DEPLOYTECH

Mars Sample Return Concept Study
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<th>Discipline</th>
<th>Team Member</th>
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<tr>
<td>Principle Investigator</td>
<td>Les Johnson (ED04)</td>
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<td>Study Leads</td>
<td>Tiffany Russell &amp; Scott Thomas (ED04)</td>
</tr>
<tr>
<td>Mission Analysis</td>
<td>Andrew Heaton (EV42)</td>
</tr>
<tr>
<td>GR&amp;A, Mass Properties</td>
<td>Dauphne Maples (ED04)</td>
</tr>
<tr>
<td>Solar Sail Propulsion</td>
<td>Roy Young (ES11)</td>
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<tr>
<td>Chemical Propulsion</td>
<td>Dan Thomas (ED04)</td>
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<tr>
<td>Avionics, Comm, &amp; GN&amp;C</td>
<td>Pete Capizzo (ED04)</td>
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<td>Power &amp; Risk Mitigation</td>
<td>Leo Fabisinski (ED04)</td>
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<tr>
<td>Thermal</td>
<td>Linda Hornsby (ED04)</td>
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<td>Spacecraft Structures</td>
<td>Janie Miernik (ED04)</td>
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<tr>
<td>Solar Sail Structures</td>
<td>Andrew Wayne (ES21), Mark Davis (ES22), Dean Alhorn (ES32)</td>
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<td>Configuration</td>
<td>Mike Baysinger (ED04)</td>
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## Study Schedule

<table>
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<tr>
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<th>Wednesday</th>
<th>Thursday</th>
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<td>June 4th</td>
<td>Study Kickoff</td>
<td>Background Research</td>
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<td>Background Research</td>
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<td>Ground Rules and Assumptions</td>
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<td>June 18th</td>
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<td>Design Session</td>
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<td>June 25th</td>
<td>Design Session</td>
<td>Design Session</td>
<td>Document</td>
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DEPLOYTECH OVERVIEW
Three year project awarded to Surrey Space Center by the European Commission

Three main objectives for the DEPLOYTECH Project

- Advance technology readiness levels of space deployable technologies
- Develop updated mathematical models to analyze deployable structures
- Develop testing methods and facilities for ground based tests of space deployable structures

Partnership between NASA MSFC and University of Surrey to aid in development of space deployable technologies

Principle Investigators: Les Johnson and Vaios Lappas
Planned design sessions and team meetings
- Two week design session with ED04 Space Systems Team, June 2012
- Monthly tag ups with DEPLOYTECH project

External product deadlines
- DEPLOYTECH Kickoff at University of Surrey, Feb. 21st
- Work Package 1 delivery, September 2012

Internal reviews
- Management Awareness, Spring 2012
- Design session out brief, July 2012
- Final product review, August 2012

Travel plans
- Project representative present during kickoff in February
- Project representatives present during the final presentations in September
Mission Overview

◆ Planetary Science Decadal Survey Mission
  ◆ Three ‘separate’ missions to remain under cost caps
    ● Astrobiology Explorer (Rover)
    ● Sample Return Orbiter (with EEV)
    ● Sample Return Lander (with Ascent Stage)

◆ MSR ERS Modifications
  ◆ Replace Sample Return Orbiter MPS system with solar sail
    ● Maintain EEV from Decadal Survey Mission
    ● Determine solar sail characteristics (size, mass, etc.)
    ● Determine new spacecraft characteristics (size, mass, etc.)
Mission Overview

- Astrobiology Explorer (Rover)
  - Launch = mid 2018
Mission Overview

- Sample Return Orbiter (with EEV)
  - Launch = 2022
Mission Overview

- Sample Return Lander (with Ascent Stage)
  - Launch 2024
GROUND RULES AND ASSUMPTIONS

C. Dauphne Maples
<table>
<thead>
<tr>
<th><strong>Orbiter</strong></th>
<th><strong>General GR&amp;A</strong></th>
<th><strong>Notes</strong></th>
</tr>
</thead>
</table>
| 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. |
<p>| 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 |</p>
<table>
<thead>
<tr>
<th>Earth Return Stage</th>
<th>General GR&amp;A</th>
<th>Notes</th>
</tr>
</thead>
</table>
| 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 |       |
ORBITER MISSION ANALYSIS

Dan Thomas
Assumptions

- Earth departure dates ranging from 2022 – 2035

- Minimum energy transfers
  - Keep departure C3s < 15 – 20 km²/s²
  - Minimize Mars arrival ΔV

- At Mars
  - Mars orbital insertion (MOI) into 250 x 33,810 km (1 Sol)
  - “Walk-in” periapsis (to 125 x 33,810 km)
  - Aerobrake to 125 x 500 km
  - “Walk-out” periapsis (to 500 x 500 km)
  - Rendezvous with sample canister
  - Deploy sail

- Miscellaneous
  - Outbound trajectory correction maneuvers (TCMs): total of 30 m/s
**Earth-to-Mars Trajectories**

**Used MIDAS to generate optimal Earth-to-Mars trajectories**

<table>
<thead>
<tr>
<th>Earth Departure</th>
<th>Outbound</th>
<th>At Mars</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Departure Date</strong></td>
<td>C3 (km²/s²)</td>
<td>Launch Declination (deg)</td>
</tr>
<tr>
<td>08/30/2022</td>
<td>15.151</td>
<td>3.319</td>
</tr>
<tr>
<td>09/30/2024</td>
<td>11.369</td>
<td>14.915</td>
</tr>
<tr>
<td>10/31/2026</td>
<td>9.229</td>
<td>28.746</td>
</tr>
<tr>
<td>11/23/2028</td>
<td>9.116</td>
<td>28.696</td>
</tr>
<tr>
<td>12/24/2030</td>
<td>10.794</td>
<td>14.994</td>
</tr>
<tr>
<td>04/17/2033</td>
<td>9.086</td>
<td>-55.268</td>
</tr>
<tr>
<td>02/21/2033</td>
<td>13.646</td>
<td>7.369</td>
</tr>
<tr>
<td>06/27/2035</td>
<td>10.372</td>
<td>8.822</td>
</tr>
</tbody>
</table>

For now, assumed an MOI ΔV = 1.1 km/s in the propellant estimations, which was used by JPL design. Therefore, orbiter will be unable to accomplish 2030 and 2033 opportunities.
Launch mass (first estimate) ~ 2110 kg

