Reduction of Martian Sample Return Mission Launch Mass with Solar Sail Propulsion

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Solar sails have the potential to provide mass and cost savings for spacecraft traveling within the inner solar system. Companies like L'Garde have demonstrated sail manufacturability and various in-space deployment methods. The purpose of this study was to evaluate a current Mars sample return architecture and to determine how cost and mass would be reduced by incorporating a solar sail propulsion system. The team validated the design proposed by L'Garde, and scaled the design based on a trajectory analysis. Using the solar sail design reduced the required mass, eliminating one of the three launches required in the original architecture.

Acronyms

ACO	Advanced Concepts Office
ACS	Attitude Control System
AR&C	Arrival, Rendezvous & Capture
C&DH	Command & Data Handling
CP1	Colorless Polymer 1
EEV	Earth Entry Vehicle
ERS	Earth Return Stage

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GN&C	Guidance Navigation and Control
GR&A	Ground Rules and Assumptions
GSD	Ground System Demonstration
HGA	High Gain Antenna
IEM	Integrated Electronics Module
Isp	Specific Impulse
JPL	Jet Propulsion Laboratory
LEO	Low Earth Orbit
LGA	Low Gain Antenna
MEL	Master Equipment List
MIDAS	Mission Design and Analysis Software
MLI	Multi-Layer Insulation
MOI	Mars Orbit Insertion
MPDM	Multi-Purpose Data Module
MPS	Main Propulsion System
MSFC	Marshall Space Flight Center
MSR	Mars Sample Return
NASA	National Aeronautics and Space Administration
ORU	Orbital Replacement Unit
OS	Orbiting Sample
PDU	Power Distribution Unit
PMAD	Power Management and Distribution
SBC	Single Board Computer
SRO	Sample Return Orbiter
SSPA	Solid State Power Amplifier
TRL	Technology Readiness Level
UHF	Ultra High Frequency

I. Introduction

Utilizing solar sails instead of traditional chemical propulsion can provide mass and cost savings for interplanetary travel within the inner solar system [1]. L'Garde and other private ventures have proven both the manufacturability of solar sail materials and various in-space deployment methods. This study involved the incorporation of an existing design of a solar sail for a Solar Sail Technology Demonstration Mission [2] by L'Garde called Sunjammer [3]. The sail design would be applied to the propulsion system of an existing Mars Exploration Sample Return Mission Architecture [4,5,6] developed by the Jet Propulsion Laboratory (JPL). The solar sail is currently in manufacturing development phases in support of NASA's Office of the Chief Technologist Advanced In-Space Propulsion Systems [7]. The purpose of this study was to estimate the reduction of mission cost and mass for an existing architecture through the inclusion of the solar sail.

Current Mars Sample Return Project Architecture

The point of departure for this study was the Mars Exploration Sample Return (MSR) Mission Architecture [4,5,6] which requires separate launches of an exploration rover, an orbiter with an Earth return stage, and a Martian ascent vehicle with a sample return canister. The current launch manifest spans six years with operations lasting an additional 4 years and costs totaling billions of dollars.

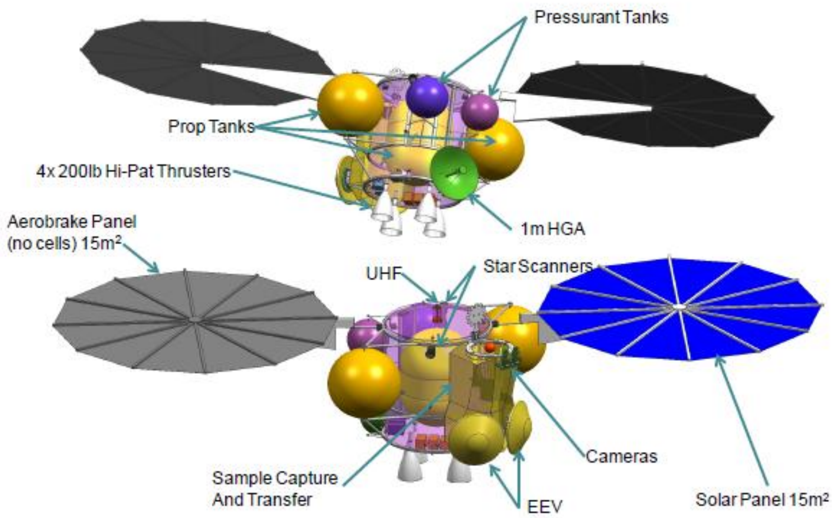

Figure 1 Sample Return Orbiter Design, Credit NASA JPL

The Sample Return Orbiter (SRO), Figure 1, consists of a spacecraft bus capable of relaying communications from Earth to the exploration rover on Mars and vice versa. A Sample Capture and Transfer canister attached to an Earth Entry Vehicle (EEV) retrieves the sample canister launched on the ascent vehicle in orbit and returns back to Earth. The amount of chemical propellant required to send the orbiter to Mars, orbit during surface operations, rendezvous and capture the Martian soil sample, and return back to Mars eliminates the option of ridesharing with any of the other two elements. The first step in the study looked into replacing the required chemical propellant for Earth return with a solar sail propulsion unit.

II. Ground Rules and Assumptions

To reduce the amount of mass the solar sail would need to propel, the SRO was split into two vehicles: Orbiter and the Earth Return Stage (ERS). The orbiter would support the primary mission of getting to Mars, performing relay communications, and assist in rendezvous and capture of the sample. The ERS would separate from the Orbiter after the sample has been se To reduce the amount of mass the solar sail would need to propel, the SRO was split into two vehicles: Orbiter and the Earth Return Stage (ERS). The orbiter would support the primary mission of getting to Mars, performing relay communications, and assist in rendezvous and capture of the sample. The ERS would separate from the Orbiter after the sample has been secured, deploy the sail, and spiral out from Martian orbit to travel back to Earth. The basic ground rules and assumptions are listed in Figures 2 and 3 for the Orbiter and ERS.

