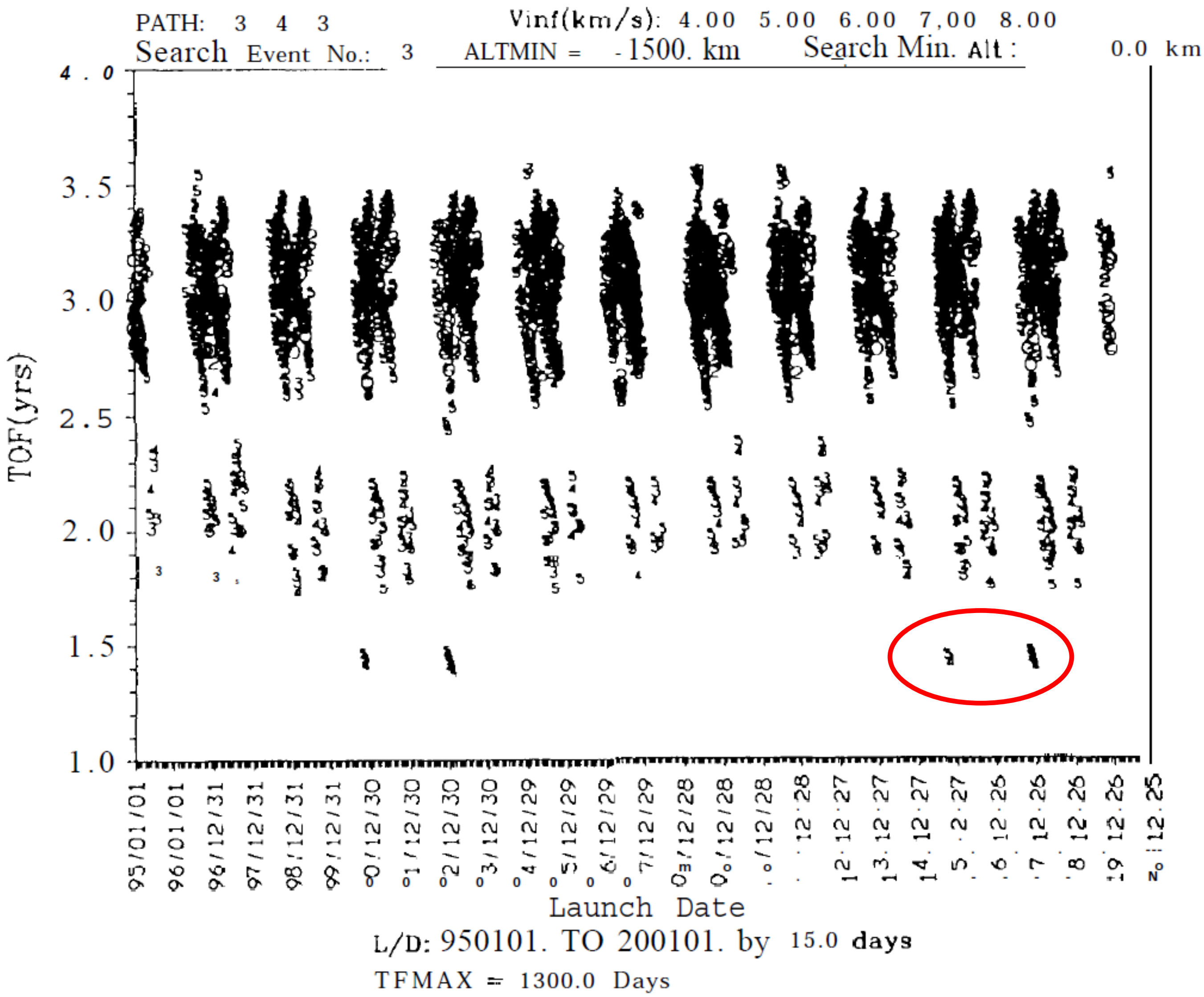

Fig.1. Mars Free-Return (1995–2020), Patel et al[1]. (The red circle highlights the near-term fast opportunities)

TABLE I. OPTIMUM EARTH-MARS FREE-RETURN SOLUTION DATES FROM MANE

Leg	Stay Time (days)	Depart		Arrive		Flight Time (days)
1		Earth	JAN 5, 2018, 7.1756 hours GMT Julian Date 58123.7990	Mars	AUG 20, 2018, 7.8289 hours GMT Julian Date 58350.8262	227.0272
2	0.0000	Mars	AUG 20, 2018, 7.8289 hours GMT Julian Date 58350.8262	Earth	MAY 21, 2019, 20.9618 hours GMT Julian Date 58625.3734	274.5472
					Total Duration	501.5744

TABLE II. OPTIMUM EARTH-MARS FREE-RETURN SOLUTION VALUES FROM MANE

Leg	V Inf *(km/s)*	Decl *(deg)*	Rt Asc *(deg)*	V Inf *(km/s)*	Decl *(deg)*	Rt Asc *(deg)*
1	6.22697	-6.48	181.24	5.42540	-7.48	233.71
2	5.42540	-11.85	200.36	8.91499	-25.40	142.87

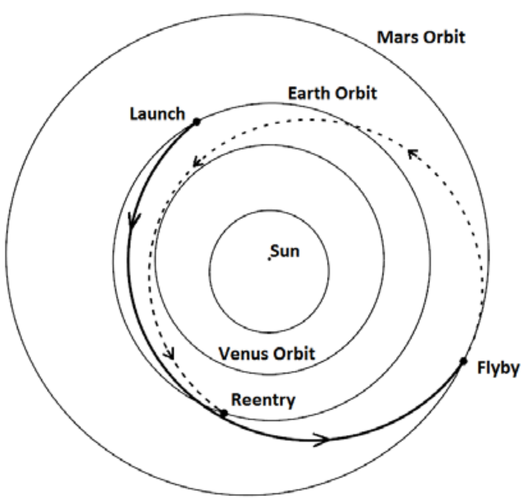

Fig.2. Numerically Integrated Earth-Mars Free-Return trajectory (Solid Out, Dashed Return)

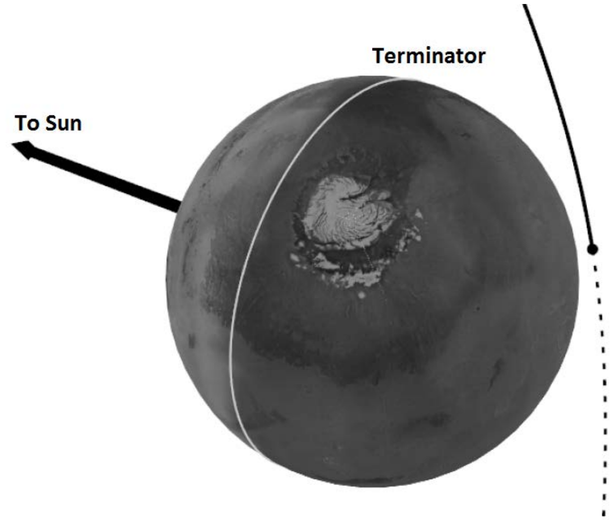

Fig.3. Mars Flyby

TABLE III. EARTH-MARS FREE-RETURN SOLUTION DATES FROM ASTROGATOR

Leg	Stay Time (days)	Depart		Arrive		Flight Time (days)
1		Earth	5 Jan 2018 07:00:00.000 UTCG	Mars	20 Aug 2018 08:18:19.619 UTCG	227.05439374
2	0.0000	Mars	20 Aug 2018 08:18:19.619 UTCG	Earth	21 May 2019 13:52:48.012 UTCG	274.23227306
					Total Duration	501.2866668

TABLE IV. EARTH-MARS FREE-RETURN SOLUTION VALUES FROM ASTROGATOR

	Departure					Arrival				
Leg	V Inf *(km/s)*	Decl *(deg)*	Rt Asc *(deg)*	V peri *(km/s)*	C3 *(km^2/s^2)*	V Inf *(km/s)*	Decl *(deg)*	Rt Asc *(deg)*	V peri *(km/s)*	C3 *(km^2/s^2)*
1	6.232	-6.554	271.053	12.578	38.835	5.417	-7.488	53.581	7.272	29.344
2	5.417	-11.94	200.12	7.272	29.344	8.837	-25.34	52.718	14.18	78.094

In summary, this trajectory requires leaving the Earth with a C3 of 38.835 km^2/sec^2 and returning with a velocity at perigee of 14.18 km/sec. The next questions are if a launch vehicle can throw sufficient mass to this C3, and if a capsule can be built with a heat shield to withstand an Earth re-entry velocity of 14.18 km/sec.

III. LAUNCH VEHICLE AND PAYLOAD VALUES

To determine if a launch vehicle can throw a space capsule to an Earth departure C3 of 38.835 km^2/sec^2 we need a value for the payload mass. In this study we use a value of 10,000 kg, which is based on an estimated SpaceX Dragon that has a dry mass of 4,200 kg and a cargo mass of 6,000 kg[7].

For a representative launch vehicle we chose to use the Falcon Heavy. According to a press release by SpaceX[8] in April, 2011, the first Falcon Heavy launch should be in 2013 or 2014. According to version 7 of the SpaceX brochure, the Falcon Heavy will be able to deliver (from Cape Canaveral) 29,610 kg to a 28.5 degree inclination 185 km circular Low Earth Orbit (LEO), and 15,010 kg to a 185 x 35,788 km Geostationary transfer orbit (GTO)[7]. Version 12 of this document[9] states the Falcon Heavy will be able to deliver 53,000 kg to a 28.5 degree inclination orbit (no altitude mentioned), and 19,000 kg to a 28.5 degree inclined GTO orbit (no specific orbit dimensions are given.) Another reference, however, on the SpaceX website itself, says that the Falcon Heavy will be able to deliver 53,000 kg to a 28.5 degree inclination LEO, and 12,000 kg to a 27 degree inclination GTO[10].