Includes:
- Launch vehicle adapter ~ 30 kg
- Orbiter + Sail burnout ~ 1300 kg
- Orbiter propellant (see below)

<table>
<thead>
<tr>
<th>Event</th>
<th>System</th>
<th>Mass (kg)</th>
<th>$\Delta V$ (m/s)</th>
<th>Isp (s)</th>
<th>Propellant Used (kg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>TCMs</td>
<td>MPS</td>
<td>2080.0</td>
<td>30.0</td>
<td>326</td>
<td>19.4 0.1</td>
</tr>
<tr>
<td>Max. MOI (with gravity losses)</td>
<td>MPS</td>
<td>2060.5</td>
<td>1100.0</td>
<td>326</td>
<td>599.8 3.7</td>
</tr>
<tr>
<td>Walk-in</td>
<td>ACS</td>
<td>1456.9</td>
<td>7.2</td>
<td>209</td>
<td>0.0 5.1</td>
</tr>
<tr>
<td>Aerobrake ACS</td>
<td>ACS</td>
<td>1451.8</td>
<td>9.0</td>
<td>209</td>
<td>0.0 6.4</td>
</tr>
<tr>
<td>Walk-out</td>
<td>MPS</td>
<td>1445.4</td>
<td>84.9</td>
<td>326</td>
<td>37.9 0.2</td>
</tr>
<tr>
<td>Sample Rendezvous and Margin</td>
<td>MPS</td>
<td>1407.3</td>
<td>250.0</td>
<td>326</td>
<td>105.9 0.7</td>
</tr>
</tbody>
</table>

Total Propellants: 763.0 16.2
Future Work

- Need better definition of the responsibilities of the small thrusters.
  - Much of the aerobrake ACS may be performed by off-pulsing main engines.
  - More ACS (and less MPS) propellant may be needed during rendezvous with the sample canister, resulting in a lower NTO/Hydrazine ratio

- May need to increase MOI $\Delta V$ to accomplish all opportunities during the repeat cycle.
ORBITER CHEMICAL PROPULSION

Dan Thomas
Assumptions

- Used existing design philosophy of four large biprop and 16 small monoprop engines

- Dual-mode propulsion system
  - Biprop hydrazine/NTO main propulsion system (MPS)
  - Monoprop hydrazine attitude control system (ACS)
  - Pressure-fed with gaseous helium

- MPS
  - Four Aerojet HiPAT engines
  - Thrust per engine: 445 N (100 lbf)
  - Specific impulse: 326 s
  - Mixture ratio: 0.85
  - Mass: 5.2 kg per engine

- ACS
  - Eight Aerojet MRM-103D rocket engine assemblies
  - Total of 16 thrusters
  - Thrust per engine: 1.02 – 0.22 N (0.23 – 0.050 lbf)
  - Specific impulse: 224 – 209 s
  - Mass: 1.27 kg per module

Input Maneuver Propellant

- MPS: 763.0 kg
- ACS: 16.2 kg
Assumptions (cont.)

◆ Propellant and pressurant storage:
  
  ◆ Fuel
    ● Number of hydrazine tanks: 2
    ● Storage pressure = 1.724 MPa (250 psia)
    ● Storage temperature = 294.3 K (70 °F)
  
  ◆ Oxidizer
    ● Number of NTO tanks: 2
    ● Storage pressure = 1.724 MPa (250 psia)
    ● Storage temperature = 294.3 K (70 °F)
  
  ◆ Pressurant
    ● Number of tanks: 2
    ● Storage pressure = 31.026 MPa (4500 psia)
    ● Storage temperature = 294.3 K (70 °F)

◆ Other assumptions:
  
  ◆ Residuals: 3%
  ◆ Ullage volume: 5%
### Orbiter Propulsion System Masses

<table>
<thead>
<tr>
<th>Item</th>
<th>Qty</th>
<th>Unit Mass (kg)</th>
<th>Basic Mass (kg)</th>
<th>MGA</th>
<th>Predicted Mass (kg)</th>
</tr>
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<tbody>
<tr>
<td>Main Engine</td>
<td>4</td>
<td>5.2</td>
<td>20.8</td>
<td>25%</td>
<td>26.0</td>
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<tr>
<td>ACS Two-thruster Module</td>
<td>8</td>
<td>1.3</td>
<td>10.2</td>
<td>25%</td>
<td>12.7</td>
</tr>
<tr>
<td>Fuel Tanks</td>
<td>2</td>
<td>10.8</td>
<td>21.6</td>
<td>25%</td>
<td>27.0</td>
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<tr>
<td>Oxidizer Tanks</td>
<td>2</td>
<td>6.5</td>
<td>13.0</td>
<td>25%</td>
<td>16.2</td>
</tr>
<tr>
<td>Pressurant Tanks</td>
<td>2</td>
<td>5.5</td>
<td>11.0</td>
<td>25%</td>
<td>13.8</td>
</tr>
<tr>
<td>Propulsion Feed Components (valves, etc.)</td>
<td>1</td>
<td>26.0</td>
<td>26.0</td>
<td>25%</td>
<td>32.5</td>
</tr>
<tr>
<td>Lines and Fittings</td>
<td>1</td>
<td>4.5</td>
<td>4.5</td>
<td>25%</td>
<td>5.6</td>
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<tr>
<td>Structural Mounts</td>
<td>1</td>
<td>10.7</td>
<td>10.7</td>
<td>25%</td>
<td>13.4</td>
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<tr>
<td><strong>Total Dry Mass:</strong></td>
<td></td>
<td></td>
<td><strong>147.1</strong></td>
<td></td>
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</table>

#### Loaded Propellant (with residuals)

<p>| | |</p>
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<tbody>
<tr>
<td>Hydrazine</td>
<td>441.5</td>
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<tr>
<td>NTO</td>
<td>361.1</td>
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<tr>
<td><strong>Total Propellant:</strong></td>
<td><strong>802.6</strong></td>
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#### Pressurant

<p>| | |</p>
<table>
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<tbody>
<tr>
<td>Gaseous Helium</td>
<td>3.1</td>
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</table>

**Total Wet Mass:** **952.8 kg**
Future Work

- Investigate potential savings provided by biprop-only system with MMH/NTO (i.e. replace small monoprop engines with biprop options).
  - Astrium provides small biprop engines with thrust levels down to 4 N (0.9 lbf)
  - Aerojet’s R-6D engine (22 N = 5 lbf) has a low minimum impulse bit and may be an option.
EARTH RETURN SYSTEM MISSION ANALYSIS

Roy Young
Andy Heaton
Myron Fletcher
Solar Sail Propulsion Fundamentals

- Solar sails use photon “pressure” on thin, lightweight reflective sheet to produce thrust; ideal reflection of sunlight from surface produces 9 Newtons/km² at 1 AU
- Net force on solar sail perpendicular to surface
- One component of force always directed radially outward
- Other component of force tangential to orbit (add/subtract \( V_o \))

![Diagram of solar sail propulsion](image)
Solar Sail Propulsion Subsystem is a 150 M on a side sail based on a scale-up of L’Garde estimated 100 M sail calculated during their Ground System Demonstration (GSD) program.