Orbiter	General GR&A	Notes
Orbiter Mission	1. Provide telecommunication relay operations for the MSR lander, fetch rover, and MAV 2. Perform orbital sample (OS) rendezvous and capture 3. Deploy solar sail on return vehicle and perform separation of return vehicle 4. AR&C sensors and OS capture mechanism will be contained on the ERS	Phase 3 of mission ops is all new for the solar sail return proposal.
Risk Class: Fault Tolerance	Risk Class C: Single fault tolerance for critical systems	
Orbital Debris and Disposal	Spacecraft will comply with appropriate sections of NASA-STD-8719.14	
Spacecraft Orbital and Cruise Attitude	No particular attitude assumed	
Rover and Lander Restrictions	The landing locations of each of the science elements should be where the orbiter can support them	
Number of launches	Reduce number of launches from 3 to 2	Current MSR mission architecture calls for three launches

Figure 2 Ground Rules and Assumptions for the Orbiter

Earth Return Stage	General GR&A	Notes
ERS Mission	1. Perform a Mars spiral departure and Earth direct entry 2. Sails will be inflated before separation 3. Separation will be performed by the orbiter 4. AR&C sensors and OS capture mechanism will be contained on the ERS	
Risk Class: Fault Tolerance	Risk Class C: Single fault tolerance for critical systems	
Orbital Debris and Disposal	Spacecraft will comply with appropriate sections of NASA-STD-8719.14	Controlled disposal at Earth
Mass Minimization	Minimize Earth return mass by utilizing solar sail and staging (leaving as much mass as possible in Mars orbit)	
Spacecraft Orbital and Cruise Attitude	Sun pointing	

Figure 3 Ground Rules and Assumptions for Earth Return Stage

III. Orbiter Chemical Propulsion Mission Analysis

To maintain consistency with the JPL MSR Architecture, the transit to Mars in this study was performed by a traditional bi-propellant chemical propulsion stage. MIDAS (MIssion Design and Analysis Software) developed by NASA JPL was used to optimize Earth-to-Mars trajectories over a date range of 2022 to 2035 .

Earth Departure			Outbound		At Mars		
Departure Date	C3 (km^2/s^2)	Launch Declination (deg)	TOF (days)	HCA (deg)	Arrival Date	Vinf (km/s)	MOI ΔV (km/s)
08/30/2022	15.151	3.319	347.024	212.10	08/12/2023	2.604	0.877
09/30/2024	11.369	14.915	332.696	209.17	08/29/2025	2.445	0.804
10/31/2026	9.229	28.746	311.054	205.71	09/07/2027	2.571	0.861
11/23/2028	9.116	28.696	300.113	212.83	09/19/2029	2.970	1.059
12/24/2030	*10.794*	*14.994*	*283.084*	*216.15*	*10/03/2031*	*3.480*	*1.342*
04/17/2033	*9.086*	***-55.268***	*199.722*	*148.44*	*11/02/2033*	*3.317*	*1.248*
02/21/2033	*13.646*	*7.369*	*260.000*	*206.40*	*11/08/2033*	*3.934*	***1.617***
06/27/2035	10.372	8.822	201.889	148.63	01/15/2036	2.630	0.889

Figure 4 MIDAS generated optimal Earth-to-Mars Trajectories

The arrival delta-V at Mars was minimized while keeping the Earth-departure C3s between about 15 – 20 km^2/s^2. At Mars, the spacecraft must perform the Mars orbit insertion (MOI) maneuver into a 1-day Sol orbit. Afterwards, the periapsis must be lowered into the Martian atmosphere, so that aerobraking can begin. Aerobraking lowers the apoapsis altitude from 33,810 km to 500 km. Then, a “walk-out” maneuver is performed to raise the periapsis out of the atmosphere and to an altitude of 500 km. Finally, the spacecraft must rendezvous with and capture the sample canister, which will be launched from the Martian surface.

For propellant computations, an MOI delta-V of 1.1 km/s is assumed. This value was previously used by JPL and is assumed to contain gravity losses. This assumption means that the launch dates in 2030 and 2033 must be avoided.

To obtain the propellant requirement, an initial mass of 2110kg was assumed based on initial study team mass estimations. Removing the launch vehicle adapter from the initial mass estimate brings the mass to a starting value of 2080 kg. The propulsion system was assumed to be a dual-mode system with hydrazine and NTO propellants. Large burns are performed by the main propulsion system (MPS), which is a bipropellant system with a specific impulse (Isp) of 326s. The smaller maneuvers are performed by the smaller, monopropellant attitude control system (ACS) thrusters, which have an Isp of 209s. The ACS engines also perform attitude control during the MPS maneuvers.

No sample return delta-V was given in the JPL design reference documentation. A value was reverse-engineered such that a launch mass of 3270kg generated a total propellant load and burnout mass consistent with their design. This resulted in a sample rendezvous (which also contains margin) delta-V of 250 m/s.

Using these assumptions, the required MPS maneuver propellant is computed to be 440.2kg, and the required ACS propellant is 9.3kg.

IV. Solar Sail Mission Analysis

Our proposed solar sail has a heritage with the L'Garde 20-meter on a side Ground System Demonstration (GSD) hardware that was developed and successfully vacuum deployed during the 2003-2006 timeframe at the Plumb Brook facility at NASA Glenn Research Center. This design had heritage from a JPL, NOAA and Air Force Space Technology 5 (ST-5) proposal to the NASA New Millennium Program. To demonstrate scalability, the GSD 20-meter system was the central section of a larger 100-meter on a side solar sail, and the mass estimates for the larger sail were derived from the 20-meter system. This hardware is essentially the 38-meter on a side SunJammer flight demonstration that has been selected by the NASA Office of Chief Technologist and will launch in 2014.

Mars Spiral Out

Once the Orbiting Sample (OS) has been captured and the sail has deployed in Martian orbit, the ERS will spiral out from Mars before performing a Mars-to-Earth transfer. The time of flight calculated to perform the Mars Spiral Out was 435 days assuming a characteristic acceleration of 0.50 mm/s^2 starting from a 500km Mars circular orbit. The sail area was assumed to be 18,887m^2 propelling a payload mass of 200kg which includes the sample. The total sail mass system was 115kg utilizing a 2.5µm Colorless Polymer 1 (CP1).