In addition, it has been reported that the Falcon Heavy can deliver 16,000 kg through Trans-Lunar Injection (TLI) and 14,000 kg through Trans-Mars Injection (TMI). These are the assumptions that we used in this analysis. They are consistent with the Red Dragon study performed by SpaceX and NASA,[12] which quoted that the Falcon Heavy could throw more than 10 tonnes to mars (with a C3 about 10 km^2/sec^2). This report states "Red Dragon injected mass [is about] 6.5 tonnes plus payload."

IV. Calculating throw Mass

Using the throw mass numbers as described above, we generated a curve of C3 as a function of throw mass. We first estimate the excess ΔV from a 200 km parking orbit to each of the trajectories described in TABLE V below:

TABLE V. ΔV Excess from 200 km LEO

	Mass *(kg)*	ΔV Excess *(m/sec)*
LEO	53,000	0
GTO	19,000	2.44
TLI	16,000	3.14
TMI	14,000	3.66

We then fit a curve to this data, as seen in Fig.4.

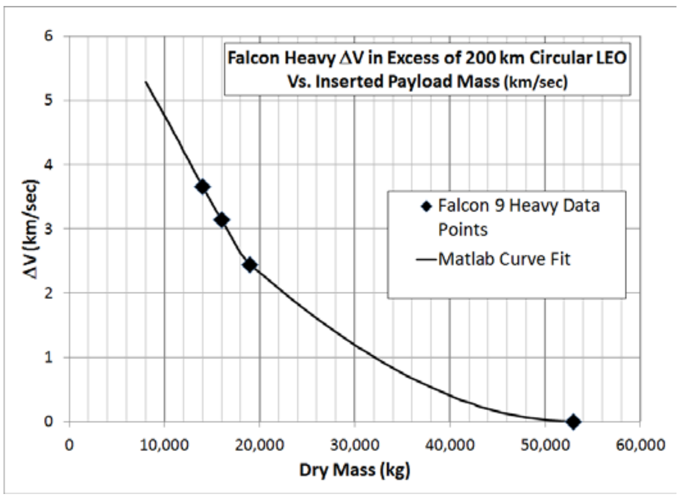

Fig.4. LEO ΔV Excess vs. Payload Mass

For a C3 of 38.8 km^2/sec^2, the ΔV excess is 4.86 km/sec. From these curves, we could further generate a payload mass vs. C3 curve for the Falcon Heavy, shown in Fig.5. This curve was generated by adding the excess velocity from Fig.4 to the state of a 250 km LEO in STK/Astrogator, and then calculating the resultant C3 for each point. From this curve you can see that the estimated mass to a C3 of 38.8 km^2/sec^2 is 9,800 kg. We rounded this to 10,000 kg for the purposes of this study.

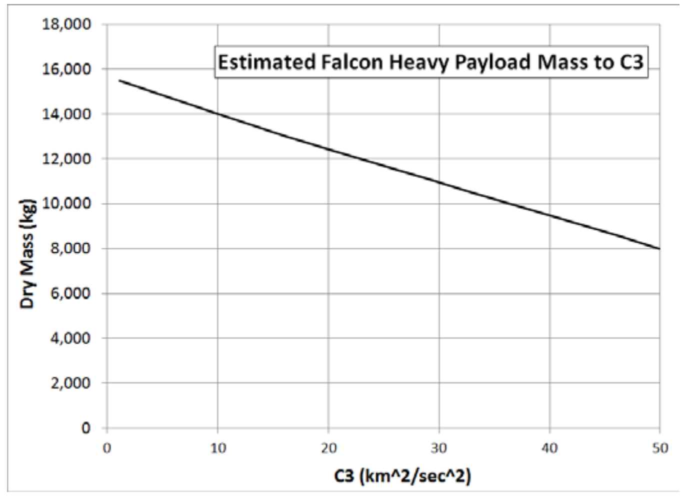

Fig.5. Falcon Heavy Payload Mass vs. C3

V. TMI Finite Burn Modeling

We use estimated parameters for the Falcon 9 second stage based on a Falcon Data Sheet[13] (This is the same stage used for the Falcon Heavy.) This stage, the V1.1 Stage 2, will use the Merlin 1D engine, which is estimated to have a vacuum thrust of 400,000 N, a vacuum ISP of 340 seconds, and a dry mass of 4.7 tonnes. To estimate the fuel usage for stage 2 above the 200 km circular LEO baseline, we made an estimate of what fuel would be required for the appropriate ΔV. Using our STK/Astrogator-based numerical integration, we were able to produce the data in TABLE VI. For the finite maneuver, we then calculated the burn duration to be 420 seconds.

We estimated 15 tonnes of payload (including about 4.7 tonnes for stage 2 dry mass) and derived the fuel load for a finite TMI maneuver to require roughly 50 tonnes of fuel beyond LEO. This compares to the reference data[13] of a 73.4-tonne total fuel load for the stage.

Note that while the SpaceX references claim that the Falcon Heavy can lift 53 tonnes to LEO, this refers to 53 tonnes of payload mass to a 200 km circular LEO. In our case of targeting a Mars Escape C3 of 38.8 km^2/sec^2, we are only assuming a payload mass of 10 tonnes. This leaves excess propellant in stage 2 in LEO for our case. TABLE VI shows an estimated 50 tonnes excess mass in LEO that we consider as excess propellant for the TMI maneuver. (We assume that there is zero propellant excess when inserting 53 tonnes into LEO.)

TABLE VI. Impulsive and Finite ΔV for Stage 2 to 2018 Mars Free Return

	ΔV *(m/sec)*	gc *(m/sec^2)*	Isp *(sec)*	Dry mass *(kg)*	Fuel mass *(kg)*	Total Mass *(kg)*
Impulsive	4863.00	9.80	340.00	15,000.00	49,556.06	64,556.06
Finite	4881.85	9.80	340.00	15,000.00	49,922.28	64,922.28

VI. Reentry Speed and Heatshield

We designed several aerocapture trajectories, shown in Fig.6, by targeting the altitude of perigee to be in the range from 56.5 km to 61.5 km. The numbers in the figure refer to the perigee altitude that caused that trajectory. A perigee of 62 km or greater escapes, and 56 km or less performs a direct reentry. We used the 1976 US Standard Atmospheric Model (developed jointly by NOAA, NASA, and the USAF) and a five-tonne mass with a heatshield area about 10.2 m. We modeled simple atmospheric drag without lift, with a constant Coefficient of Drag (Cd) of 1.3 in the numerical integrator. (This value is similar to Apollo.) Fig.7 shows that the velocity (dashed line, with right-hand axis) during reentry peaks at about 14.2 km/sec velocity. This could possibly be reduced by changing the launch trajectory or Mars flyby conditions, which we will look at in future studies.

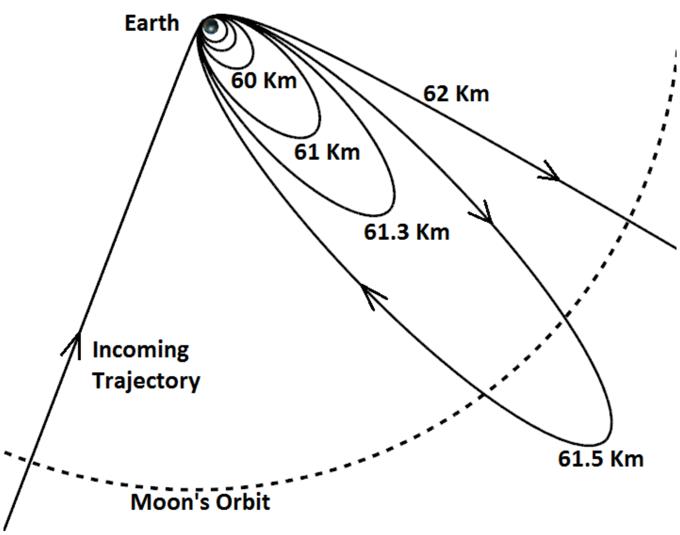

Fig.6. Aerobrake and Reentry Trajectories for Various Perigee Altitudes

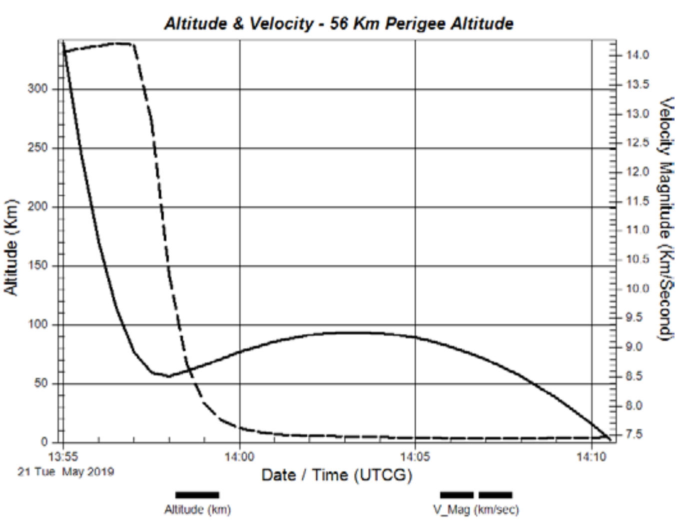

Fig.7. Altitude and Velocity during Direct Reentry

The purpose of the aerocapture before the reentry is to reduce the g-loads on the capsule and the crew. Studies looking at reentry trajectories from Mars have attempted to define acceptable effects on the crew, and in particular, the g force astronauts can withstand after long duration spaceflight. [14]

Fig.8 shows the g force due to atmospheric drag during aerocapture for the range of perigee altitude from 56.5 km to 62 km. The lowest perigee has the highest g force, and the peak load ranges from just under 6 g's to just over 9 g's. Of course, an aerocapture increases the length of the mission, up to an additional 10 days or so if the apogee reaches to lunar orbit. Because the service module of the capsule would be released before aerobraking, the power system of the capsule would likely rely on batteries. The post-aerobraking orbit must be optimized with these considerations, as well as with the reentry conditions.

