National Aeronautics and Space Administration

DEPLOYTECH

Mars Sample Return Concept Study

MSFC Advanced Concepts

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Study Roster

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

Study Schedule

	Monday	Tuesday	Wednesday	Thursday	Friday
June 4th			Study Kickoff	Background Research	Background Research
June 11th	Background Research	Background Research	Ground Rules and Assumptions	Ground Rules and Assumptions	Ground Rules and Assumptions
June 18th	Ground Rules and Assumptions	Design Session	Design Session	Design Session	
June 25th	Design Session	Design Session	Document	Document	

DEPLOYTECH OVERVIEW

DEPLOYTECH Introduction

- 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

DEPLOYTECH Introduction

- 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

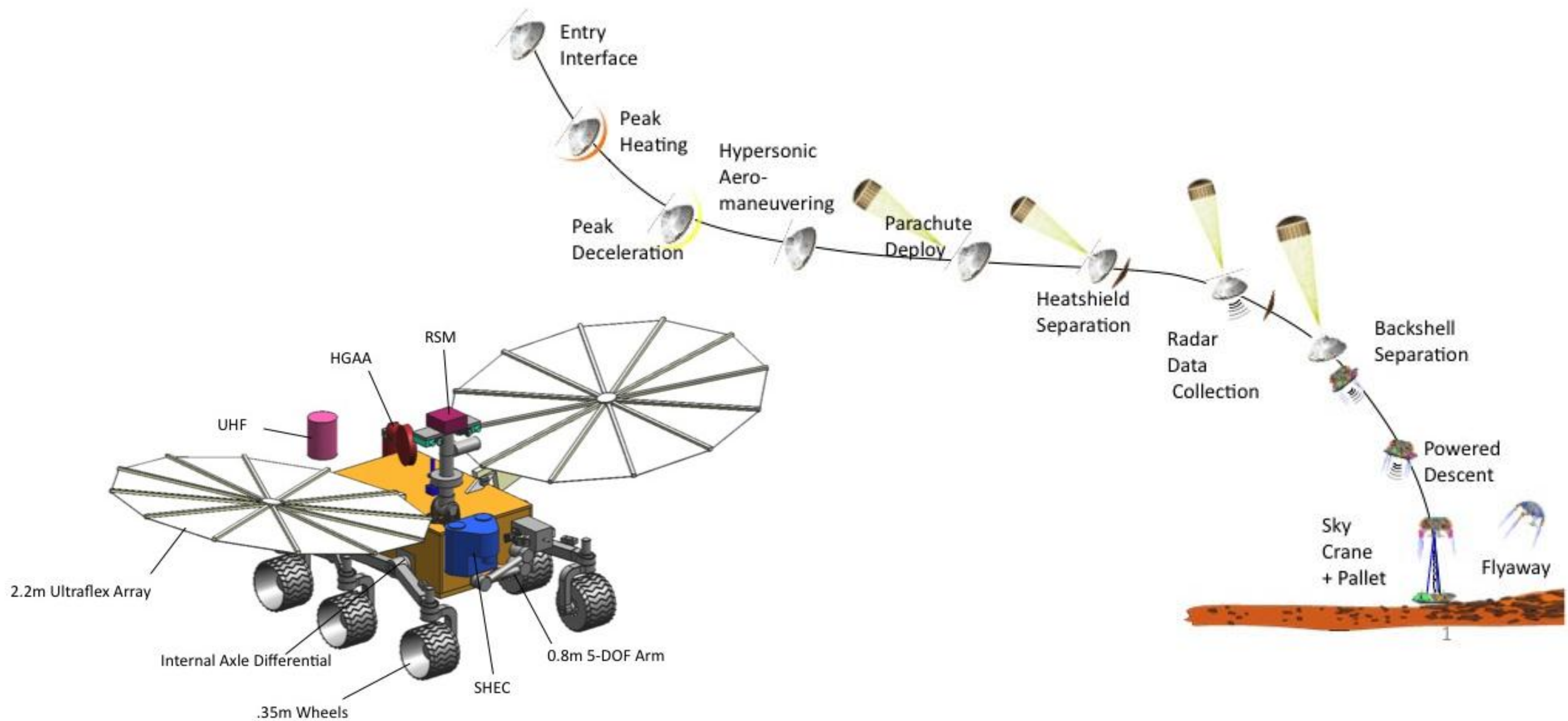
MARS SAMPLE RETURN

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.)