Current 38 M SunJammer flight demo for the Office of Chief Technologist.

20 M GSD built and tested from 2003-2006.

Overall Strategy:
- Concept leverages ST-5 Phase A and Team Encounter Experience.
- Sail membrane, AL coated 2 µm Mylar attached with stripped net.
- Lightweight semi-monocoque boom with sub-tg rigidization.
- 4 Vane thrust vector control.
Load bearing longitudinal uni-directional fibers
- Fibers impregnated with sub-tg resin, rigid below -20° C
- 0.48 AU design requires greater fiber density

Spiral wrap
- Stabilizes longitudinal fibers
- Allows over pressurization for deployment anomalies

Bonded Kapton bladder and Mylar
- Encapsulation “skin” carries shear
- Aircraft fuselage like structure
Beam Design

Beam structure

- Sail structure is stressed for solar loading in one direction for mass efficiency
- Truss system comprised of mostly tension elements, minimal rigid components
- Highly mass efficient, ~36 g/m linear density

7.5m beam stowed to ~.5m

Deployed 7.5m beam
Net/Membrane Sail Design

- **Net Membrane**
  - Sail supported by high modulus and low CTE net
  - Additional membrane materials allows thermal compliance
  - Sail properties effect local billow between net members only
  - Global sail shape is stable

- **Advantages**
  - Net defines the overall sail shape
  - Stability and geometry of the sail is effectively decoupled from membrane properties
  - Sail shape, and hence thrust vector, sailcraft stability and performance are predictable and stable
  - No high local stress concentrations in the sail, loads are transferred through the net, not the membrane
  - Very scalable, larger net/membrane sails simply add additional net elements to control overall shape
Deployment Sequence

- Simultaneous 4-beam deployment
- Measurement system allows automated closed loop control of relative boom positions via pressure

- Carrier ACS inactive
- Telescopic booms kept warm by heater wires
- Sail deployment driven & controlled by telescopic booms

fully deployed sail & beam (boom pressure stabilized)

deploying sail & beam
still-packaged sail & beam

- Once deployed, turn off heater to allow to rigidize
- Vent booms through non-propulsive jets

Outboard stripe wires scallop to assist inboard stripe pullout
Outboard sail edge is permanently scalloped
## Solar Sail Propulsion System Mass

<table>
<thead>
<tr>
<th>Solar Sail</th>
<th>Basic Mass (kg)</th>
<th>Contingency (%)</th>
<th>Predicted Mass (kg)</th>
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</thead>
<tbody>
<tr>
<td>2.0 ERS Solar Sail</td>
<td>125.08</td>
<td>12</td>
<td>139.59</td>
</tr>
<tr>
<td>2.1 CP1 Sail Material, coatings, ripestop, and stripe net</td>
<td>101.81</td>
<td>8</td>
<td>110.21</td>
</tr>
<tr>
<td>2.2 Solar Sail Beam</td>
<td>14.90</td>
<td>26</td>
<td>18.73</td>
</tr>
<tr>
<td>2.3 Attitude Control System. Tip vanes</td>
<td>8.37</td>
<td>27</td>
<td>10.65</td>
</tr>
</tbody>
</table>
Assumptions
- Area = 18,887 m²
- Payload Mass = 200 kg
- Sail Mass = 115 kg
- 2.5 µm CP1

Characteristic Acceleration = 0.50 mm/s²
**Assumptions**

- Area = 18,887 m²
- Payload Mass = 200 kg
- Sail Mass = 115 kg
- 2.5 µm CP1
- Sun Synchronous Orbit
- Start at 500 km Circular orbit
- 93.2 inclination

**Characteristic Acceleration** = 0.50 mm/s²

**Time of Flight is 435 days**
Assumptions

- McInnes local optimal control
- Leaving Mars at perihelion
- Initial orbit Mars orbit

Characteristic Acceleration = 0.50 mm/s²

- Vinf of arrival of 6.4 km/s²
  - Entry velocity of 12.8 km/s²

- Time of Flight is 507.5 days
Forward Work

◆ Mars Spiral-Out
  ◆ Adapt to changing sun-synchronous condition (inclination must change with altitude)
  ◆ Investigate use of sail to rotate orbit plane
  ◆ Check sail aero drag at 500 km

◆ Earth-Mars Transfer
  ◆ Trade initial condition of orbit (start at different location)
  ◆ Fully optimize in S5 (previous published cases arrived at Earth with ~ 0 Vinf)
Forward Work

- **General**
  - Consider incorporating Mars Ascent Vehicle (MAV) into trades
  - Consider higher orbiter start altitude

- **Observations**
  - Solar sail return from Mars can reduce mass of return vehicle at cost in time of flight
  - Valuable lessons learned for other solar sail missions
CONFIGURATION

Mike Baysinger
Orbiter

- Aerobrake Panel
- NTO tank (2)
- Hydrazine tank (2)
- Orbiter avionics
- Sample Capture
- 100 lb thrusters
- Solar Panel

National Aeronautics and Space Administration
ERS

Top structure

Earth return avionics

Boom supports

Lower structure

EEV
Deployed Sail
Atlas Shroud

Orbiter/EEV

Lander/Rover
STRUCTURES

Janie Miernik
Andy Wayne
Mark Davis
Vehicle structure modeled and analyzed with Finite Element Model Analysis and Post-processing (FEMAP) software to produce the Finite Element Analysis (FEA) mesh.