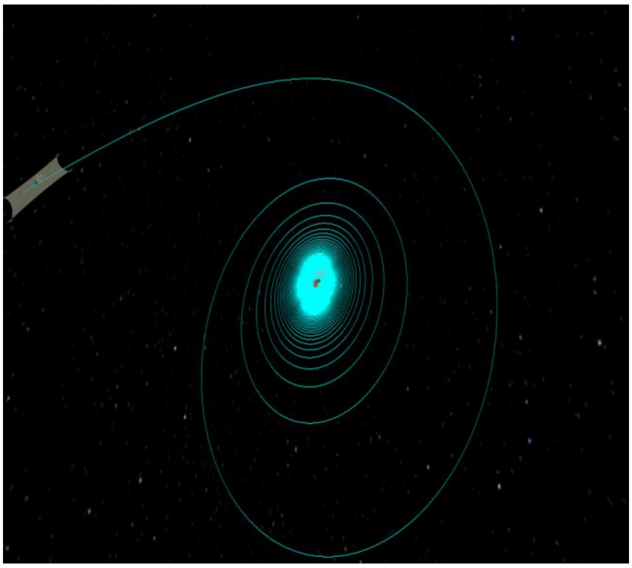

Figure 5 ERS Spiral Out from Mars

Mars-to-Earth Transfer

After an estimated time of 435 days, the ERS will begin the Mars-to-Earth Transfer. Assuming the 0.50mm/s^2 characteristic acceleration, the ERS will leave from Mars perihelion to achieve an Earth entry velocity of 12.8km/s^2 after 507.5 days.

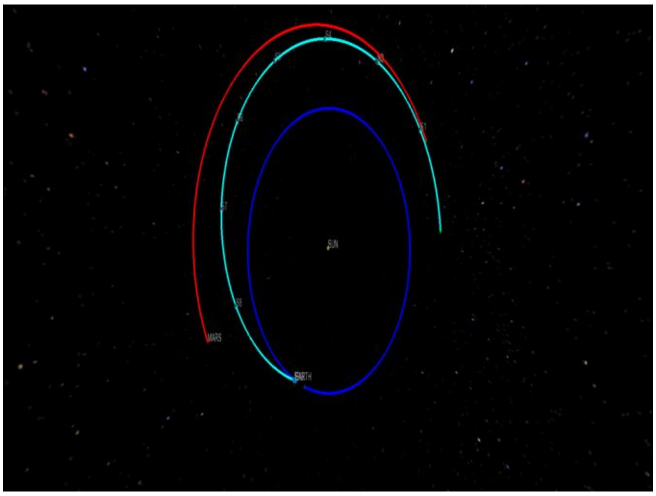

Figure 6 Mars-to-Earth Transfer of 507.5 days

V. Configurations

The Orbiter and ERS launch as one spacecraft and separate after Arrival, Rendezvous & Capture (AR&C) of the OS. The Orbiter will perform primary operations during launch, Earth-to-Mars transit, MOI, and Mars operations. The ERS will be kept in "keep alive" mode during that timeframe. The following section will outline the configurations of the Orbiter and ERS.

Orbiter

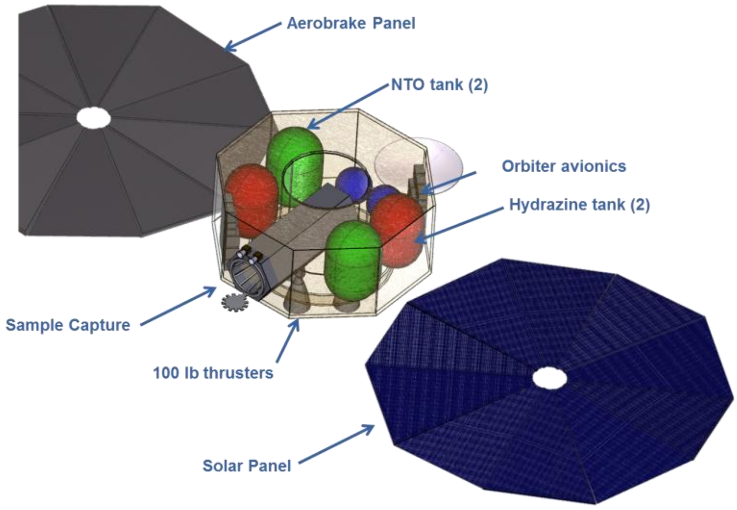

Figure 7 Orbiter Configuration

The configuration of the orbiter, Figure 7, consists of an octagonal bus. The orbiter bus holds all of the orbiter systems. It holds four bi-propellant tanks which feed the main propulsion and reaction control propulsion system. The sample capture system is integrated such that the EEV interface is in the central core. The orbiter avionics are attached to the outer structural walls which also provides the required thermal radiating area. The orbiter also has one 6m diameter solar array panel and a dummy solar array panel that is used as an aerobrake panel.

Earth Return Stage

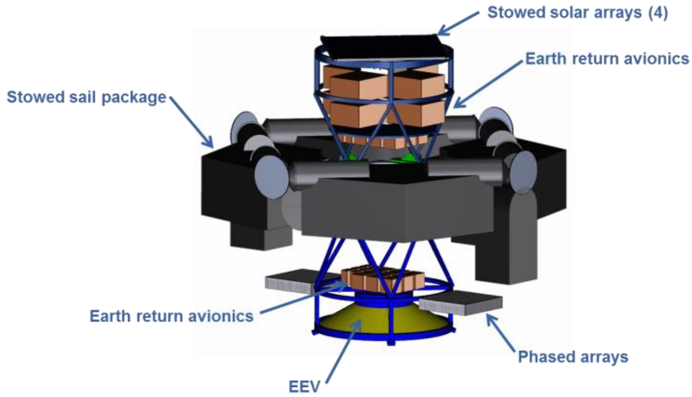

Figure 8 Earth Return Stage (ERS), Sail Stowed

The ERS is comprised of the L'Garde sail package and some new structure that holds the ERS avionics and the EEV capsule. It also has supporting solar arrays and communication phased arrays. See Figure 9. The sail/ERS mates to the top deck of the orbiter. The spacing of the two is set such that the sail package sits on the top deck just as the EEV capsule mates to the sample capture assembly. See Figure 9 for details.