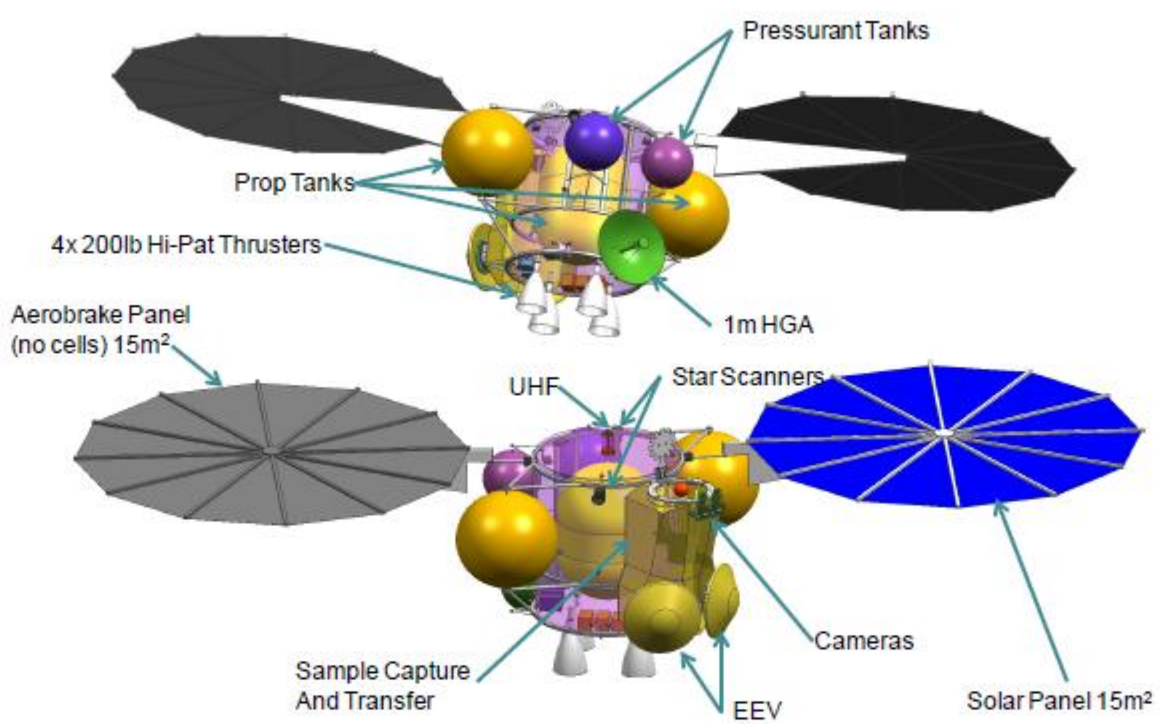
Mission Overview

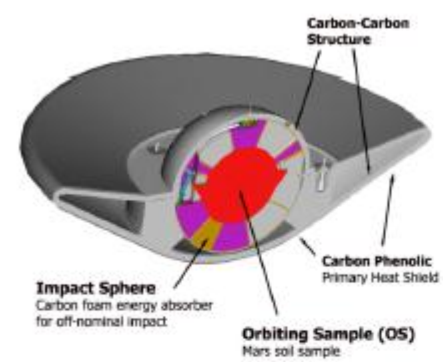
- Astrobiology Explorer (Rover)
  - Launch = mid 2018



Mission Overview

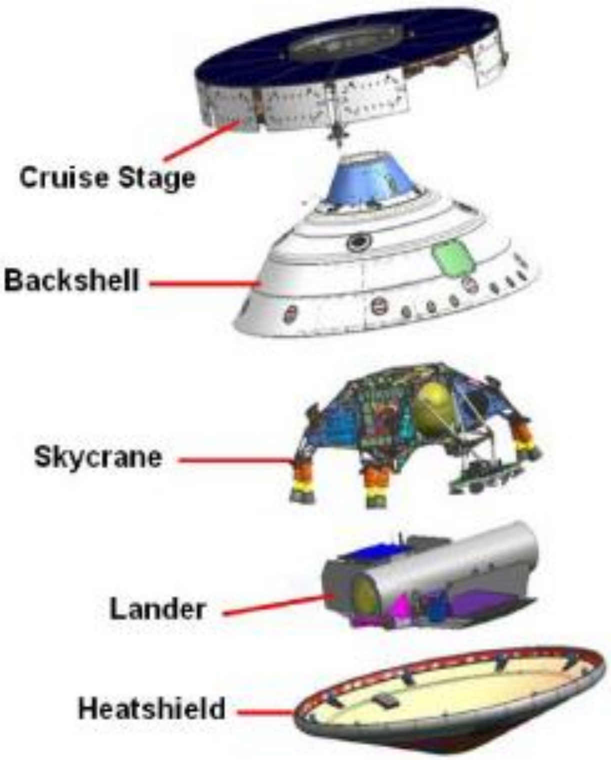
- Sample Return Orbiter (with EEV)
  - Launch = 2022