Hypersizer® FEA software optimized the structure for minimum mass.

Load Cases: Atlas V static launch loads:

- Axial: +6, -2g
- Lateral: +/- 2g

Factors of Safety:

- Al 2024 Metallic Material: 1.4

Deployed sail structural analysis not performed.
Launch adapter not included in mass.
Sailcraft structural mass sized by:
- Designing an ERS retaining only essential components and the EEV
- Balancing mass about the plane of the sail
- Sizing panel thickness for thermal properties and manufacturing limitations

Structures using the same mass as the JPL MSR study:
- Sample Capture and Transfer concept and hardware
- Orbiter balanced aerobraking panel
- Solar array, drive, latch/release and boom assemblies
- Antenna gimbal assembly

Structures using the same mass as the L’Garde MSR study:
- Extrapolated sail, boom, beam masses and sail main support.
- Sail deployment mechanisms: inflation tanks and sail retention doors.

Separation systems for ERS and EEV are Planetary Systems Corp (PSC) motorized Lightbands.
**Analysis Results**

- **FEA Results:** Orbiter plus ERS
  - 2 mm translation
  - FEMAP returned very high Margin of Safety (MOS) with 3.175 mm (1/8th in) thick panels
  - Hypersizer sized panels and beams thinner; reducing mass by ~80%
  - All structure aft of and including the sail carrier is jettisoned after sail deployment

![Diagram](image-url)
Analysis Results

* FEAST: ERS
  - Avionics panel/shelves were maintained at 3.175 mm (1/8th in) to aid heat rejection.

FEMAP model versus Pro-E depiction of ERS undeployed configuration
## Mass Summary

### Deploytech Orbiter

<table>
<thead>
<tr>
<th>Item</th>
<th>Basic Mass (kg)</th>
<th>Contingency (%)</th>
<th>Predicted Mass (kg)</th>
</tr>
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<tbody>
<tr>
<td>1.0 Structures</td>
<td>139.45</td>
<td>23</td>
<td>171.78</td>
</tr>
<tr>
<td>1.1 Primary structure</td>
<td>94.00</td>
<td>25</td>
<td>117.50</td>
</tr>
<tr>
<td>1.2 Secondary structures</td>
<td>16.00</td>
<td>30</td>
<td>20.80</td>
</tr>
<tr>
<td>1.3 Solar array/aerobrake booms, drive assys</td>
<td>7.20</td>
<td>25</td>
<td>9.00</td>
</tr>
<tr>
<td>1.4 Aerobrake panel, balanced</td>
<td>15.20</td>
<td>10</td>
<td>16.72</td>
</tr>
<tr>
<td>1.5 ERS Separation adapter, Orbiter side</td>
<td>7.05</td>
<td>10</td>
<td>7.76</td>
</tr>
<tr>
<td><strong>Total Structures</strong></td>
<td><strong>177.47</strong></td>
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<td><strong>219.85</strong></td>
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### Deploytech ERS

<table>
<thead>
<tr>
<th>Item</th>
<th>Basic Mass (kg)</th>
<th>Contingency (%)</th>
<th>Predicted Mass (kg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1.0 Structures</td>
<td>38.02</td>
<td>26</td>
<td>48.07</td>
</tr>
<tr>
<td>1.1 Primary structure</td>
<td>19.00</td>
<td>25</td>
<td>23.75</td>
</tr>
<tr>
<td>1.2 Secondary structures</td>
<td>9.00</td>
<td>30</td>
<td>11.70</td>
</tr>
<tr>
<td>1.3 Solar array booms (4)</td>
<td>8.00</td>
<td>30</td>
<td>10.40</td>
</tr>
<tr>
<td>1.4 ERS Separation adapter</td>
<td>2.02</td>
<td>10</td>
<td>2.22</td>
</tr>
<tr>
<td><strong>Total Structures</strong></td>
<td><strong>177.47</strong></td>
<td></td>
<td><strong>219.85</strong></td>
</tr>
</tbody>
</table>
Conclusions

◆ Structures mass of the orbiter is 18.8% of total dry mass, 11.3% of total/wet mass.
  ◆ The sail carrier and covers are considered orbiter mass as they will jettison with the orbiter.

◆ ERS Structures mass is 13.5% of ERS mass, including EEV payload.
  ◆ Aluminum structure mass is sufficiently low using thin-walled tubing; composites are not recommended.
Forward Work

- Develop sail structural analysis to verify vendor results and that structural supports and vehicle attachments are sufficient.
- Size the ERS sail carrier and retention doors in light composite.
- Divide orbiter structural panels into more properties to be able to optimize specific areas for component mounting and reduce mass overall.
SAIL STRUCTURES

Mark Davis
◆ The 20m ABAQUS finite element model was obtained from David Sleight (LaRC).

◆ Unfortunately the model in it’s current form does not run due to some changes in ABAQUS version that prevents proper convergence of the solution.

◆ The modification of the 20m model to change the convergence criteria is ongoing, while efforts are made to Obtain the L’Garde NASTRAN models.

◆ Once Either the ABAQUS model is converging or the L’Garde NEI NASTRAN model is obtained and verified the model can be scaled for the purposes of the DEPLOYTECH MSR Study.
Once either the NASTRAN or ABAQUS Finite Element Model is shown to run correctly then the Analyst can scale the model to be the correct size.

If then scaled the inner and outer diameter of the booms can be adjusted to give a reasonable about of deflection under solar pressure and then a modal run can be performed to determine the natural frequency of the system based on the new boom diameter.
- Forward Work on Solar Sail Model.
  - Get model to converge.
  - Scale Model to correct size.
  - Verify mass properties for integrated system.
  - Put solar pressure loads on model and modify boom diameter to control boom displacement.
  - Determine natural frequency of the Sail payload system using a modal run.
POWER

Leo Fabisinski
### Key Ground Rules & Assumptions

<table>
<thead>
<tr>
<th>Assumption</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Power Provision</strong></td>
<td>Separate Solar Conversion Systems, Common Power Electronics</td>
</tr>
<tr>
<td><strong>Solar Irradiance</strong></td>
<td>1361 W / m²</td>
</tr>
<tr>
<td><strong>Max Distance from Sun</strong></td>
<td>1.67 AU</td>
</tr>
<tr>
<td><strong>Bus Voltage</strong></td>
<td>28 V Nominal</td>
</tr>
<tr>
<td><strong>Max Solar Incidence Angle</strong></td>
<td>35°</td>
</tr>
<tr>
<td><strong>Ground Reference</strong></td>
<td>Common Ground Reference while connected</td>
</tr>
</tbody>
</table>
Power Required