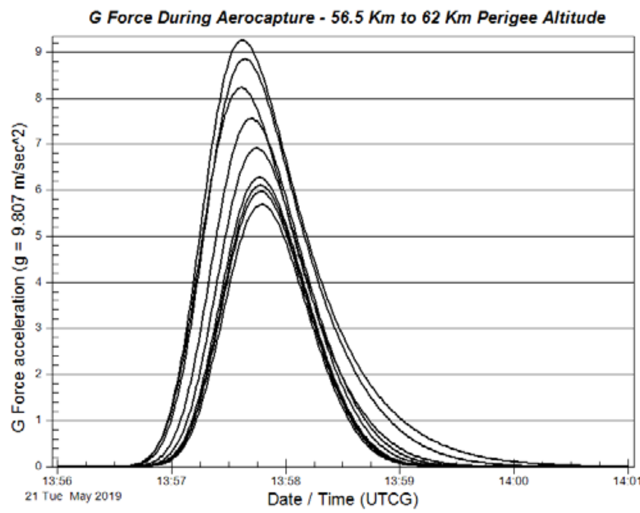

Fig.8. G Force during Aerobraking

After jettisoning structures that must be released before reentry, the spacecraft is estimated to be 5,000 kg. The reentry of a 5,000 kg, 3.6 m diameter spacecraft into Earth's atmosphere present some challenges from an aerodynamics, aerothermodynamics, and thermal protection system (TPS) perspective. The mission calls for both an aerocapture maneuver and a reentry at Earth. To date, no aerocapture maneuver of this type has been attempted either at Earth or other planetary destinations. The atmospheric entry speed for the aerocapture is estimated at 14.2 km/sec, which would make it the fastest reentry of any manned vehicle by far. The fastest, successful reentry of a man-made, but unmanned, vehicle to date was the sample return capsule for the NASA Stardust mission, which reentered at 12.6 km/s with an estimated total stagnation heat flux of 1,200 W/cm^2 [14,15,16].

Assuming that a baseline vehicle architecture for reentry is a Dragon capsule, several TPS related issues that require more study. First, we need to predict with higher fidelity the reentry trajectory and aerothermal loads. The details of the reentry will have a significant impact on the thermal loads. The total heat load during Earth reentry will determine how thick the TPS material will need to be to protect the underlying structure. The peak heat flux will drive the selection of the actual TPS material. The Dragon capsule heatshield uses a variant of Phenolic Impregnated Carbon Ablator (PICA), which has been demonstrated in actual flight environments up to 1,200

W/cm^2[17]. Some very preliminary aerothermal predictions of the Mars return aerocapture maneuver show peak stagnation heating of over 3,500 W/cm^2, dominated by shock layer radiation heating. To date there has not been any ground testing of PICA at those heating heat flux levels. However, there is no data indicating that PICA would not perform as predicted under those conditions, using existing analytical models. Ground testing, such as arc-jet testing, would need to be conducted to verify PICA's performance.

Second, currently operational manned spacecraft have been designed to operate in LEO conditions for missions that are nominally up to six months to a year in duration. The micro-meteoroid and orbital debris (MMOD) and thermal environments for these missions are well characterized and systems are in place to mitigate or control their effects. Some investigation of the space environments during the 500-day Mars mission would be required in order to predict the impacts that these environments may have on the TPS and determine any necessary design changes.

Third, TPS performance during reentry is influenced by two primary factors: peak heat flux and total heat load, as well as other factors. Trade studies will need to be performed to determine if using an aerocapture maneuver to spread out the heat load is an optimal approach. The longer soak times associated with the aerocapture may lead to even more stringent TPS requirements than a direct reentry. Using a direct skipping entry trajectory with banking maneuvers to bleed off energy, may work just as well as the aerocapture/reentry combination. This study would include trajectory optimization, predictions of aerothermal environments, and TPS sizing calculations.

And lastly, the PICA TPS material has not been used or tested at the high heat fluxes that can be expected on the Earth reentry from Mars. As mentioned earlier, ground based arc-jet testing would need to be performed to verify that PICA performs acceptably at those conditions. This test data would also be used to validate the material response models used in TPS sizing analyses. TPS thickness is the primary adjustable design parameter available to satisfy TPS requirements. Higher heating conditions usually lead to thicker TPS. Due to its manufacturing processes, PICA can be fabricated in thicknesses up to about ten inches. If sizing predictions indicate that the required PICA thickness is more than that, another TPS option may be needed. In addition, the thicker the material gets, the more difficult it is to accommodate mechanical loads. PICA is not a mechanically strong material, and induced strains in the material due to flexure of the underlying structure or differential thermal expansion, may lead to mechanical failure (fracture) of the material. In some cases these mechanical loads on the TPS material can be minimized by using a strain isolation pad or by increasing the stiffness of the heatshield structure.

The design and validation of a heatshield capable of performing properly under Mars return conditions will be challenging, but not insurmountable. There are several options in both the trajectory design and the TPS design that trade off each other, and the system will need to be optimized as a whole.

VII. ECLSS Assumptions

We conducted a study to determine the feasibility of designing, developing, testing and integrating an ECLSS in time for an optimized fast trajectory opportunity of January 2018. To do this we developed assumptions about the mission and crew needs on which to base ECLSS architectures.

We used a mission duration of 500 days (d) in a SpaceX Dragon class of vehicle. Crew size is a primary driver so we compared crew sizes from one to four people, and determined that two crew is optimal given mass and volume constraints. The ECLSS was assumed to meet only basic human needs to support metabolic requirements of two 70 kg men, with a nominal metabolic rate of 11.82 MJ/d. Crew comfort is limited to survival needs only. For example, sponge baths are acceptable, with no need for showers.

We also had to determine what technology readiness levels (TRL) were acceptable given schedule constraints. Low TRL technologies may present low mass and high efficiency options, but are untested so present schedule and operational risk. We therefore opted to use high technology readiness levels, or State of the Art (SOA) technologies, although advanced technologies were also studied.

Because volume is highly constrained, we assumed that no EVA is required and no pressure suits are provided. This is because EVA adds significant volume due to the space needed for don/doff and suit/equipment storage. This means that all systems requiring maintenance must be accessible from inside the pressure vessel. The high-risk nature of the mission also drives equipment to be simple, accessible and easy to maintain, with ample spares, and redundant critical systems. Specific assumptions made for each subsystem are described below.

VIII. ECLSS Subsystems

ECLSS is made up of several subsystems: air, water, food, thermal, waste and human accommodations (often referred to as crew systems). Human accommodations takes into consideration the crew's personal, exercise and medical provisions, radiation protection, how much free volume should be required for each person, and pressure suit needs.

Fig.9 shows primary ECLSS subsystems and their interfaces.