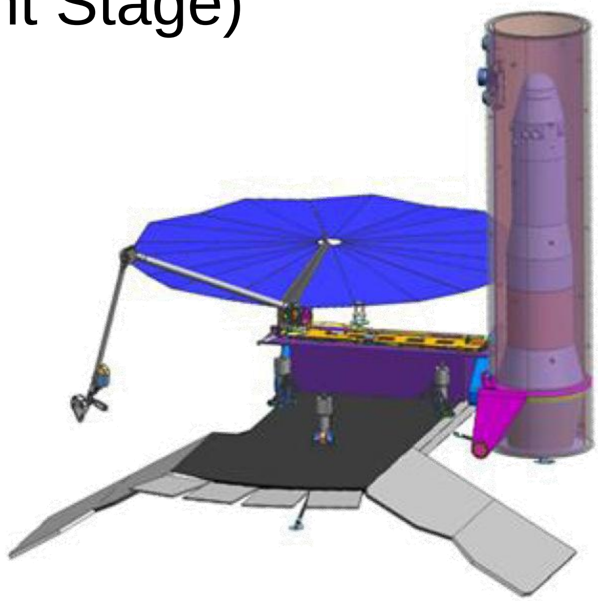


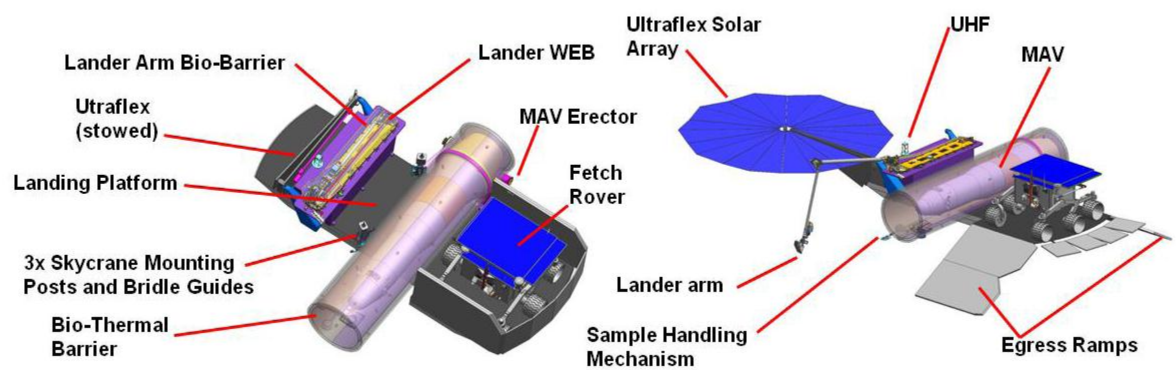


Mission Overview

- Sample Return Lander (with Ascent Stage)
  - Launch 2024







GROUND RULES AND ASSUMPTIONS

C. Dauphne Maples

Ground Rules and Assumptions

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

Ground Rules and Assumptions

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	

ORBITER MISSION ANALYSIS

Dan Thomas

Assumptions

- Earth departure dates ranging from 2022 – 2035
- Minimum energy transfers
  - Keep departure C3s < 15 – 20 km^2/s^2
  - 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

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

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.