## Orbiter

<table>
<thead>
<tr>
<th>Power Use Profile</th>
<th>Power (W)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Stand-By</td>
<td>183</td>
</tr>
<tr>
<td>Propulsive Maneuver</td>
<td>343</td>
</tr>
<tr>
<td>Rendezvous and Capture</td>
<td>489</td>
</tr>
<tr>
<td>Flight with Attitude Control</td>
<td>501</td>
</tr>
</tbody>
</table>

+ 30% Design Margin = **651 W**

## Earth Return Stage

<table>
<thead>
<tr>
<th>Power Use Profile</th>
<th>Power (W)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Keep-Alive</td>
<td>65</td>
</tr>
<tr>
<td>Stand-By</td>
<td>89</td>
</tr>
<tr>
<td>Normal Sail Flight</td>
<td>246</td>
</tr>
</tbody>
</table>

+ 30% Design Margin = **320 W**
Power System Design Approach

- Reference design’s solar array is sized as an aero-brake – Keep this array (with structure and mechanisms) as-is in the orbiter design.

- Replace the orbiter’s Power Management and Distribution (PMAD) system with a lower mass design based on the MESSENGER spacecraft. This PMAD system will return with the Earth Return Stage (ERS).

- Design a much lighter solar array consisting of 4 rigid panels for the ERS and cross-strap it to the PMAD for use on the return flight.

- Design the power harness to ‘break away’ when the ERS separates to jettison the Orbiter solar arrays and secondary batteries.
System Block Diagram

- Power Distribution Unit (PDU)
- Batteries
- Flight Computer
- Breakaway Harness
- Power System Electronics - Array Regulation & Battery Charge
- Flight Computer

Orbiter

Earth Return

Junction Box

Power

Data / Command
## Orbiter

<table>
<thead>
<tr>
<th>Item</th>
<th>Unit Mass (kg)</th>
<th>Qty</th>
<th>Total Mass (kg)</th>
<th>MGA (%)</th>
<th>Predicted Mass (kg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>UltraFlex Solar Array Assembly</td>
<td>20.2</td>
<td>1</td>
<td>20.2</td>
<td>10%</td>
<td>22.2</td>
</tr>
<tr>
<td>Power Distribution Unit</td>
<td>11.5</td>
<td>1</td>
<td>11.5</td>
<td>5%</td>
<td>12.1</td>
</tr>
<tr>
<td>Solar Array Junction Box</td>
<td>1.3</td>
<td>1</td>
<td>1.3</td>
<td>5%</td>
<td>1.4</td>
</tr>
<tr>
<td>Secondary Battery</td>
<td>19.2</td>
<td>3</td>
<td>57.6</td>
<td>10%</td>
<td>63.4</td>
</tr>
<tr>
<td>Cables, Breakaway Harness</td>
<td>16.0</td>
<td>1</td>
<td>16.0</td>
<td>30%</td>
<td>20.8</td>
</tr>
<tr>
<td><strong>Total Mass</strong></td>
<td><strong>106.6</strong></td>
<td></td>
<td></td>
<td><strong>12%</strong></td>
<td><strong>119.8</strong></td>
</tr>
</tbody>
</table>
Earth Return Stage

<table>
<thead>
<tr>
<th>Item</th>
<th>Unit Mass (kg)</th>
<th>Qty</th>
<th>Total Mass (kg)</th>
<th>MGA (%)</th>
<th>Predicted Mass (kg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Rigid Solar Array Panel</td>
<td>3.2</td>
<td>4</td>
<td>12.8</td>
<td>10%</td>
<td>14.1</td>
</tr>
<tr>
<td>Integrated Power Systems Electronics Box (PSE)</td>
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<td>1</td>
<td>6.6</td>
<td>5%</td>
<td>6.9</td>
</tr>
<tr>
<td>Power Distribution Unit</td>
<td>11.5</td>
<td>1</td>
<td>11.5</td>
<td>5%</td>
<td>12.1</td>
</tr>
<tr>
<td>Solar Array Junction Box</td>
<td>1.3</td>
<td>1</td>
<td>1.3</td>
<td>5%</td>
<td>1.4</td>
</tr>
<tr>
<td>Cables, Harness</td>
<td>4.5</td>
<td>1</td>
<td>4.5</td>
<td>30%</td>
<td>5.9</td>
</tr>
<tr>
<td><strong>Total Mass</strong></td>
<td></td>
<td></td>
<td><strong>36.7</strong></td>
<td>10%</td>
<td><strong>40.4</strong></td>
</tr>
</tbody>
</table>
Conclusions

- Original Orbiter power system design is modular - may be lightened by replacing power distribution with lower mass unit.

- Martian sun-sync orbit allows ERS to leave massive energy storage batteries behind, resulting in a power system mass of only 40.4 kg.

- For small solar power systems, 4 rigid panel array configuration is lighter than UltraFlex or other ‘Fan’ arrays.
Integrate the power distribution, solar array regulation, battery charge and power conversion into a small, single-board unit to reduce the mass of the power system.

Redesign the energy storage to use lighter batteries.

Further refine the breakaway harness to minimize mass while assuring low-risk operation.
AVIONICS

Pete Capizzo
Orbiter general mission assumptions:
- Provide telecommunication relay operations for the MSR lander and fetch rover
- Perform orbital sample (OS) Autonomous Rendezvous and Capture (AR&C)
- Deploy solar sail on Earth Return Stage (ERS), and perform separation of ERS
- Single fault tolerance for critical systems
- Controlled de-orbit required

Orbiter communications - GR&A
- Total Earth-link communication data rate: 10 kbps downlink, 2 kbps uplink
- Orbiter downlink frequency: once per day nominally, continuous during AR&C
- Total on board memory required: 4 Gbyte

Orbiter GN&C - GR&A
- Maneuvers and Attitude Control will be accomplished using RCS, no reaction wheels
- Pointing accuracy: 505 arcsec, Pointing knowledge: 252 arcsec

(Except for ERS deployment and separation, all Requirements are taken from JPL MSR)
ERS general mission assumptions:

- Sails will be inflated before separation
- Separation will be performed by the orbiter
- AR&C sensors and OS capture mechanism will be located on the Orbiter
- Fault tolerance: Single fault tolerance for critical systems

ERS Communications - GR&A

- The ERS will not be sending any science data
- Download frequency, once per day
  - sufficient for health and status data

ERS GN&C - GR&A

- Maneuvers and Attitude Control will be accomplished using the solar sail tip-vanes
  - No reaction wheels used
- The ERS has no high accuracy pointing requirements
  - AR&C GN&C will be done by the Orbiter
**Orbiter C&DH**
- The JPL MSR C&DH system was baselined for the Orbiter
- A redundant flight computer is used for single fault tolerance
- Some circuit board modifications may be required, sail deployment and separation
- Pyrotechnic units and jettison controllers were added to perform the ERS separation

**Orbiter communications**
- The JPL MSR communication system was baselined for the Orbiter
- This system is used to relay data from the MSR lander and fetch rover
- This same system is assumed sufficient to support ERS deployment and separation
- UHF-band used for Orbiter communications with the lander and fetch rover

**Orbiter GN&C**
- High accuracy GN&C will be required during close range approach and capture
  - Duration of high pointing accuracy assumed 2 hours
- For most of the other mission time, high accuracy GN&C will not be required
**ERS C&DH**

- For the ERS C&DH, the Messenger spacecraft Integrated Electronics Module (IEM) system was baselined
  - Messenger data handling capabilities may be more than what is needed for the ERS
- An alternative trade using the SpaceMicro Proton computer suite was done
  - It would be under half the mass of the IEM, but over twice the power required
  - With the impact on the power system, only 2 or 3 kg of mass would be saved

**ERS Communications**

- For the ERS communications, the Messenger spacecraft communication system was baselined
  - This system allows over 2 kbps of data to be transmitted from Mercury to Earth
  - 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.
- Smaller arrays or SSPAs could be considered, but to keep the high TRL the system was unaltered
ERS GN&C

- All the ERS sail attitude and control hardware is included in sail section of the MEL
- The use of MSFC MPDM remote electronic distribution units is suggested to minimized system mass
  - These units are small and lightweight, about 100 grams each
- No high accuracy sun pointing is required for navigation of the ERS, low accuracy components should work:
  - Pointing accuracy 2.5 deg assumed sufficient
  - Pointing knowledge 1 deg assumed sufficient
  - AeroAstro star trackers (0.1 deg)
  - Northrop LN200 IMU (< 0.15 deg/sqrthr random walk)
- An observation camera is included in the ERS ACS
  - This camera is for monitoring the deployment of the sail and the ERS separation
  - It can also be used to monitor the deployment of the EEV for earth entry
Phased arrays are linear polarization. All other antennas are right-hand circular polarization.
## Orbiter Avionics Mass Summary

<table>
<thead>
<tr>
<th>Subsystem</th>
<th>Basic Mass (kg)</th>
<th>Contingency (%)</th>
<th>Predicted Mass (kg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Command and Data Handling</td>
<td>48.6</td>
<td>20%</td>
<td>58.44</td>
</tr>
<tr>
<td>Guidance Navigation and Control</td>
<td>11.96</td>
<td>3%</td>
<td>12.32</td>
</tr>
<tr>
<td>Communications</td>
<td>29.15</td>
<td>6%</td>
<td>30.76</td>
</tr>
<tr>
<td>Vehicle Systems</td>
<td>19.70</td>
<td>13%</td>
<td>22.33</td>
</tr>
<tr>
<td>Total</td>
<td>109.41</td>
<td></td>
<td>123.85</td>
</tr>
</tbody>
</table>

## ERS Avionics Mass Summary

<table>
<thead>
<tr>
<th>Subsystem</th>
<th>Basic Mass (kg)</th>
<th>Contingency (%)</th>
<th>Predicted Mass (kg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Command and Data Handling</td>
<td>20.30</td>
<td>11%</td>
<td>22.48</td>
</tr>
<tr>
<td>Guidance Navigation and Control</td>
<td>5.10</td>
<td>3%</td>
<td>5.26</td>
</tr>
<tr>
<td>Communications</td>
<td>24.61</td>
<td>3%</td>
<td>25.35</td>
</tr>
<tr>
<td>Vehicle Systems</td>
<td>1.80</td>
<td>3%</td>
<td>1.85</td>
</tr>
<tr>
<td>Total</td>
<td>51.81</td>
<td></td>
<td>54.94</td>
</tr>
</tbody>
</table>
No unusual or extravagant avionics is required of a solar sail return stage

The existing orbiter design with some minor modifications is sufficient

Using a proven flight system for the ERS like that of the Messenger spacecraft will reduce risk
**Forward Work**

**Orbiter:**
- For the AR&C operation the slew rates need to be determined.
- Determine close proximity high accuracy pointing time, assumed 2 hours, but may be more.

**ERS:**
- Trade smaller phase arrays antennas and SSPA against TRL levels and risk.
THERMAL

Linda Hornsby
Thermal Desktop® model of orbiter & ERS bus developed to assess subsystems interface temperatures

Structural panels double as radiative surfaces.

Structure modeled as aluminum, panel thickness consistent with structural design

MLI blankets are assumed to cover all spacecraft surfaces

Subsystems equipment heat loads are imposed on the structure and modeled as area averaged loads

Propellant tank thermal control is achieved with MLI and heaters

RCS thrusters, antennas, solar array & array mechanisms are not part of Pre-Phase A analysis
Subsystems avionics and power systems are mounted to the exterior structural panels of the bus.