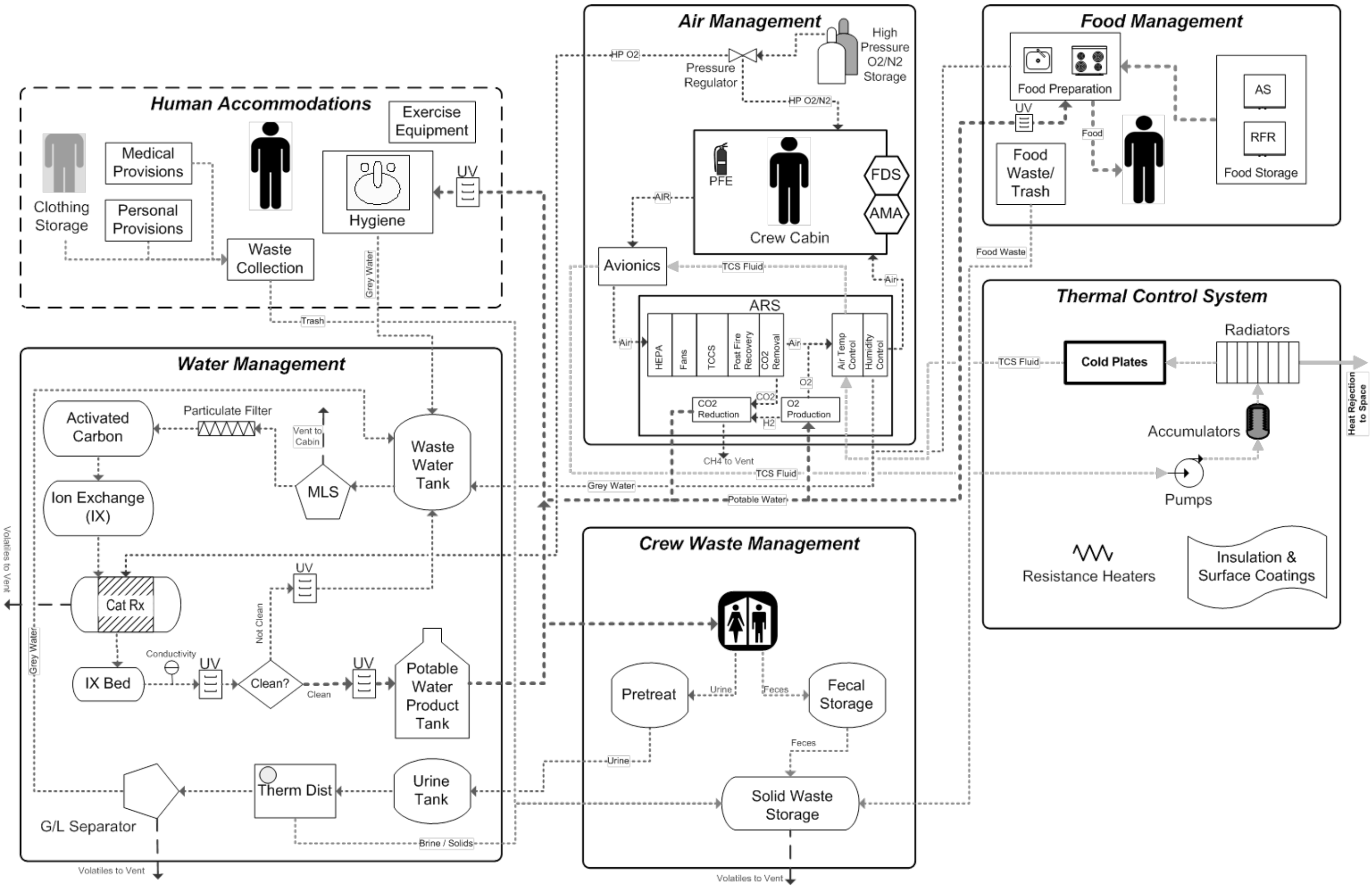

Fig.9. Integrated ECLSS Interfaces

The cabin atmosphere assumes 101kPA (14.7psi) with a nominal makeup of 78% Nitrogen (N_2), 21% Oxygen (O_2), and 1% other. Subsystem components and stored gasses are included for makeup gasses for leakage and gas equivalent storage for one cabin repressurization, though no depressurization accommodations are included (e.g. no pressure suits).

We reviewed numerous technology options to revitalize the cabin air, and determined that an electrolyzer and Sabatier work in tandem to provide necessary air revitalization needs. The electrolyzer produces O_2 and H_2 from water. The Sabatier processes CO_2, but needs H_2 to do so, which it receives from the electrolyzer. The amount of water needed to fully replenish all O_2 requirements was included in the water subsystem calculations, and the difference in H_2 production and H_2 requirements are balanced to determine the stored H_2 required for this subsystem.

The water subsystem assumes basic metabolic requirements for drinking, hygiene, and food processing. Water is added to accommodate O_2 production. Various configurations include no recycling, recycling condensation from the atmosphere, and recycling water from the biological waste. Hygiene water is a variable that can be used to reduce mass for this subsystem, and water recycling is assumed for reduced launch mass, discussed further in section 8, discussing ECLSS sizing. The SOA technology for condensate recovery utilizes multi-filtration, ion exchange, and catalytic oxidation.

Food estimates include packaging and storage factors, as well as a galley and food processing equipment such as heating, preparing, and disposing of waste. While it is probable that all food is consumed, factors to account for food adhesion to the packages are included.

The thermal system is based on a redundant loop radiator, and is sized for the ECLSS power as estimated for the different configurations, plus an assumed 2kW of heat coming from air-cooled components such as avionics.

The waste system provides the interface for collecting and storing human biological waste. Technologies that are used to recycle water from the liquid waste are included in the water system as noted above. Personal provisions are limited to items such as clothing and hygiene products.

Free volume estimations are based on minimum NASA requirements, and are only used to calculate makeup gasses that are required for the full duration. No consideration of privacy, or separate sleeping quarters was contemplated for this study. The mass and volume associated with free volume needs are not included in the ECLSS mass and volume, as we assume that they will be carried at the overall vehicle level, as is typical.

Long term exposure to microgravity has demonstrated that deconditioning must be counteracted with routine vigorous exercise so we assume the use of resistive exercise equipment that provides full range of motion exercise of all primary muscle groups. Medical equipment and supplies are provided to address emergencies.

In addition to spacecraft materials of construction, radiation protection is provided by a water shield made up of water in storage for other subsystems. Other options reviewed were Hydrogen-Impregnated Carbon Nano-fibers, and a Liquid Hydrogen Shell. Further study needs to be done to find creative solutions for radiation protection, including the amount of radiation and the level of risk of a high radiation event deemed to be acceptable. Evaluating these risk factors will include an exploration of crew age, gender, and various exposure types; and will affect future ECLSS trade studies.

IX. ECLSS SIZING

The crew's metabolic rates drive parameters such as O_2 consumption and CO_2 production. TABLE VII shows interface values for each subsystem based on an average crew member (CM) of 70 kg and metabolic rate of 11.82 MJ/d (17). As demonstrated, the most significant consumable is water at 3.909 kg/CM-d, followed by 0.835 kg/CM-d of O_2 and 0.617 kg/CM-d of dry food.

We used a Water Lean approach, with a reduced water requirement of 1.2 kg/CM-d as shown in TABLE VIII.

TABLE VII. CREW METABOLIC INTERFACE VALUES (HANFORD, 2006, TABLE 3.3.8) [18]

Interface	**Avg. Single Crew-Member per Day**			**Total 2CM 500d***	
Overall Body mass		70	kg	140	Kg
Respiratory Quotient		0.869		N/A	
	Input	***Output***			
Air					
Carbon Dioxide Produced		0.998	kg /CM-d	998	Kg
Oxygen Consumed	0.835		kg /CM-d	835	Kg
Water					
Potable Water Consumed	3.909		kg /CM-d	3,909	Kg
Fecal Water		0.091	kg /CM-d	91	Kg
Respiration and Perspiration Water		2.277	kg /CM-d	2,277	Kg
Urine Water		1.886	kg /CM-d	1,886	kg
Metabolically produced Water	0.345		kg /CM-d	345	kg
Food					
Dry Food Consumed	0.617		kg /CM-d	617	kg
Thermal					
N/A					
Waste					
Fecal Solid Waste (dry basis)		0.032	kg /CM-d	32	kg
Perspiration Solid Waste (dry basis)		0.018	kg /CM-d	18	kg
Urine Solid Waste (dry basis)		0.059	kg /CM-d	59	kg

*Does not include packaging and storage containers.

TABLE VIII. CREW QUALITY OF LIFE WATER CONSUMPTION VALUES (HANFORD, 2006) [18]

	Mass Required (kg/CM-d)			**Total 2CM 500d**
	Water Rich	***Nominal***	***Water Lean***	***Kg***
H_2O – Shower	2.30	0.00	0.00	0.0
H_2O – Body Wash	2.00	2.00	0.91	910
H_2O – Urine Flush	0.50	0.50	0.30	500
H_2O – Oral Hygiene	0.37	0.37	0.00	0
Totals:	5.17	2.87	1.21	1410

For this study we conservatively used 0.758 kg/CM-d as shown in TABLE IX. Given the limitations of the mission, this will certainly be reduced by eliminating or reducing items such as toilet paper, tape, dry wipes, detergent, disinfectant and clothing.

TABLE IX. CREW PERSONAL PROVISIONS (HANFORD, 2006)[18]

Item	**Mass (kg/CM-d)**	**Total 2CM 500d (kg)***
Feminine Health	0.008	8
Toilet Paper	0.028	28
Gloves	0.007	7
Dry Wipes	0.013	13
Detergent	0.058	58
Disinfectant	0.056	56
Paper	0.077	77
Tape	0.033	33
Clothing	0.486	486
Total (Female)	**0.766**	**766**
Total (Male)	**0.758**	**758**

*Does not include packaging and storage containers.

For this study, a nominal leak rate and a tolerable volume (shown in TABLE X) have been used for calculating the makeup gas. We also assume a standard sea level atmosphere of 101.353 kPa (14.7psi) and a partial pressure makeup of 21% O_2, 78% N_2, and 1% other.

TABLE X. ECLSS ATMOSPHERE CONTROL (HANFORD, 2006)

Parameter	Value	Unit	Total 2CM 500d*
Leak Rate			
Lower	0.00	%/d	0%*
Nominal	0.05	%/d	25%*
Upper	0.14	%/d	70%*
Cabin Volume			
Tolerable	5.10	m^3/CM	10.2 m^3
Performance	9.91	m^3/CM	19.8 m^3
Optimal	18.41	m^3/CM	36.8 m^3

*Assumes Nominal Leak rate with Tolerable Cabin Volume resulting in a leaked atmosphere of 2.25 m^3

Contingency operations that require pressure suits such as a full cabin depressurization are not included in this analysis. There are new technologies for post fire atmospheric recovery that preclude the need to vent the toxic atmosphere. Additionally, although minor, the mass of the cabin atmosphere at launch is not included but the mass of all gasses that must be stored in tanks is included. It is assumed that all gasses are stored at high pressure. This carries a minor weight penalty relative to cryogenic storage, but the operational simplicity is considered to be conservative and the overall mass penalty is negligible compared to total system mass for this study.