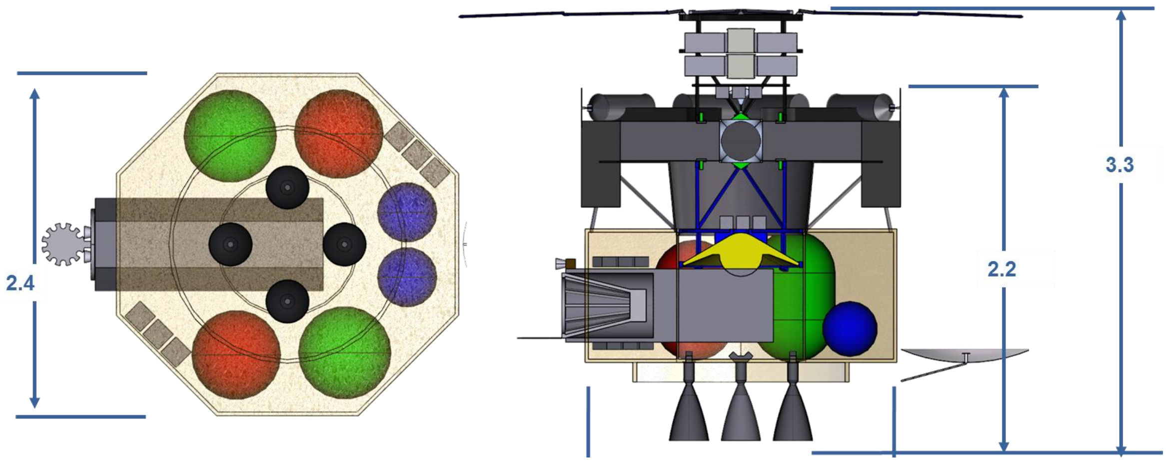

Figure 9 Orbiter & ERS launch configuration and dimensions

The total height of the ERS and Orbiter is about 3.3 m tall and 2.4 m across the base. The Orbiter/ERS would be launched as a dual manifest with the Lander/Rover. The two would fit on a volume basis but may not fit based on mass and launch vehicle performance.

VI. Spacecraft Structure Analysis

Orbiter Structures

All of the sail, boom and beam mass and approximate dimensions were taken from a previous L'Garde MSR study or extrapolated from study or experimental data. These masses were placed in the FEMAP model as mess elements. The sail main support, inflation tanks and the sail retainment doors were not sized; L'Garde's mass was used. The sail retainment doors as well as thermal treatments and much of the power and avionics mass was "smeared" over some structural panels as "non-structural mass" to account for all of the predicted mass.

The FEMAP model was modeled using Al 2024 with 1/8th" thick panels. Support beams below the sail carrier were circular Al tubing with 1/8th" wall thickness. Hypersizer sized most panels to the minimum and would have sized them thinner, but all panel thicknesses were limited to a minimum of .060 inch. See Figure 10.

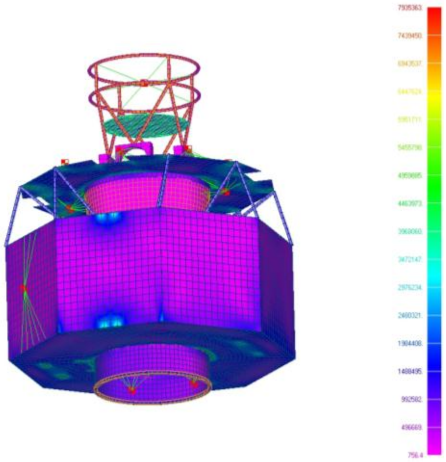

Figure 11 Orbiter FEMAP Model

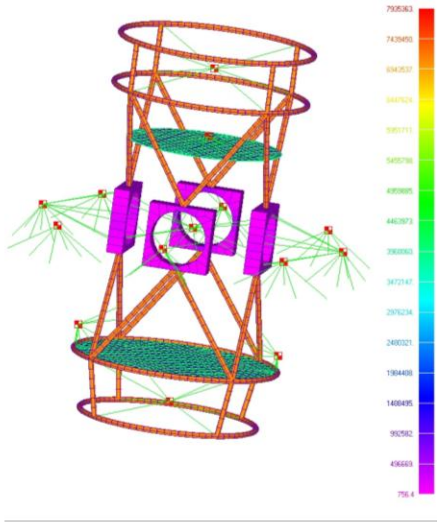

Figure 10 ERS FEMAP Model

ERS Structures

The combined vehicle configuration, the SRO and the ERS were subjected to body loads equal to 6 g's axially and 2 g's laterally at 45 degree intervals radially. The Al structure was meshed and showed a very high Margin of Safety (MOS). When the FEA results were fed into Hypersizer, panel and beam masses were reduced by over 80%. Minimum mass was limited in Hypersizer sizing to select reasonable raw material for manufacturing and provide thick enough panels to conduct heat away from mounted electronics.

FEMAP model versus Pro-E depiction of ERS un-deployed configuration. The ERS structure was completely Al tubing with a couple of $1/8^{th}$" thick round avionics panels. These small ERS panels were maintained at $1/8^{th}$" thick to facilitate thermal rejection. See Figure 11.

VII. Solar Sail Structures

Sail Structures for the study breakdown into three sections: boom design, beam design, and the net/membrane sail design.

Boom Design

The booms are designed in a semi-monocoque configuration. High modulus longitudinal fibers impregnated with a Sub Tg resin are used to rigidize the structure after deployment and are oriented to absorb the compressive loads in the booms, while the lateral fibers absorb the inflation loads and stabilize the longitudinal fibers and the cross section. The external layer doubles as a bladder and shear web.

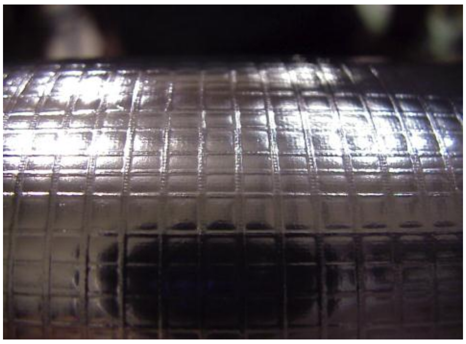
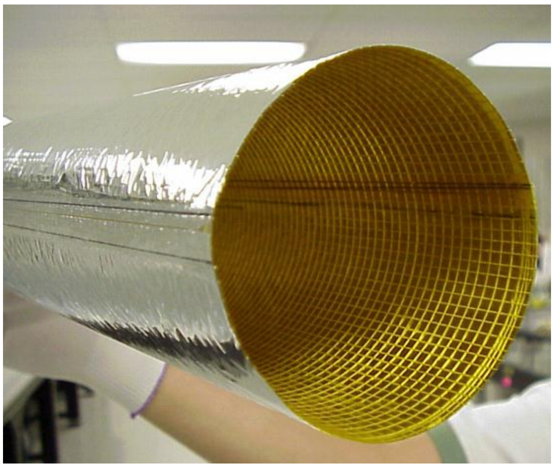

Figure 12 L'Garde Booms

Sub Tg or cold rigidization takes advantage of the increase in modulus of certain materials below their glass transition temperature (Tg). Sub Tg rigidizable structures are simple, reliable and are completely passive and in general require no heaters or vents. Multilayer insulation (MLI) will be used to mitigate the effects of on-orbit thermal gradients and to lower overall boom temperature, adding margin to sub Tg rigidization. Thin heater lines are packaged along with the boom material when stowed to keep the boom warm enough to be flexible, and double as actuator control lines during the mission. They deploy concurrently with the boom providing a heat source until the deployment sequence is complete and then will provide power and control the tip vanes at the ends of the booms.