Mission ΔVs and Maneuver Propellants

- Launch mass (first estimate) ~ 2110 kg

  Includes:

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

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

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 ΔV 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

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

Total Dry Mass: 147.1

Loaded Propellant (with residuals)	
Hydrazine	441.5
NTO	361.1

Total Propellant: 802.6

Pressurant	
Gaseous Helium	3.1

Total Wet Mass: 952.8 kg

Future Work

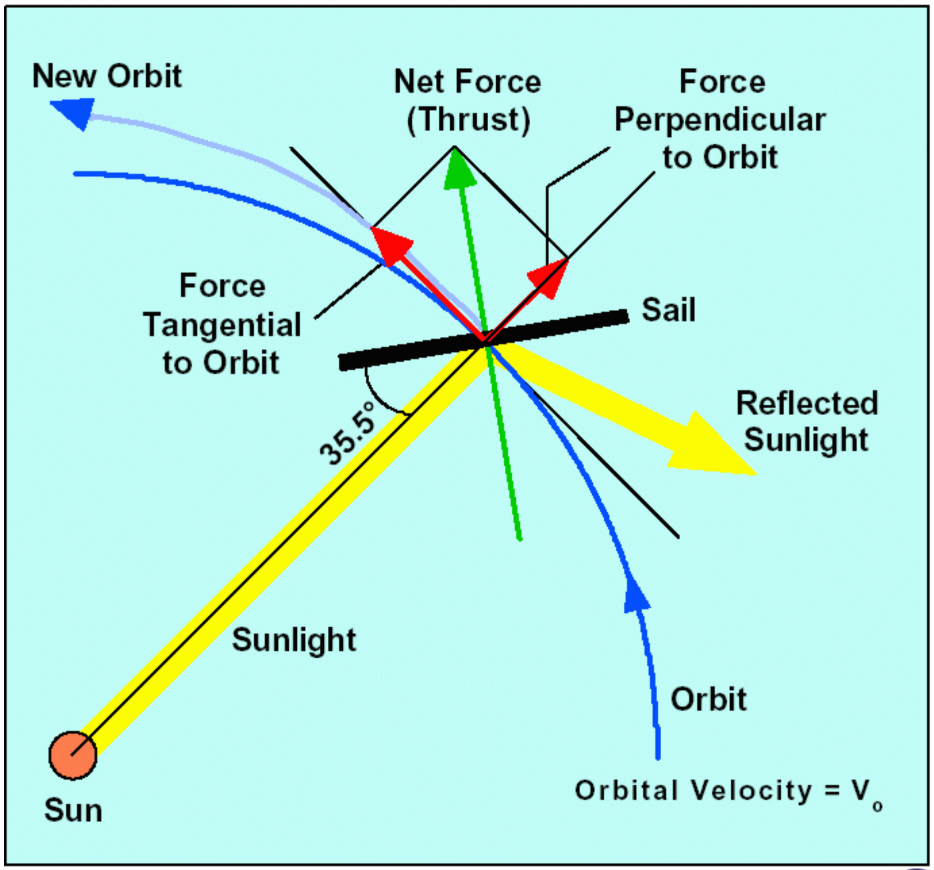
- Investigate potential savings provided by biprop-only system with MMH/NTO (i.e. replace small monoprop engines with biprop options).
  - Astrium provides small biprop engines with thrust levels down to 4 N (0.9 lbf)
  - Aerojet's R-6D engine (22 N = 5 lbf) has a low minimum impulse bit and may be an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