Based on the biological needs, the largest total consumables by mass are water, oxygen, and dry food (in order). For waste production, the largest producers by mass are waste evaporative water, waste water, and CO_2. Given that water is both the highest required consumable and the largest waste product, a water recycling system is highly desirable in order to reduce the overall system mass. In addition, electrolyzer and Sabatier systems working in tandem reduce mass by using waste products from one system as a feed source for the other. Fig.10 shows the interdependencies between the various ECLSS recycling systems.

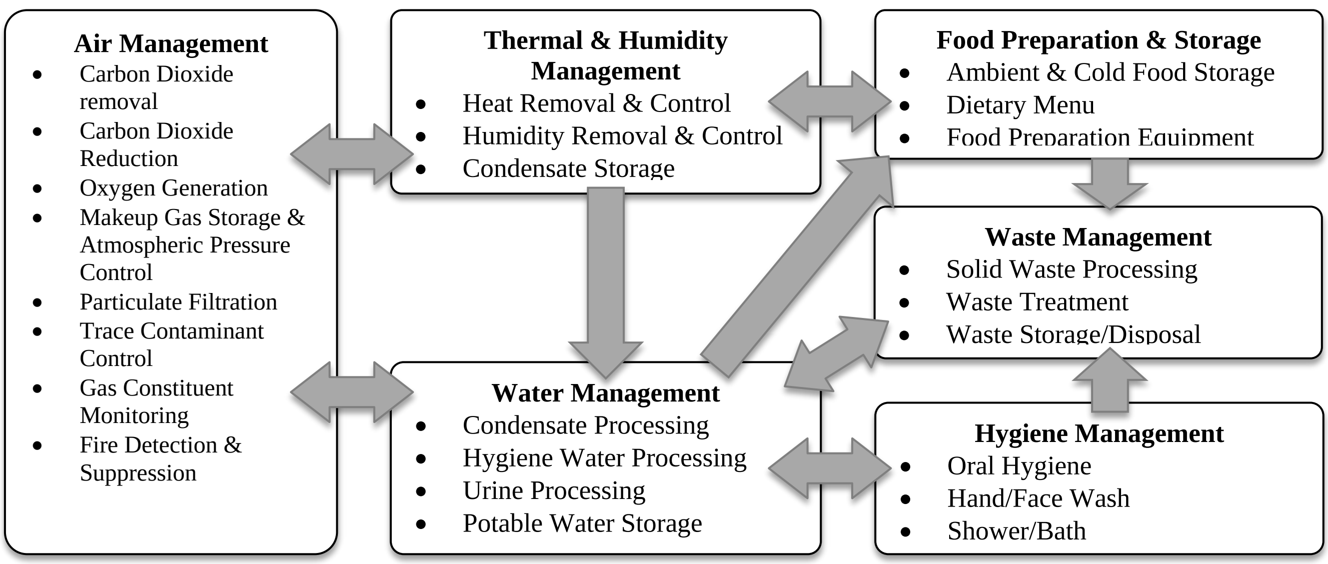

Fig.10. ECLSS Consumable Dependency Interdependency Diagram

We studied various configurations of technologies to minimize the mass of the ECLSS architecture at launch, while minimizing complex systems or technology development. Because water, oxygen and CO_2 are primary drivers of ECLSS as shown above, we emphasized water reclamation, CO_2 removal, and O_2 generation.

Life support systems that depend on storage of all consumables are typically the simplest and most reliable, but if used for this mission would result in untenable mass and volume. For mission durations that exceed approximately 240 CM-days, the benefit from recycling of the life support consumables trades well[19]. For this reason, we determined to focus on the simplest recovery systems that use proven technology and provide the most effective recycling capability.

We studied five options based on SOA technologies in operation, which were then combined to produce the most cost effective and proven architectures for further study. Each option had a different focus as follows:

- No consumable reclamation (100% supply of water and oxygen at time of launch)
- Processing humidity condensate (includes hygiene water) to recover 100% of the water as potable; hygiene water recovery is accomplished through air-drying of wash cloths

- Resupply of oxygen via electrolysis of water
- Reduction of CO2 via a Sabatier to make potable water
- Processing of urine and flush water to recover 75% of the water as potable

As part of the trade study we performed sensitivity analyses to determine the impact of a range of crew sizes and mission durations (Fig.11 and Fig.12). In addition, we quantified the impact of utilizing various spare philosophies with only critical spares, dual or in some cases triple spaces (Fig.13). In addition, we compared advanced life support technologies with operational SOA technologies. The base mass of each system was included, and scaled where appropriate based on such parameters as the number of modules (e.g. pressure vessels) or crew size during. A mass growth allowance of 15% was included for all empty mass hardware to account for installation factors. We also compared the effects of two SpaceX Dragon class vehicles versus one, to increase volume for consumables and crew, shown in Fig.14 below.

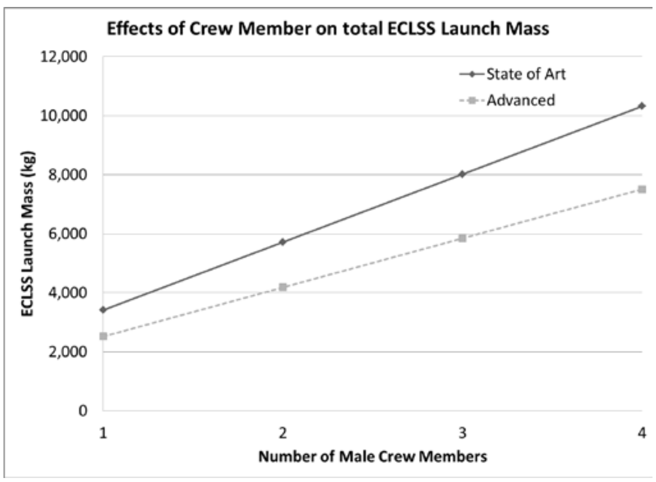

Fig.11. Crew Member vs. Total ECLSS Launch Mass

Fig.14 shows the resultant ECLSS launch mass for various ECLSS technology configurations and variables. As the chart shows, there are several feasible solutions when comparing the total system mass to the preliminary ECLSS Mass Target of 6,000 kg or lower.

After narrowing down the various configurations to these solutions, a final technology suite was chosen based on two crewmembers and a 500-day mission. TABLE XI shows the resulting masses, volume and power needs.