Chemical propulsion tanks are located inside the bus.
Orbiter Analysis Results

Mars Orbit

Degrees C

- CDS 40 W
- Comm – Xband 66 W
- Batteries & ACS 61.5 W
- Comm – UHF 53.5 W

Mars Transit
(11W tank heater power)
### Orbiter Heat Dissipation & Operating Temperatures

<table>
<thead>
<tr>
<th>Component</th>
<th>Total Heat Dissipation</th>
<th>Operating Temp Range</th>
</tr>
</thead>
<tbody>
<tr>
<td>ACS (IMU Assembly)</td>
<td>24 W</td>
<td>-40 C to 85 C</td>
</tr>
<tr>
<td>Command &amp; Data System</td>
<td>40 W</td>
<td>-40 C to 85 C</td>
</tr>
<tr>
<td>Communication System (UHF)</td>
<td>53.5 W</td>
<td>-40 C to 85 C</td>
</tr>
<tr>
<td>Communication System (X-band)</td>
<td>66 W</td>
<td>-40 C to 85 C</td>
</tr>
<tr>
<td>Batteries (3)</td>
<td>37.5 W</td>
<td>0 C to 45 C</td>
</tr>
<tr>
<td>Propellant and Pressurant Tanks (Heaters)</td>
<td>11 W</td>
<td>10 C to 30 C</td>
</tr>
<tr>
<td><strong>Total</strong></td>
<td><strong>232 W</strong></td>
<td></td>
</tr>
</tbody>
</table>

All predicted interface temperatures are within acceptable range of 0 to 30 degrees C for the trip to Mars and Mars orbit cases analyzed.
The Earth Return Stage is designed with multiple avionics shelves.

Subsystems equipment heat loads are equally distributed on the shelves.

ERS structure is wrapped with MLI.
ERS Analysis Results

- **Operational Phase**
  - Avionics Shelf (35 to 45 deg. C)
  - 138.5 W per Shelf

- **Keep Alive Phase**
  - Avionics Shelf (-16 to -6 deg. C)
  - 66 W per Shelf
ERS Heat Dissipation & Operating Temperatures

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

All predicted interface temperatures are within acceptable range of -16 to 45 degrees C for the operational and keep alive cases analyzed.
## Thermal Model Surface Optical Properties

<table>
<thead>
<tr>
<th>Material Description</th>
<th>Material</th>
<th>Absorptivity</th>
<th>Emissivity</th>
</tr>
</thead>
<tbody>
<tr>
<td>Spacecraft Bus &amp; ERS Internal Surfaces</td>
<td>Black Annodized</td>
<td>.9</td>
<td>.9</td>
</tr>
<tr>
<td>Orbiter &amp; ERS External Surfaces (EOL)</td>
<td>EOL Silverized Teflon</td>
<td>.25</td>
<td>.87</td>
</tr>
<tr>
<td></td>
<td>MLI $\varepsilon^* = .02$</td>
<td>.6</td>
<td>.09</td>
</tr>
<tr>
<td>Propulsion &amp; Pressurant Tanks</td>
<td>MLI $\varepsilon^* = .02$</td>
<td>.6</td>
<td>.09</td>
</tr>
</tbody>
</table>
# Thermal Control Mass Summary

<table>
<thead>
<tr>
<th>Deploytech Orbiter Bus</th>
<th>Basic Mass (kg)</th>
<th>Contingency (%)</th>
<th>Predicted Mass (kg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>3.0 Thermal</td>
<td>22.14</td>
<td>30</td>
<td>28.78</td>
</tr>
<tr>
<td>3.1 Insulation</td>
<td>13.6</td>
<td>30</td>
<td>17.68</td>
</tr>
<tr>
<td>3.2 Heaters/Thermistors</td>
<td>0.70</td>
<td>30</td>
<td>0.91</td>
</tr>
<tr>
<td>3.3 Paint/Coatings</td>
<td>2.04</td>
<td>30</td>
<td>2.65</td>
</tr>
<tr>
<td>3.4 Misc</td>
<td>3.40</td>
<td>30</td>
<td>4.42</td>
</tr>
<tr>
<td>3.5 Propulsion Thermal Control</td>
<td>2.40</td>
<td>30</td>
<td>3.12</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Deploytech ERS</th>
<th>Basic Mass (kg)</th>
<th>Contingency (%)</th>
<th>Predicted Mass (kg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>3.0 Thermal</td>
<td>16.82</td>
<td>30</td>
<td>21.87</td>
</tr>
<tr>
<td>3.1 Insulation</td>
<td>10.40</td>
<td>30</td>
<td>13.52</td>
</tr>
<tr>
<td>3.2 Heaters/Thermistors</td>
<td>0.70</td>
<td>30</td>
<td>0.91</td>
</tr>
<tr>
<td>3.3 Paint/Coatings</td>
<td>3.12</td>
<td>30</td>
<td>4.06</td>
</tr>
<tr>
<td>3.4 Misc</td>
<td>2.60</td>
<td>30</td>
<td>3.38</td>
</tr>
</tbody>
</table>
Conclusions

- Based on preliminary, pre-Phase A level analyses, 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.
Forward Work

- Add details to thermal model, including solar arrays, avionics & power system box geometry and solar sail geometry.
- Assess impact of thermal surface degradation over mission life.
- Analyze hot and cold bounding cases.
- Analyze transient cases incorporating heat capacitance and power timeline
MASS PROPERTIES

C. Dauphne Maples
<table>
<thead>
<tr>
<th>Category</th>
<th>Basic Mass (kg)</th>
<th>Contingency (%)</th>
<th>Contingency (kg)</th>
<th>Predicted Mass (kg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1.0 Structures</td>
<td>139.45</td>
<td>23%</td>
<td>32.33</td>
<td>171.78</td>
</tr>
<tr>
<td>2.0 Chemical Propulsion</td>
<td>117.67</td>
<td>25%</td>
<td>29.42</td>
<td>147.09</td>
</tr>
<tr>
<td>3.0 Thermal</td>
<td>22.14</td>
<td>30%</td>
<td>6.64</td>
<td>28.78</td>
</tr>
<tr>
<td>4.0 Avionics</td>
<td>109.41</td>
<td>13%</td>
<td>14.44</td>
<td>123.85</td>
</tr>
<tr>
<td>5.0 Power</td>
<td>106.60</td>
<td>12%</td>
<td>13.22</td>
<td>119.82</td>
</tr>
<tr>
<td><strong>DRY MASS TOTAL</strong></td>
<td><strong>495.27</strong></td>
<td><strong>19%</strong></td>
<td><strong>96.04</strong></td>
<td><strong>591.31</strong></td>
</tr>
<tr>
<td>6.0 Non-Propellant Fluids</td>
<td>26.46</td>
<td></td>
<td></td>
<td>26.46</td>
</tr>
<tr>
<td>7.0 Payload</td>
<td>149</td>
<td>30%</td>
<td>44.70</td>
<td>193.70</td>
</tr>
<tr>
<td><strong>INERT MASS</strong></td>
<td><strong>175.46</strong></td>
<td></td>
<td></td>
<td><strong>220.16</strong></td>
</tr>
<tr>
<td><strong>TOTAL LESS PROPELLANT</strong></td>
<td><strong>670.73</strong></td>
<td></td>
<td></td>
<td><strong>811.47</strong></td>
</tr>
<tr>
<td>8.0 Maneuver Propellant</td>
<td>779.20</td>
<td></td>
<td></td>
<td>779.20</td>
</tr>
<tr>
<td><strong>TOTAL ORBITER MASS</strong></td>
<td><strong>1449.93</strong></td>
<td></td>
<td></td>
<td><strong>1590.67</strong></td>
</tr>
</tbody>
</table>
Mass Properties - ERS