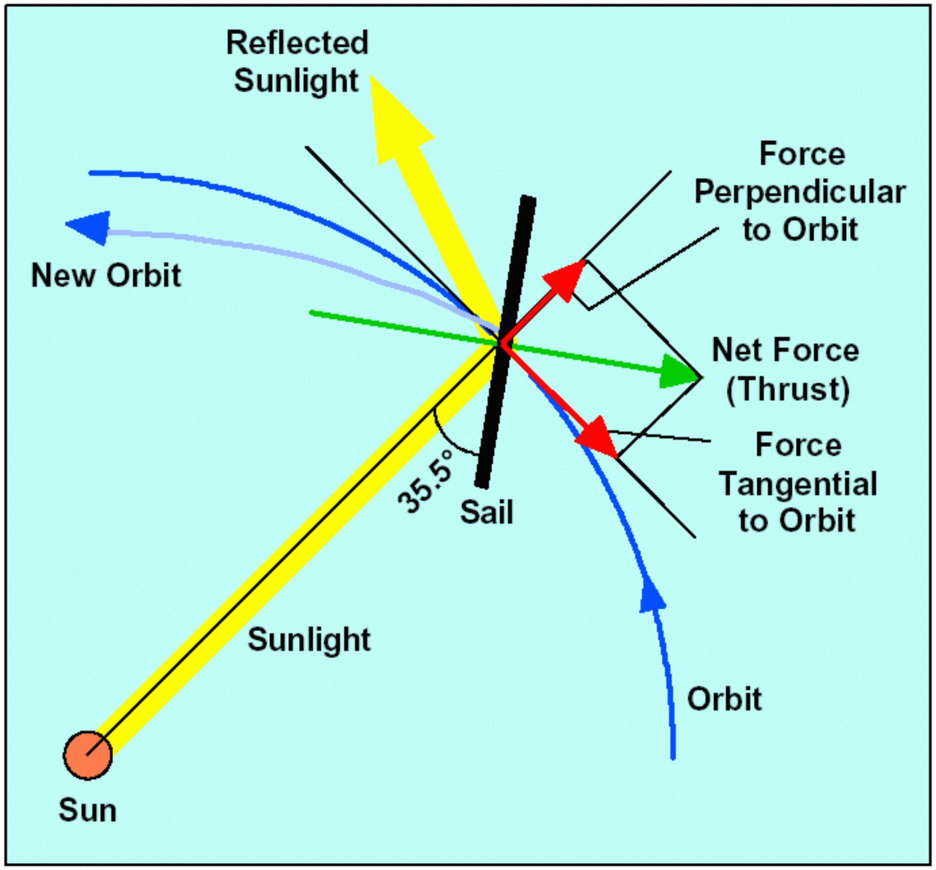
EARTH RETURN SYSTEM MISSION ANALYSIS

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/km^2 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)

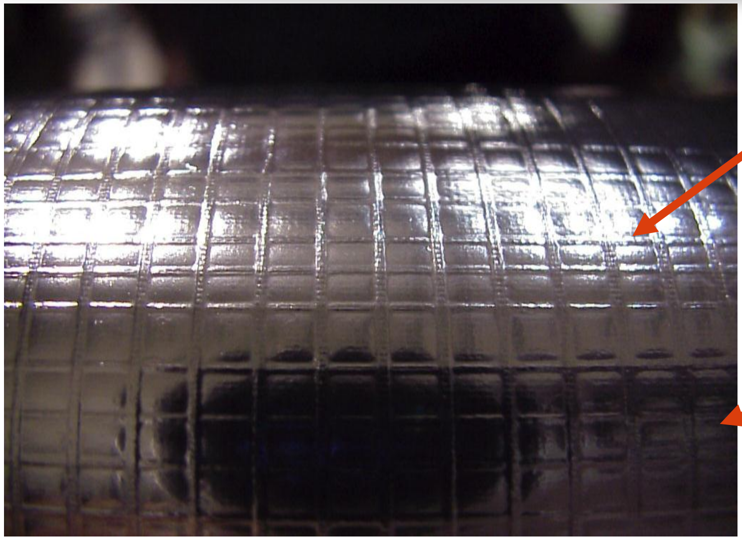


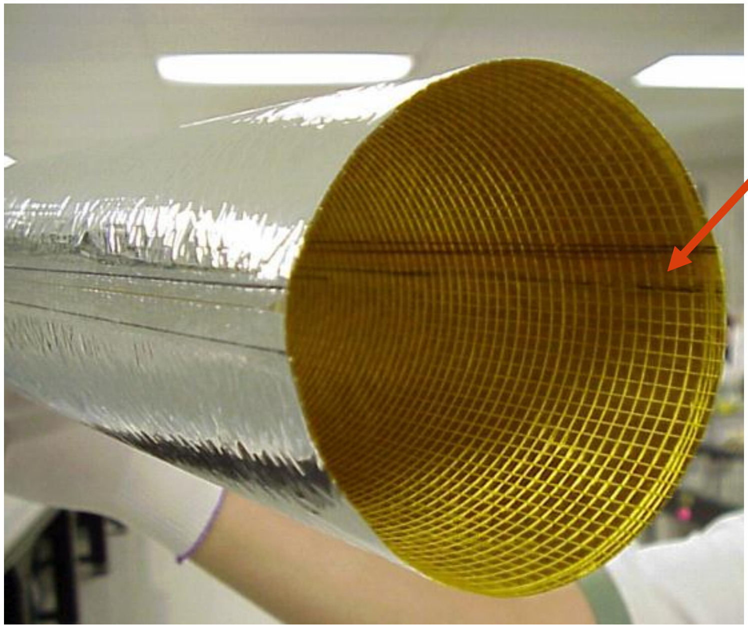


Solar Sail Heritage

- Solar Sail Propulsion Subsystem is a150 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