Beam Design

The beam is composed of a boom and spreader system. The spreader system is composed of rings, spreader bars and tension lines. Low mass rigid rings are mounted periodically along the boom. They serve to stabilize the boom cross-section, provide a hard point for mounting of the spreader bars and also provide an ideal point for mounting of the net/membrane lines. Additionally the spreader system consists of a number of tension elements running along the boom. The anchor lines running from the spreader bar tips to the rings provide a shear component transmitting bending loads. The longeron lines running from the boom tip, along the spreader bar tips provides a large sectional moment of inertia to the truss to resist the bending generated by the solar flux. Finally the diagonals crossing over along alternating spreader bar tips increase the torsional stiffness of the truss while providing longitudinal redundancy in the lines in case one or more lines are severed. See Figure 13.

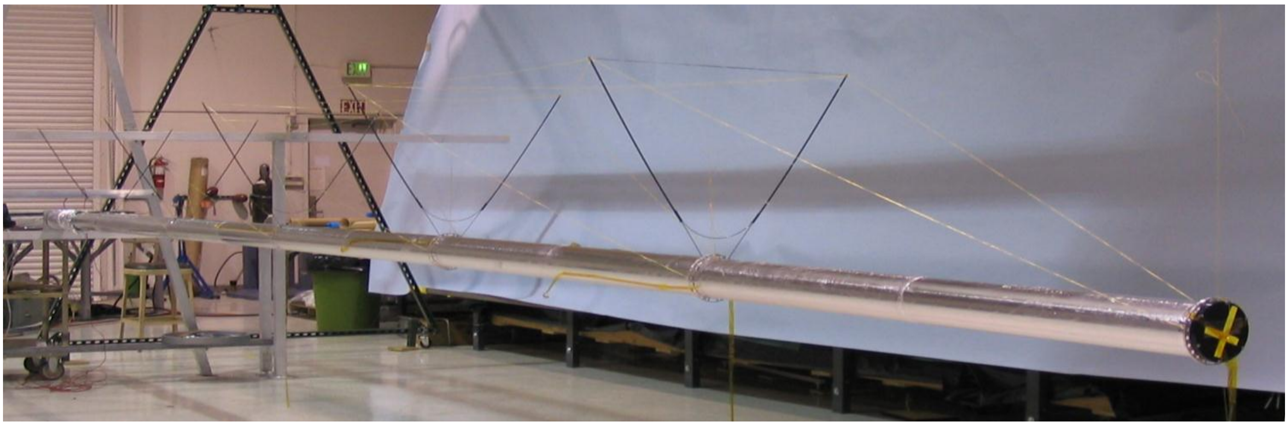

Figure 13 Deployed boom with spreader bars

Sail Net Membrane Design

The sail membranes are fabricated from easily acquired and metalized 2.5µm CP1 material. Rip stop fibers, used to halt rips across the membrane, are bonded to the sail in a 1 m grid pattern. Grounding straps are incorporated to provide a conductive path between the front and back surfaces. Also included are the net chords that define the sail shape and provide the attachment points to the rings on a quadrant beams. The wrinkles in the radial direction are formed by a small amount of extra material specifically designed to absorb any lateral deformations in the film due to thermal effects.

Sail stress loads are transferred through the sail material to the nearest net element that then transfer the load to the structure. Utilizing this method there are minimal stress concentrations in the sail and attach points. Low stress concentrations in the sail membrane have a host of benefits. Small holes and minor damage are far less likely to propagate and tear. Degradation in the film due to environmental effects such as particle radiation and UV are similarly less likely to result in a failure. See Figure14.

Figure 14 L'Garde solar sail membrane with netting

Figure 15 depicts the deployment of the L'Garde Solar Sail. For deployment the sail attitude will be normal to the sun-sail line. The carrier ACS will shut off for deployment so as not to impart loads on the sailcraft structure. Passive stability should keep the sail sun-normal, but in the event some bandwidth is exceeded, the carrier ACS will be used. The sequence initiates with the opening of the canister lids. Next the vanes rotate into the plane of the sail and the vane booms are now deployed bringing with them the vane membranes. The main sail deployment initiates with the deployment of the spreader bars. The main booms are then deployed bringing with them the sail quadrants. All four beams will deploy deploy simultaneously, taking the sail with them. Deployed length is measured and fed back to control relative beam deployment via pressure. After completion of the main sail deployment the canister is ejected to lighten the sailcraft. The final step in the deployment is to deploys the solar arrays for spacecraft power generation.