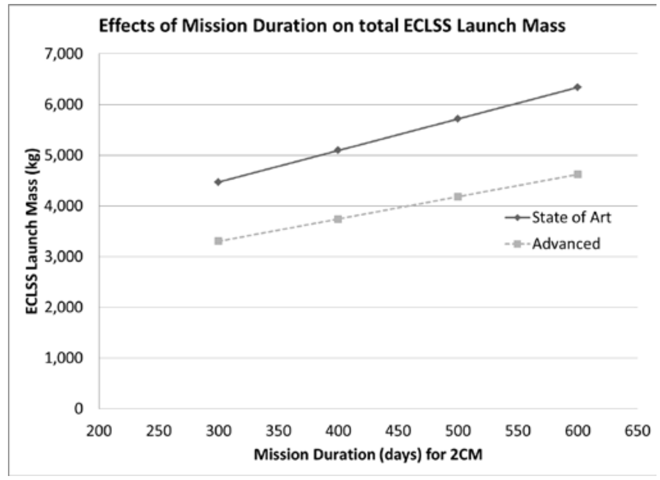

Fig.12. Effects of Mission Duration on total ECLSS Launch Mass

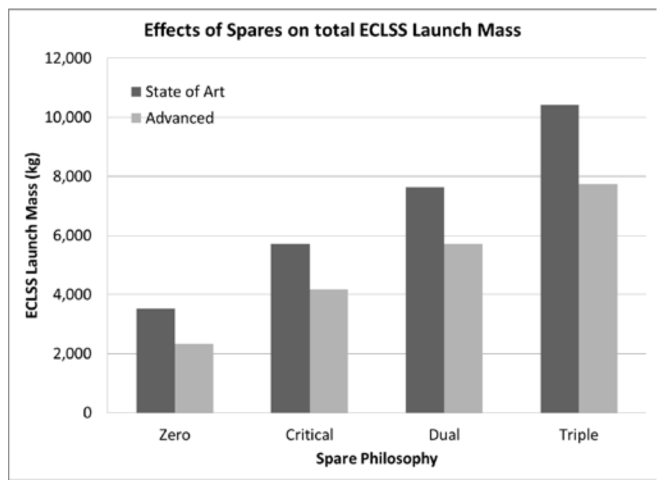

Fig.13. Effects of Spares on total ECLSS Launch Mass

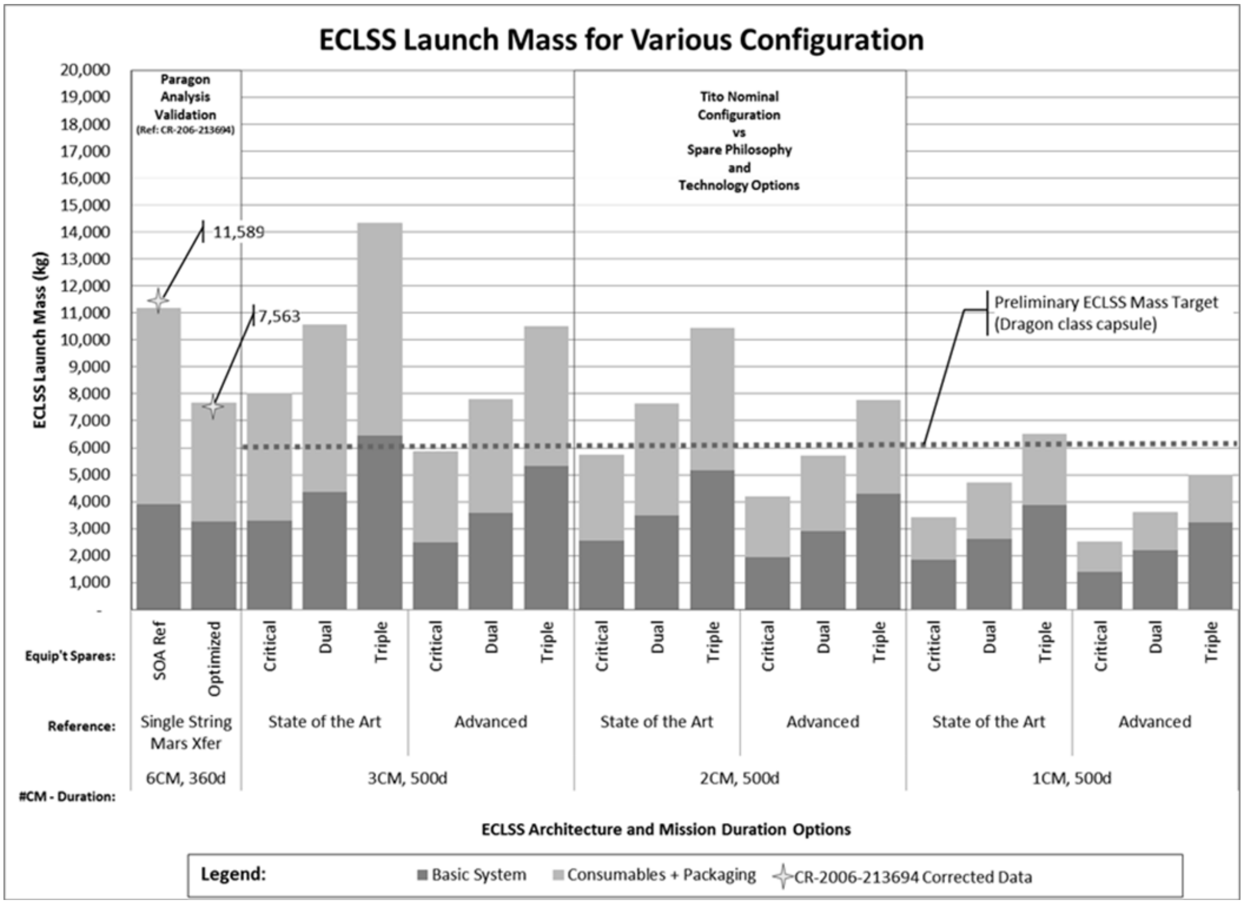

Fig.14. ECLSS Launch Mass for Various Configurations. Several meet the target launch mass of <6,000 kg.

TABLE XI. ECLSS MASTER EQUIPMENT LIST

Component	System Total*			
	Mass *(kg)*	**Vol†** *(m^3)*	**Peak** *(W)*	**Avg** *(W)*
Air *(high press storage, mole sieve, Sabatier, electrolysis)*	897	1.7	2,626	1,870
Water *(tanks, multi-filtration, ion exchange, distil, catalytic oxidation)*	2,235	5.1	529	193
Food *(dry packed, storage, water heater)*	1,384	14.0	1,860	39
Thermal‡ *(nominal redundant single loop)*	479	1.0	300	99
Waste *(urine, feces, and vomitus collection, and solid waste stabilization/storage)*	259	0.7	174	7
Human Accommodations *(clothing, hygiene, medical provisions, radiation shelter, etc.)*	347	1.8	-	-
Basic System:	2,432	6.6	5,489	2,208
Consumables:	3,131	17.7	-	-
Total:	**5,545**	**24.2**	**5,489**	**2,208**

*Mass and Power estimates based on ANSI/AIAA G-020-1992, Guide for Estimating and Budgeting Weight and Power Contingencies for Spacecraft Systems
† Volumes are total volume and does not account for packaging factors.
‡ Thermal System assumes that the base spacecraft has thermal control capability for existing systems.

X. ECLSS DEVELOPMENT SCHEDULE

At a very high level, Fig.15 summarizes a development schedule that follows a traditional design cycle required to successfully execute on a program of this magnitude for advanced life support systems. The schedule assumes that SOA technologies are used, thereby minimizing schedule risk from the pitfalls associated with technology development. Development incurs two major risks: (1) the hardware will not be ready in time; (2) the operational history and confidence in the technical solution remain immature due to the hard deadline of the contemplated mission, increasing risk to the overall mission success. These factors will likely bias the choice of technology toward SOA.

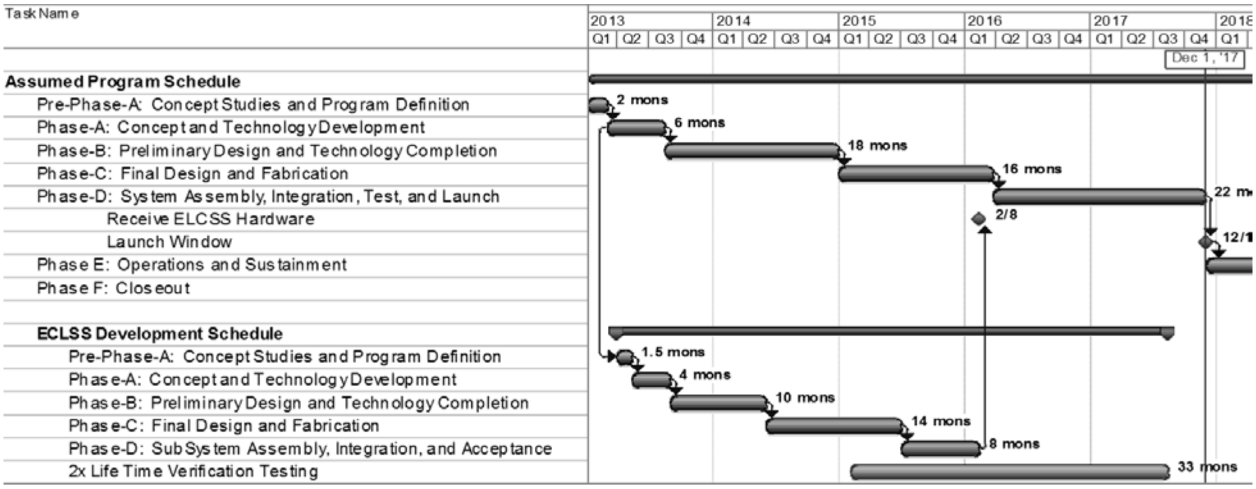

Fig.15. Notional Development Schedule

It is assumed that the ECLSS hardware would be required to be design, developed, and delivered to the Prime integrator approximately two years ahead of the launch date to accommodate system level integration, and sufficient testing to thoroughly vet the practical operating characteristics, limits and weaknesses. This drives an ECLSS hardware delivery date of February, 2016.

XI. ECLSS Summary

Preliminary results indicate several candidate options exist that are based on current state of the art TRL 9 technologies. Inclusion of select TRL 7-8 technologies may reduce consumables and provide increased margin and redundancy at the expense of program schedule and cost risk.

The ability to maximize both volume and mass available by the recycling of consumable materials is critical for any long duration mission. Given a launch date of January, 2018, we estimate there to be limited but sufficient time to design, manufacture, test, and integrate the spacecraft ECLSS.

We are confident that an ECLSS architecture for a 501-day mission with one or two crewmembers can be developed within the time frame allowed.

XII. Crew Selection and Training

Long duration isolated, confined environments have analogies on Earth. These include over-winter stays at the South Pole, nuclear submarine missions, Biosphere 2, the Mars-500 experiment and the multinational MIR and ISS missions. These experiences have shown that crew dynamics, well-being and mental health can be positively and negatively impacted by a wide variety of factors. The limited experience has also pointed toward crew selection criteria that deviate from the typical A-type personality bias and take into account factors of personal accommodation and behavioral self-knowledge, sometimes referred to as resilient personalities. Other factors include the dynamics of male/female mixed crews, and the fact that the inspirational aspect and impact on the human population may be enhanced by representation by both genders.

The volume and mass considerations have limited the crew number to two individuals. At a current value of roughly 7 m^3, the representative spacecraft free volume is deemed adequate. However, given the requirement for food and water storage and additional equipment and access, the free volume could shrink considerably. Prior studies we have performed indicated that, for this mission duration, crew volumes of less than 3-5 m3 per person would border on untenable. Available crew volume is a significant consideration for this mission[20]. The increased need for isolation during privacy functions (defecating, body cleansing), and activities related to crew exercise (using simple stretchable resistance devices) suggest more space than some historical two-crew vehicle such as Gemini which had approximately 1.25 m^3/person.

Long duration also requires increased mission control/crew training and psychological bonding. The crew must implicitly trust mission control team members, who must be highly sensitive to the psychological needs of the crew. This trust, and the dynamics under which it evolves, is both a starting point and goal of the training of the crew/mission control team.

Finally, crew selection must be seen as a continuous process. Training, personal issues, or even accidents mean that crew selection must start with a pool of candidates and logically winnow down the final two people through testing, training, and evaluation. A team of experts to evaluate the candidates will be required to provide input to this process and develop a team that has a high probability of success. Experience has shown that it is extremely difficult to train, select, and evaluate a crew team without those individuals having had experience living and working in an isolated confined environment for an extended period of time, preferably for the full mission duration, but for at least six months.

XIII. Role of NASA

NASA is critical for providing key technologies, technical expertise, and test facilities – particularly in entry-systems, ECLSS, computational support and human factors. The support envisaged is similar to that which NASA's predecessor organization, the National Advisory Committee of Aeronautics (NACA), provided for the burgeoning aeronautics industry from 1915-1958.