<table>
<thead>
<tr>
<th></th>
<th>Basic Mass (kg)</th>
<th>Contingency (%)</th>
<th>Contingency (kg)</th>
<th>Predicted Mass (kg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1.0</td>
<td>Structures</td>
<td>38.02</td>
<td>26%</td>
<td>10.05</td>
</tr>
<tr>
<td>2.0</td>
<td>Solar Sail</td>
<td>125.08</td>
<td>12%</td>
<td>14.51</td>
</tr>
<tr>
<td>3.0</td>
<td>Thermal</td>
<td>16.82</td>
<td>30%</td>
<td>5.05</td>
</tr>
<tr>
<td>4.0</td>
<td>Avionics</td>
<td>51.81</td>
<td>6%</td>
<td>3.12</td>
</tr>
<tr>
<td>5.0</td>
<td>Power</td>
<td>36.70</td>
<td>10%</td>
<td>3.60</td>
</tr>
<tr>
<td><strong>DRY MASS TOTAL</strong></td>
<td><strong>268.43</strong></td>
<td><strong>14%</strong></td>
<td><strong>36.33</strong></td>
<td><strong>304.77</strong></td>
</tr>
<tr>
<td>6.0</td>
<td>Payload</td>
<td>52.05</td>
<td></td>
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</tr>
<tr>
<td><strong>TOTAL ERS MASS</strong></td>
<td><strong>320.48</strong></td>
<td></td>
<td></td>
<td><strong>356.82</strong></td>
</tr>
</tbody>
</table>
### Mass Properties – Total Launch Mass

<table>
<thead>
<tr>
<th></th>
<th>Basic Mass (kg)</th>
<th>Predicted Mass (kg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Orbiter Mass</td>
<td>1449.93</td>
<td>1590.67</td>
</tr>
<tr>
<td>ERS Mass</td>
<td>320.48</td>
<td>356.82</td>
</tr>
<tr>
<td><strong>TOTAL LAUNCH MASS</strong></td>
<td><strong>1770.41</strong></td>
<td><strong>1947.49</strong></td>
</tr>
</tbody>
</table>
STUDY CONCLUSIONS
Primary objective to reduce baseline mission from three to two launches

Required to maintain baseline mission objectives for all three elements, only redesigning the Earth return propulsion system

ACO MSR Orbiter and ERS had a launch mass too large to be packaged on the Max-C Rover or Lander launch vehicles

<table>
<thead>
<tr>
<th></th>
<th>Predicted Launch Mass (kg)</th>
<th>Launch Vehicle</th>
<th>Launch Capability (kg)</th>
<th>Contingency (kg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Max-C Rover</td>
<td>4457.4*</td>
<td>Atlas V 531</td>
<td>4980</td>
<td>522.6</td>
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<tr>
<td>MSR Orbiter</td>
<td>3270</td>
<td>Atlas V 551</td>
<td>4770</td>
<td>1500</td>
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<tr>
<td>Lander and Ascent Vehicle</td>
<td>4668</td>
<td>Atlas V 551</td>
<td>5130</td>
<td>462</td>
</tr>
<tr>
<td>ACO MSR Orbiter and ERS</td>
<td>1947.49</td>
<td>--</td>
<td>--</td>
<td>--</td>
</tr>
</tbody>
</table>

* 300kg was bookmarked in the baseline architecture as a placeholder for ExoMars
Conclusions

- The study was unable to eliminate one launch due to final launch mass numbers
  - Restricted changes to GR&A
  - Maintained subsystem orbiter components and functions
- 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
- Met all mission requirements
NASA ELV Performance Estimation Curve(s)
High Energy Orbits
Please note ground rules and assumptions below.

Source: http://elvperf.ksc.nasa.gov
Forward Work

- Revisit 2 launch configuration by adjusting the GR&A
  - Reduce Max-C and Orbiter mass
  - Eliminate Max-C Rover
  - Re-evaluate launch vehicle options
  - Use solar sail for all interplanetary primary propulsion

<table>
<thead>
<tr>
<th>Launch Vehicle Options</th>
<th>Series Option</th>
<th>2024 Launch Capability (kg)</th>
<th>2026 Launch Capability (kg)</th>
<th>2028 Launch Capability (kg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Atlas V</td>
<td>531</td>
<td>3900</td>
<td>4200</td>
<td>4200</td>
</tr>
<tr>
<td>Atlas V</td>
<td>541</td>
<td>4400</td>
<td>4700</td>
<td>4800</td>
</tr>
<tr>
<td>Atlas V</td>
<td>551</td>
<td>5000</td>
<td>5150</td>
<td>5250</td>
</tr>
<tr>
<td>Falcon 9</td>
<td>v1.0</td>
<td>1300</td>
<td>1400</td>
<td>1450</td>
</tr>
<tr>
<td>Falcon 9</td>
<td>v1.1</td>
<td>2500</td>
<td>2600</td>
<td>2700</td>
</tr>
</tbody>
</table>

Estimated values from NLSII performance values
BACKUP
Power System Electronics Box

- MESSENGER Heritage
- Solar array regulation
- Peak Power Tracking (PPT)
- Battery Charge Control
- Direct redundant serial interface to Power Distribution Unit
Solar Array Junction Box

- MESSENGER Heritage
- Solar array string isolation circuits.
- Solar array shunt monitors.
- Input power fuses
Power Distribution Unit

- MESSENGER Heritage
- Load circuit switching
- MIL-STD-1553 interface to Flight Computer
- Propulsion thruster control switching
- Attitude Control actuator switching
- Overload protection for all circuits