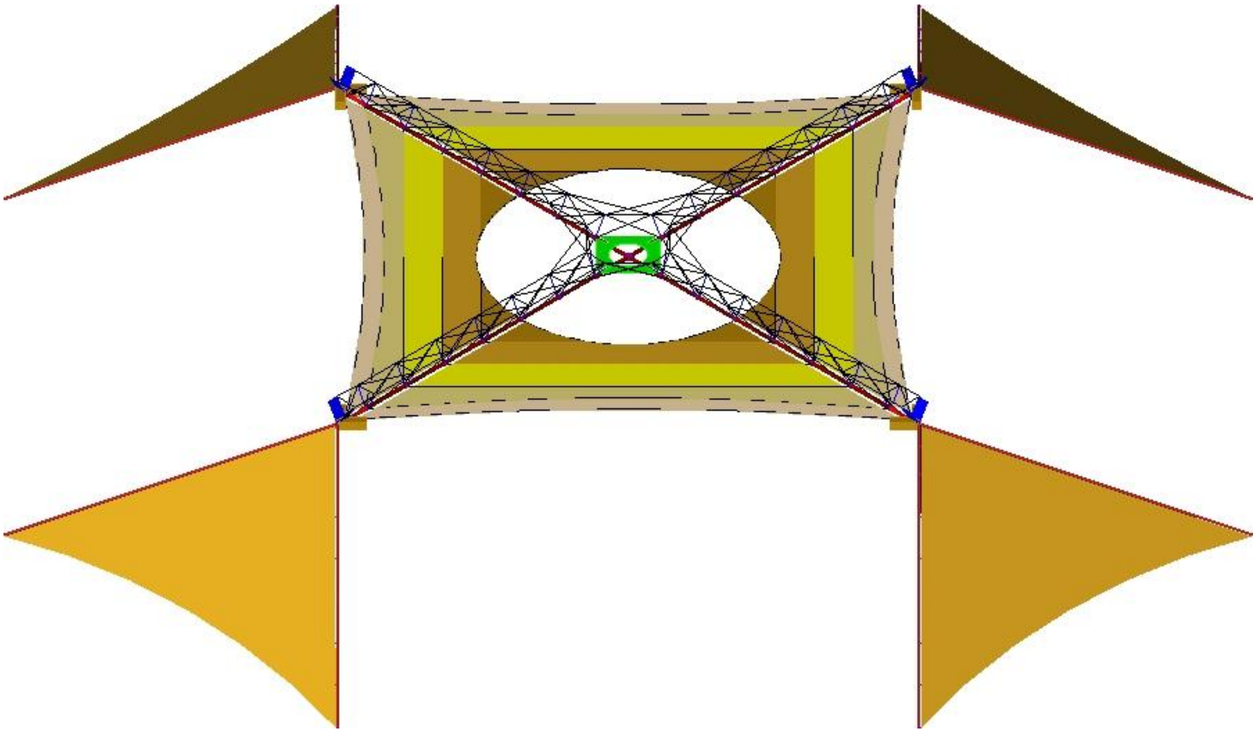

Figure 15 Deployment of the L'Garde Solar Sail and tip vanes

VIII. Power

The ERS and orbiter will have their own solar arrays and power distribution, but the array regulation and battery charge control are common and will return with the ERS. Each stage has its own flight computer and Power Distribution Unit (PDU) which affords redundancy on the trip to Mars. The two stages are connected by a breakaway harness connector which can be separated on command. After separation, the orbiter solar arrays will not work anymore and the batteries can no longer be charged. The batteries will then power the orbiter during its disposal while the ERS deploys its arrays and sails back to Earth. The solar irradiance – 1361 W/m^2 is a minimum value chosen to be conservative at a 1.67 AU solar distance, which is the apophasis of Mars. The 35° incidence angle is taken from the mission analyst's estimate of the maximum angle required to support solar sail propulsion on the return trajectory.

Each power use profile represents a particular set of loads required to be switched 'On' during a portion of the mission. To get the 'design to' power level, we took the maximum power required among all profiles for each stage and add 30% design margin. Martian sun-sync orbit allows ERS to leave massive energy storage batteries behind, resulting in a power system mass of only 40.4 kg.

IX. Avionics

Orbiter C&DH – Approach

The JPL MSR C&DH system was baselined for the Orbiter. All primary functions are contained in a single flight computer, including data acquisition, data processing, flight commands, and data storage. Also included is a communications interface board and an image processing board. A special function board is assumed to handle the AR&C operations. Two exactly alike flight computers are used for single fault tolerance.

The communication system includes a 1 meter parabolic pointing High Gain Antenna (HGA) for direct link to Earth using 34 meter DSN antennas in X-band, and a 15 watt SSPA. This system is used to relay data from the MSR lander and fetch rover back to Earth, and to uplink commands and operation modifications. This same system should be more than sufficient to support any sail deployment and ERS separation status and uplinks.

The Orbiting Sample (OS) may be several hundred kilometers down range from the Orbiter after the Mars ascent. The Orbiter will use the UHF beacon signal to track and maneuver to the OS using the main engines. Once within 10 km of the OS, the Optical Navigation cameras of the capture system will be used to maneuver the Orbiter to within 1 meter. At this point, it's assumed a close proximity lider system, or some similar capable system, guides the Orbiter to capture the OS within the basket without making contact. At this point, the capture door can close and the basket frame work used to dissipate the OS spin momentum.

Orbiter GN&C – Approach

The pointing accuracy required for the Orbiter is driven by the AR&C maneuver fidelity. In the MSR orbiter, 505 arcsec accuracy is specified. However, for most of the mission time this high accuracy is not required. A 2.5 degree low accuracy is assumed sufficient during low accuracy demand times. Typical solar array sun pointing is only 5 degrees. The gimbal HGA can maintain tight Earth pointing independent of the Orbiter's pointing accuracy.

ERS C&DH – Approach

For the ERS C&DH, the Messenger spacecraft Integrated Electronics Module (IEM) system was baselined. Messenger is a Mercury surveillance mission, in which images are transmitted back to earth. Although the data handling capabilities are probably more than what is needed for the ERS, it has proven heritage for reduced risk. Two IEM units are booked on the ERS for single fault tolerance.

For the ERS communications, the Messenger spacecraft communication system was baselined. It uses 11 watt solid state power amplifiers (52 w input) and phased array antennas that give about 24 dBi of gain. This system allows over 2 kbps of data to be transmitted from Mercury to Earth in X-band to 34 meter DSN antennas. Since the distance from Mars to Earth is nearly the same as Mercury to Earth (about 1.5 AU), it can be assumed that this system will work for the ERS at Mars.

Since the ERS will not be transmitting science data, it's assumed that 1 kbps is a sufficient data rate for transmitting navigation information and health and status data. Either smaller arrays or lower power SSPAs can be considered, but to keep the high TRL level the system was unaltered.

Fanbeam antennas are used for a 31.25 bps uplink capability anywhere within the forward looking hemisphere. LGA are included for emergency backup operations.

ERS GN&C - Approach

All the ERS sail attitude and control hardware is included in the sail structures. These components would include sun sensors on the tipvanes, along with the tipvane motors and required electronic interface units and cabling. The use of MSFC Multi-Purpose Data Module (MPDM) for a remote electronic distribution units is suggested to minimized system mass. These units are small and lightweight, about 100 grams each. It's believed one unit in each tip vane could support 3 sun sensors along with controlling and monitoring the tipvane motors. A single power cable and data bus cable would run the length of the booms to these MPDM units at the tipvanes. The power cable is thought to dual as a heater cable during deployment. An appropriate switched power circuit would need to be included in the MPDMs for this operation.