The NACA Charter of 1915 stated that it was "to supervise and direct the scientific study of the problems of flight, with a view to their practical solution, and to determine the problems which should be experimentally attacked, and to discuss their solution and their application to practical questions."[21] NASA has begun to perform this function for the emerging commercial space industry, and its role in this Mars free-return project is crucial. It is useful to examine some of the key aspects of NACA and how it will be applied to privately-funded human missions to other worlds with the ultimate goal of human expansion into the Solar System. A cogent discussion of how NASA can perform a "NACA-like" function for the emerging commercial space sector has been provided by Glenn Bugos and John "Jack" Boyd of the NASA Ames Research Center.[22] We extract the following concepts from them.

NASA's space support to this and other private space projects as described by Bugos and Boyd should exhibit several key features. First, NASA must understand that these private missions are not NASA programs and that it should not be directly involved in hardware and operational aspects of the mission. A second key concept is that NASA should address "reverse salients" in a technological sense as defined by technology historian Thomas Parke Hughes.[23] In military terms a reverse salient is a part of a front line that cannot keep up. NACA addressed those technological reverse salients for aeronautics in the 1920s-1950s such as swept wings, landing gear, aviation safety etc. For our expansion into the solar systems, several reverse salients emerge: very high speed atmospheric re-entry, communications, radiation and zero-gravity human survival, truly closed-loop life support systems and, eventually in-situ resource utilization. A third concept explained by aerospace historian Walter Vicenti[24] is that NACA did not focus on inventing new theory or concepts – rather they focused on knowing the engineering limits of their current technology. For the proposed missions to succeed, NASA must provide not only key technological information on such things as re-entry options, but also the limits of what we know about it. Fourth, NASA-provided information must be available to all, not just this team, so that it will enable others to verify it and use it for their own purposes.

NASA needs to support this mission and others as its predecessor NACA did for the aviation industry. The enabling act that established NASA in 1958 transferred all functions of NACA to NASA and included space activities in these functions.[25] Subsequent NASA legislation has only strengthened and expanded these objectives, as evidenced in the NASA Authorization Act of 2010 supporting Commercial Orbital Transportation Services. NASA's support of privately-funded human Mars missions is another great leap in a process begun in 1915.

XIV. Concluding Remarks

A manned Mars free-return mission is a useful precursor mission to other planned Mars missions. It will develop and demonstrate many critical technologies and capabilities needed for manned Mars orbit and landing missions. The technology and other capabilities needed for this mission are needed for any future manned Mars missions. Investments in pursuing this development now would not be wasted even if this mission were to miss its launch date. Although the next opportunity after this mission wouldn't be for about another 13 years, any subsequent manned Mars mission would benefit from the ECLSS, TPS, and other preparation done for this mission, In fact, often by developing technology early lessons are learned that can reduce overall program costs. Working on this mission will also be a means to train the skilled workforce needed for the future manned Mars missions.

In January, 2018, there is a rare opportunity for a fast free-return flyby of Mars with total mission duration of 501 days. The trajectory will only need small correction maneuvers after trans-Mars injection. Based on open-source information SpaceX is expected to have the Falcon Heavy ready in time, and it seems capable of launching the needed mass. The reentry conditions at Earth are challenging but there are no known reasons they cannot be met with several options that we have described. A capsule the size of the SpaceX Dragon has the appropriate volume for a two person crew. This investigation has shown that the ECLSS technology can be developed and tested in time for this mission.

The purpose of this investigation was to understand the technical feasibility of this mission. The programmatics of this mission still need to be investigated, and the authors are working on this now.

The analysis presented here points the way toward an ambitious but high pay-off journey to Mars to show the world that the human race has the vision and the means to become a multi-planet species. Not unlike the flights across the Atlantic in the 1920s, the pilots will be challenged by unforeseen circumstances and the uncertainties of equipment, environment, and endurance.

As we have seen, a motivated space organization was able to accomplish a very complex Apollo 11 mission which included multiple rendezvous, multiple maneuvers, a landing, and a launch from the Moon. In our opinion this Mars free return mission is a simpler mission that can be achieved in a similar time frame, especially considering the modern testing capabilities as well as the advanced modeling and simulation technology available today.

Sending humans on an expedition to Mars will be a defining event for humanity as well as an inspiration to our youth. Social media provides an opportunity for people to meaningfully participate in the mission, likely making this the most engaging human endeavor in modern history. The mission will address one of the most fundamental technical challenges facing human exploration of space, keeping the humans alive and productive in deep space.

We are confident that this mission is feasible, and this investigation did not reveal any showstoppers. We believe that the cost would be significantly less than previous estimates for manned Mars missions, and could be financed privately without relying on government funding. We also believe that with current and near-term technology it is possible to be ready for a Mars flyby launch in January, 2018 and that we should not pass on this mission since the next opportunity does not occur again until 2031.

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BIOGRAPHIES

Dennis A. Tito is the Chief Executive Officer of Wilshire Associates Incorporated, a leading provider of investment management, consulting and technology services.

Tito earned a B.S. in Astronautics and Aeronautics from NYU College of Engineering and a M.S. from Rensselaer in Engineering Science. He began his career as an aerospace engineer with NASA's Jet Propulsion Laboratory at the age of 23. While serving at JPL, he was responsible for designing the trajectories for the Mariner spacecraft missions to Mars and Venus. Although he left to pursue a career in investment management, Tito remained interested in and committed to the exploration of space.

On April 28, 2001, Tito, who served as a crewmember of an eight-day Russian Soyuz taxi mission to the International Space Station, fulfilled his 40 year dream to orbit the earth.

Grant Anderson, P.E., Senior Vice President of Operations/Chief Engineer and co-founder of Paragon, has 28 years of experience designing power, thermal and life support systems for human rated spacecraft. Anderson is responsible for product fulfillment from requirements to final product shipment. From 1999 to 2007, Anderson led the Thermal Control System and Environmental Control and Life Support System design for every human rated spacecraft program undertaken by Lockheed. He is currently the Principal Investigator (PI) for high-reliability flexible pressurized structures as well as the former Program Director under which Paragon's CCDEV1 effort was successfully executed and resulted in the PDR level Air Revitalization System for current commercial spacecraft.

Additionally, he has played major roles in single-loop fluid testing for advanced active control thermal systems, advanced radiator structural technology and overall ELCSS definition for multiple commercial spacecraft. Previous experience includes being Project Design Lead for the ISS solar arrays, the largest solar arrays ever built; Cabin design and build of the first CEV mockup, now on display at Johnson Space Center; Engineering lead for experimental flight hardware on five shuttle flights, two Mir missions, a Russian Progress, and the first commercial payload on ISS; he also led the engineering on a flight qualified Micro-gravity Cell Culture System.

Anderson holds two degrees from Stanford University in Mechanical Engineering (BS) and Aeronautical and Astronautical Engineering (MS) and is a registered Professional Engineer. He is an Associate Fellow if the American Institute of Aeronautics and Astronautics recognized for expertise in Life Support Systems.

Barry W Finger is Paragon's Director of Advanced Concepts and Business Developments where he directs early customer funded pursuits, proposals for externally funded programs, and internally funded product maturation initiatives. He also manages Paragon's commercial crew product line and serves as one of the companies ECLSS subject matter experts. Since joining the company in 2006, Finger has held several senior technical leadership roles. Most recently, he served as the Commercial Crew Transport–Air Revitalization System (CCT-ARS) Project Manager, for which he also developed the system architecture and led the proposal effort. Under a fast-paced 10 month Space Act Agreement with NASA during the first Commercial Crew Development program (CCDev1), Finger's team designed the CCT-ARS through flight PDR and manufactured and tested a high fidelity EDU that verified all primary performance parameters. Finger also served as the ECLSS technical lead on two previous human spaceflight programs for Paragon – the Excalibur Almaz ECLSS program (2008-2010) and the Rocketplane Kistler COTS ECLSS program (2006-2007).

Prior to joining Paragon, Finger served as the ECLSS Manager for Bigelow Aerospace. In this position he led a lean team of engineers and technicians that developed life support systems for commercial human spaceflight vehicles. His was responsible for the detailed design of vehicle systems and the requisite company test facilities, as well as coordination and planning of the work across two company facilities. Finger previously served as the International Space Station (ISS) Sustaining Engineering (SE) Program Manager for Honeywell Engines Systems and Services (ES&S). In this role he managed all ISS SE activities related to the support of ES&S flight hardware and systems on-board ISS. Finger also served as an ECLSS design engineer for Hamilton Sundstrand at the Johnson Space Center (JSC) and the Bionetics Corporation at the Kennedy Space Center (KSC). He was the lead engineer for the Post Processing Subsystem, which was operated continuously for over one year as part of the JSC Advanced Water Recovery System Integrated Test. At KSC he served as the Lead Engineer for the Dynamac Corporation where he managed the activities of the Advanced Life Support and Space Biology (ALSSB) engineering and technical support staff.

Finger earned a Master of Science in Aerospace Engineering and Bachelor of Science in Aerospace Engineering from the University of Florida in 1992, and 1990, respectively.

John Carrico is the Chief Scientist at Applied Defense Solutions, working on flight dynamics mission analysis, operations, development, and systems integration. Carrico has supported several operational Earth and Lunar spacecraft missions, including the Clementine (DSPSE) lunar mission, GOES, GRO, ISTP, ROCSAT, WMAP, UFO, FLTSAT, IBEX, and LCROSS. He has designed and written trajectory design algorithms and software used worldwide for analysis and spacecraft operations. Carrico has been working in flight dynamics for 24 years in the areas of rendezvous; low-Earth; geostationary; lunar; formation flying, proximity operations, Lissajous; and asteroid and interplanetary missions. Carrico also has performed research and development in the chemical monitoring and detection field.

Gary A Lantz currently serves as a Sr. Aerospace Engineer in Paragon's Advanced Concepts and Integration Department. He is experienced in all phases of vehicle development and has a broad knowledge of vehicle subsystems including structures, propulsion, electronics, avionics, environmental controls, and thermal control. His direct responsibilities have included vehicle configuration, aerodynamics, performance and trajectories, configuration and data management as well as systems engineering processes and procedure in commercial aviation, Department of Defense, and NASA.

Prior to joining Paragon, Lantz worked at Northrup Grumman as a Sr. Project Integration Engineer, Systems Engineering & Integration Team Lead, and Deputy Facility Manager. At Science Applications International Corporation (SAIC) he managed program Level I safety and mission assurance requirements for the Constellation Program, Lunar Landing Projects, ORION Crew Exploration Vehicle, and ARES launch vehicle. At Rocketplane Limited he led the overall vehicle architecture and design, trajectory evaluation, and weight estimations. Additional vehicle integration and performance from concept through first flight were developed at Cessna's Aircraft Company's on the Cessna's CJ2+, CJ3, and Citation Mustang vehicles as well as customer delivery and flight test certification of the Citation Excel, Encore, and Bravo. At Helicomb International, he developed composite maintenance and repair procedures, operated autoclaves, developed composite structure tooling, and performed both destructive and non-destructive testing for DOD certification. Lantz received a B.S. in Mechanical /Aerospace Engineering from Oklahoma State University.

Michel Loucks received a BA in Physics/Astronomy from Whitman College in 1985 and an MS in Aerospace Engineering Sciences from the University of Colorado in 1991. He co-founded Space Exploration Engineering (SEE) in 1995, after working as an operations and trajectory planning expert for Orbital Sciences Corporation. While at Orbital, he was the mission operations manager for the OXP-1 and OXP-2 spacecraft. With SEE he has served as the orbit raising manager for the OSC Seastar (aka Orbview-II) spacecraft, both for the initial orbit-raising activities in 1997 and again for orbit raising operations in 2010. He has served as a trajectory consultant for NASA Goddard Spaceflight Center on the WMAP, TRIANA and LCROSS missions. He served as trajectory lead for the NASA Interstellar Boundary Explorer (IBEX) mission in 2008 and is currently serving as the cis-lunar trajectory lead for the NASA/Ames Lunar Atmosphere and Dust Environment Explorer (LADEE) which is scheduled to fly in 2013. Loucks regularly teaches a trajectory design class on behalf of Analytical Graphics Inc., using the STK/Astrogator software tool for space mission design and operations. He co-wrote the 4-day class which has been taught in Japan, India, Canada, Europe and the US regularly since 2004. Loucks has co-authored papers on IBEX, lunar lander trajectory design, and the application of fuzzy logic in lunar landing and proximity operations.

Taber MacCallum is CEO, CTO and co-founder of Paragon Space Development Corporation, a leading ECLSS company.

He is presently involved in the design of life support and thermal control systems for commercial manned suborbital spacecraft, as well as hazardous environment life support technology development for the US Navy divers. Being a Master Diver, he is the program's test diver. He was Principal Investigator on four biological experiments on Shuttle, Mir and ISS, which produced the first animals to complete multiple life cycles in space.

MacCallum was a member of the first two-year mission living and working inside Biosphere 2, a three-acre materially closed ecological system that supported the life of the eight human inhabitants – the first fully bioregenerative life support system for human life support on another planet. He was responsible for the design, implementation and operation of the atmosphere and water management systems as well as the self-contained paperless analytical laboratories for Biosphere 2 and its Research and Development Center. MacCallum also served as Safety Officer and Assistant Medical Officer on the Biosphere 2 Resident Research Team. He has published numerous papers resulting from his work at Biosphere 2, on space related issues, medical issues and on the experience of living and working in an Isolated Confined Environment.

Dr. Jonathan B. Clark is an Associate Professor of Neurology and Space Medicine at Baylor College of Medicine and teaches operational space medicine at BCM's Center for Space Medicine (CSM). Dr. Clark serves as Space Medicine Advisor for the National Space Biomedical Research Institute (NSBRI). He also is Clinical Assistant Professor in Preventive Medicine and Community Health at the University of Texas Medical Branch in Galveston, Dr. Clark is Medical Director of the Red Bull Stratos Project, a manned stratospheric balloon freefall parachute flight test program, which on 14 October 2012 successfully accomplished the highest stratospheric freefall parachute jump from 128,100 feet, achieving human supersonic flight without an aircraft at 834 miles per hour. His professional interests focus on the neurologic effects of extreme environments and crew survival in space.

He received Bachelor of Science from Texas A&M University, his Doctor of Medicine from the Uniformed Services University of the Health Sciences and a Master's of Public Health from the University of Alabama in Birmingham. Dr. Clark completed his residency at Bethesda Naval Hospital and is board certified in Neurology and Aerospace Medicine. Dr. Clark is a Fellow of the Aerospace Medical Association.

He was a Member of the NASA Spacecraft Survival Integrated Investigation Team from 2004 to 2007 and a Member of the NASA Constellation Program EVA Systems Project Office Standing Review Board from 2007 to 2010. He served as Chief Medical Officer for Excalibur Almaz, an orbital commercial space company, from 2007 to 2012.

Dr. Clark worked at NASA from 1997 to 2005 and was a Space Shuttle Crew Surgeon on six shuttle missions. He also was Chief of the Medical Operations Branch and an FAA Senior Aviation Medical Examiner at the NASA JSC Flight Medicine Clinic. Prior to joining NASA in 1997, Dr. Clark was Head of the Spatial Orientation Systems Department at the Naval Aerospace Medical Research Laboratory in Pensacola FL; Head of the Aeromedical Department at the Marine Aviation Weapons and Tactics Squadron One in Yuma AZ; and Head of the Neurology Division and Hyperbaric Medicine at the Naval Aerospace Medical Institute in Pensacola FL. Dr. Clark served 26 years on active duty with the U.S. Navy. He was a DOD Space Shuttle Support Flight Surgeon covering two space shuttle flights and flew combat medical evacuation missions in Operation Desert Storm with the U.S. Marine Corps. He qualified as a Naval Flight Officer, Naval Flight Surgeon, Navy Diver, U.S. Army parachutist and Special Forces Military Freefall parachutist.

Jane Poynter is the President and co-founder of Paragon. She holds a patent for the Autonomous Biological System, a key payload life support technology that was used to support the life of aquatic animals in experiments that resulted in the first animals to complete multiple generations in space. Ms. Poynter has logged over 3 years of experiment time in space on the Shuttle, Mir, and ISS, with the longest experiment lasting 18 months on board ISS. She has also served as SPACEHAB's Chief Scientist for its Ecosystem in Space experiment on the International Space Station, and three experiments with ants, bees and fish, which flew on STS-107, the Space Shuttle Columbia. In collaboration with MIT and Draper Laboratory, she developed design approaches and mass/cost analyses of bioregenerative life support systems for long duration space exploration.

Poynter was a member of the original team to live and work inside Biosphere 2 for the first two-year mission, for which she led the design and implementation of the Intensive Agriculture. She also led the project's Biospheric Research and Development Center. She has published numerous papers on Biosphere 2 systems and the behavioral aspects of living and working in Isolated Confined Environments.

Tom Squire earned both a B.S. in Aerospace Engineering and a M.S. in Mechanical Engineering from Arizona State University. He has 26 years of experience in the field of design and analysis of thermal protection systems (TPS) for atmospheric entry vehicles. He began his career at Aerotherm Corporation, where he developed numerical tools to predict the thermal performance and material response of TPS materials on ballistic reentry vehicles. He spent several years doing similar work at Lockheed Missiles & Space. In 1991 he began working for ELORET Corporation as a contractor in the Thermal Protection Materials & Systems Branch at NASA Ames Research Center. He was responsible for performing finite element thermal and mechanical response analyses of TPS designs and materials in support of various NASA vehicle development projects, including X-30 (NASP), X-33, X-34, X-37, and Orion. He has also supported TPS material development projects, such as those for Silicone Impregnated Reusable Ceramic Ablator (SIRCA), Ultra-High Temperature Ceramics (UHTC), and Toughened Uni-piece Fibrous Reinforced Oxidation-Resistant Composite (TUFROC). Tom was one of the lead developers of the web-based TPSX Materials Database (http://tpsx.arc.nasa.gov). In 2000, he accepted a civil service position with NASA Ames and for the last four years he has served as Deputy Chief of the Thermal Protection Materials Branch.

Simon P. ("Pete") Worden, Ph.D. (Brig. Gen., USAF, Ret.) is Director of NASA's Ames Research Center (ARC) at Moffett Field, Calif. Before joining NASA, he held several positions in the United States Air Force and was research professor of astronomy at the University of Arizona, Tucson. He is a recognized expert on space issues – both civil and military. Dr. Worden has authored or co-authored more than 150 scientific papers in astrophysics, space sciences, and strategic studies. He served as a scientific co-investigator for two NASA space science missions, and received the NASA Outstanding Leadership Medal for the 1994 Clementine mission. He has been named the 2009 Federal Laboratory Consortium Laboratory Director of the Year.