The same MPDMs are suggested for use by the ERS solar arrays for monitoring those sun sensors. Low accuracy AeroAstro star trackers (0.1 deg) and Northrop LN200 IMUs are suggested for navigation of the ERS since no high accuracy sun pointing is required (+/- 2.5 deg from the 35 degree sun angle pointing accuracy required). The EEV retention and pyrotechnic release mechanism control is assumed in the secondary structures.

An observation camera is included in the ERS. This camera is for monitoring the deployment of the sail and the ERS separation from the Orbiter. It can also be used to monitor the deployment of the EEV for earth entry. The IKAROS mission used deployment cameras for visual observation.

X. Thermal Control

Orbiter Analysis

A passive thermal design concept was developed for the Orbiter and ERS. Thermal control will utilize components including multilayer insulation, high emissivity paint and coatings, heatpipes, heaters, etc. to maintain spacecraft subsystem components within acceptable temperature ranges. There are no dedicated radiators, however spacecraft structural panels act to dissipate avionics heat by conduction and also act as radiative surfaces. The orbiter bus outer surfaces are covered in low absorptivity materials in order to cold bias the structure. Propellant tanks located in the orbiter are wrapped In MLI (multi-layer insulation) and propellant tank heaters are sized for a solar synchronous orientation. RCS thrusters, antennas, solar arrays and solar array mechanisms are not part of the analysis.

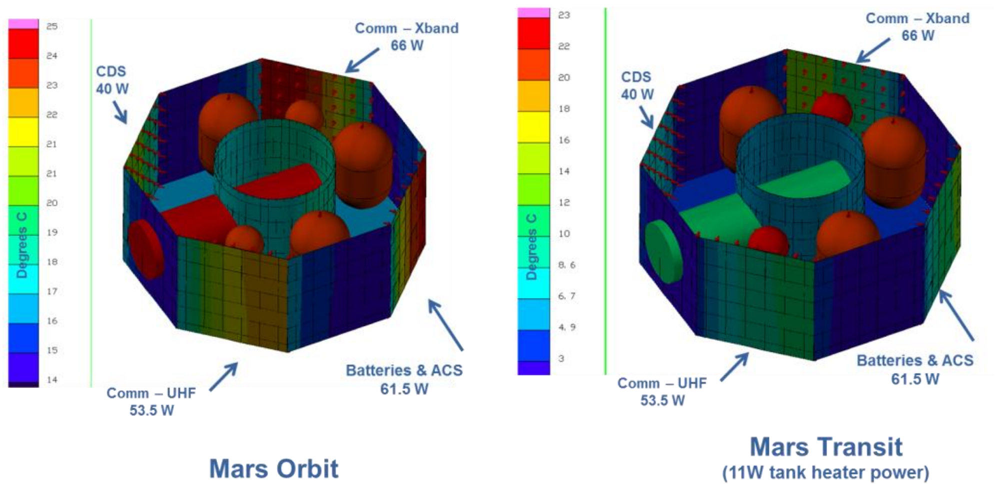

Figure 16 Steady state Thermal Desktop® model of orbiter bus developed to assess subsystems interface temperatures.

A system level thermal model of the spacecraft orbiter bus structure was developed using Cullimore and Ring Technologies' Thermal Desktop® (www.crtech.com) to assess support structure and subsystem equipment interface temperatures. The structure is modeled as aluminum and the panel thickness is consistent with the structural design. Environmental heat loads were calculated for a Mars orbit altitude of 450 km. Mars transit environmental heat loads were calculated at 500,000 km from Mars. Steady state spacecraft structure temperatures for both Mars orbit and transit to Mars were generated in order to determine a range of temperatures to be expected during the course of the flight for the orbiter bus. Steady state temperature profiles for the Orbiter is shown in Figure 16. Heat loads were distributed on the spacecraft panels according to box locations, as shown. Structure temperatures range from 14 °C to 25 °C for the bus during orbit about Mars and 3 °C to 23 °C for the bus during transit to Mars. Additionally 11 watts of heater power is required for the propellant and pressurant tanks for the Mars transit case. Environmental heat loads are averaged about an orbital period for the Mars orbiting case and a single position at an altitude of 500,000 km from Mars was used for the Mars transit case.

	Operational Heat Dissipation (Return to Earth)	Keep Alive Heat Dissipation (Transit to Mars & Mars Orbit)	Operating Temp Range
ACS (IMU Assembly)	24.0 W	-	- 40 C to 85 C
Command & Data System	52 W	26.0 W	- 40 C to 85 C
Avionics Heaters	-	36.0 W	n/a
Power Systems	70 W	70 W	- 40 C to 85 C
Communications (X-band)	131 W	-	- 20 C to 70 C
Total	277 W	132 W	

Table 1 Subsystem equipment heat loads on the Orbiter

Subsystems equipment heat loads used as input to the orbiter thermal analysis are shown in Table 1. A total of 232 W of spacecraft power/heat dissipation was considered in the orbiter thermal analysis, this includes 11 watts of heater power for the propellant tanks. All heat loads are imposed directly on the structure and modeled as area averaged heat loads.

ERS Analysis

The Earth Return Stage is designed with multiple avionics shelves to contain the subsystems equipment. For purposes of the preliminary model, the equipment heat load is equally divided between the upper avionics shelf and the lower shelf near the sample return mechanism. The ERS structure is wrapped with multilayer insulation. ERS thermal analysis results show that during the operational phase of the mission the avionics shelves maintain an average temperature of between 35 and 45 degrees C. The total heat load of 277 watts for the ERS during the operational mission phase is divided between the forward and aft avionics shelves. The operational phase is defined as the transit from Mars orbit to Low Earth Orbit. During the trip from Earth to Mars and during most of the Mars Orbit period, the ERS is maintained in a keep-alive mode. A reduction in avionics power during this period results in an average avionics shelf temperature of between -16 and -6 degrees C. A total head load of 132 watts was divided between the forward and aft avionics shelves.

A summary of the heat dissipation and operating temperature range for the subsystems avionics and power systems equipment contained on the ERS is shown in Table2 for both the operational and keep alive mission phases. The keep alive mission phase required 36 watts of heater power to maintain interface temperatures above -16 degrees C. All predicted interface temperatures are within acceptable range of -16 to 45 degrees C for the mission phases analyzed.

	Operational Heat Dissipation (Return to Earth)	Keep Alive Heat Dissipation (Transit to Mars & Mars Orbit)	Operating Temp Range
ACS (IMU Assembly)	24.0 W	-	- 40 C to 85 C
Command & Data System	52 W	26.0 W	- 40 C to 85 C
Avionics Heaters	-	36.0 W	n/a
Power Systems	70 W	70 W	- 40 C to 85 C
Communications (X-band)	131 W	-	- 20 C to 70 C
Total	277 W	132 W	

Table 2 Heat dissipation and operating temperature ranges for ERS

Optical properties used for the thermal model surfaces are itemized in the table and were taken from various sources including Spacecraft Thermal Control Handbook, NASA Spacecraft Thermal Coatings Reference, and vendor data. The spacecraft bus and ERS internal surfaces are assumed to be black annodized to optimize radiative exchange within the enclosure. Silverized Teflon is used as an outer layer on the MLI blankets covering both the orbiter bus and ERS stages. Degraded optical properties were used due to the extensive length of the mission.

Total thermal control mass for the configuration is estimated at 50.65 kg which includes a 30% margin. Based on the analysis, all interface temperatures are within acceptable range for the defined orientation and mission phase. Thermal management is accomplished with typical, flight proven components and no technology development is required.

XI. Mass Properties

The Orbiter basic Dry Mass total is 495 kg and is a rollup of the following subsystem masses: Structures, Chemical Propulsion, Thermal, Avionics, and Power. The predicted Dry Mass increases to 591 kg once a 19% Contingency (96 kg) is added to the basic Dry Mass.

After adding non-propellant fluids, payload, and maneuver propellant to the basic Dry Mass, the Orbiter Total Mass increases to 1,450 kg. This total increases to 1,591 kg once contingency is added. See Table 3 for details.

		Basic Mass (kg)	Contingency (%)	Contingency (kg)	Predicted Mass (kg)
1.0	Structures	139.45	23%	32.33	171.78
2.0	Chemical Propulsion	117.67	25%	29.42	147.09
3.0	Thermal	22.14	30%	6.64	28.78
4.0	Avionics	109.41	13%	14.44	123.85
5.0	Power	106.60	12%	13.22	119.82
Dry Mass		**495.27**	**19%**	**96.04**	**591.31**
6.0	Non-Propellant Fluids	26.46			26.46
7.0	Payload	149.00	30%	44.70	193.70
Inert Mass		175.46			220.16
Total Less Propellant		670.73			811.47
8.0	Maneuver Propellant	779.20			779.20
Total Orbiter Mass		**1449.93**			**1590.67**

Table 3 Total Orbiter Launch Mass

The Earth Return Stage (ERS) basic Dry Mass total is 268 kg and is a rollup of the following subsystem masses: Structures, Solar Sail (Propulsion), Thermal, Avionics, and Power.

The predicted Dry Mass consists of the basic Dry Mass total and the contingency (margin) added by each subsystem, which averages 14% (36 kg); this increases the Dry Mass total to 305 kg.

The ERS basic Dry Mass increases to 320 kg once the payload (52 kg) is added. After contingency is added, the Total ERS mass increases to 357 kg. See Table 4.

		Basic Mass (kg)	Contingency (%)	Contingency (kg)	Predicted Mass (kg)
1.0	Structures	38.02	26%	10.05	48.07
2.0	Solar Sail	125.08	12%	14.51	139.59
3.0	Thermal	16.82	30%	5.05	21.87
4.0	Avionics	51.81	6%	3.12	54.94
5.0	Power	36.70	10%	3.60	40.30
Dry Mass Total		**268.43**	**14%**	**36.33**	**304.77**
6.0	Payload	52.05			52.05
Total ERS Mass		**320.48**			**356.82**

Table 4 Total ERS Mass + OS Payload

The total launch mass is 1,948 kg and includes the Orbiter Mass 1,591 (kg) and the ERS Mass (357 kg). See Table 5 for details.

	Basic Mass (kg)	Predicted Mass (kg)
Orbiter Mass	1449.93	1590.67
ERS Mass	320.48	356.82
Total Launch Mass	**1770.41**	**1947.49**

Table 5 Total Launch Mass of the combined Orbiter and ERS

XII. Study Conclusion

The primary objective was to reduce the baseline mission from three to two launches. The study was required to maintain the baseline mission objectives for all three elements, only redesigning the Earth return propulsion system. The Advanced Concepts Office MSR Orbiter and ERS had a launch mass too large to be packaged on the Max-C Rover or Lander launch vehicles if the design only replaced the ERS chemical propulsion system with the solar sail propulsion unit. However, the amount of mass saved was significant. The original architecture design had an Orbiter launch mass of 3270kg. The solar sail propulsion unit reduced the launch mass by 1322.5 kg. See Table 6 below.

	Predicted Launch Mass (kg)	Launch Vehicle	Launch Capability (kg)	Contingency (kg)
Max-C Rover	4457.4	Atlas V 531	4980	522.6
Lander and Ascent Vehicle	4668	Atlas V 551	5130	462
ACO MSR Orbiter and ERS	1947.49	--	--	--

Table 6. Three MSR elements to be launched include the new ACO MSR Orbiter and ERS. The first trade of replacing the ERS propulsion unit with a solar sail greatly reduced the launch mass but not enough to ride share with the Max-C Rover or Lander and Ascent Vehicle.

A second trade, called Delta, consisted of utilizing solar sail propulsion on both trips. If the orbiter used a solar sail as the main propulsion method for transit to and from Mars, the orbiter would be able to launch with the Max-C Rover eliminating the need for one of the three launches. The updated mass numbers are located in Table 7 below.

Delta	Predicted Mass (kg)
Max-C Entry System	1550.7
Descent Stage	1313.1
Pallet	327.5
Max-C Rover	364.5
ACO Delta Orb+ERS	508.8
Total	**4064.6**

Delta Launch Summary	
2028 Atlas V 551 Mass (kg)	5150
Delta Mass (kg)	4064.6
Launch Contingency:	**21%**

Table 7 Updated mass numbers from the Delta trade replacing the entire chemical propulsion unit with a solar sail propulsion unit.

In conclusion, the team was able to feasibly repackage the Orbiter and Earth Return System into a configuration that allows stowage and deployment of the 150 M solar sail system based on the scalable L'Garde design. The team concluded that the addition of the solar sail for the trip to Mars and back to Earth eliminated the need for one of three planned launches therefore reducing the mission operations timeline and launch manifest costs. The team estimated that even with the reduced mass, all mission requirements would be met. Future work would involve re-evaluating launch vehicle options and further design modifications to the spacecraft to further reduce mass.

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