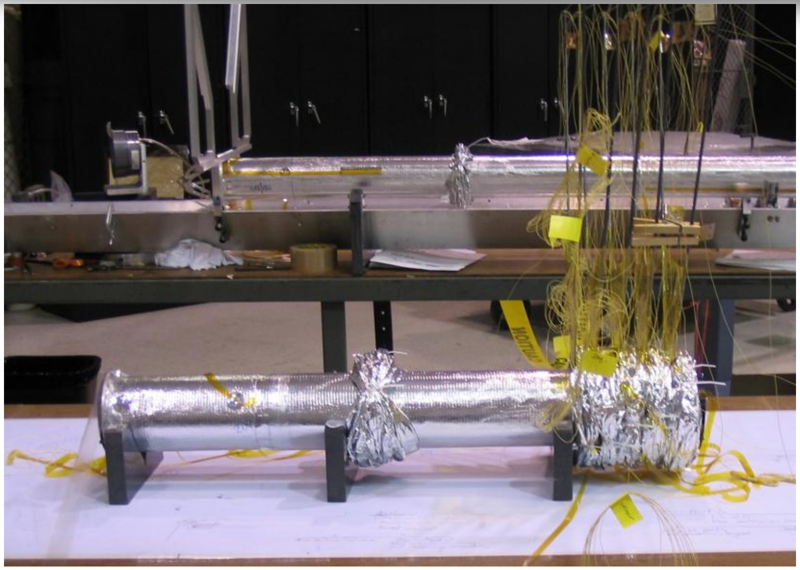
Boom Design





- Load bearing longitudinal uni-directional fibers
  - Fibers impregnated with sub-tg resin, rigid below -20º 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

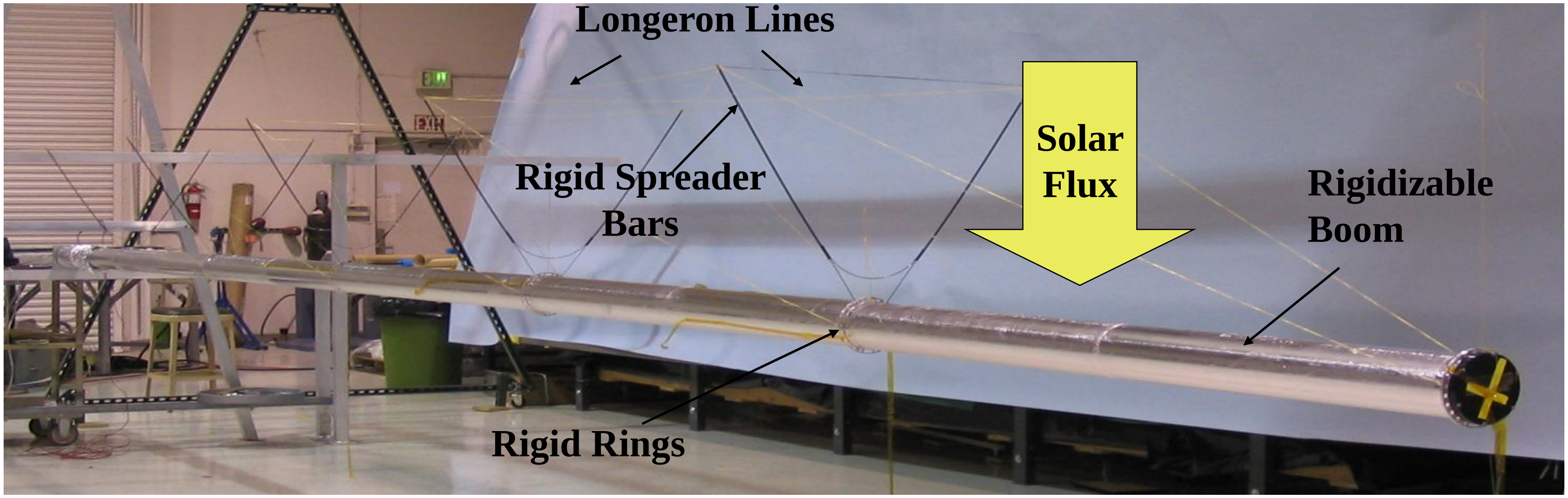


7.5m beam stowed to ~.5m

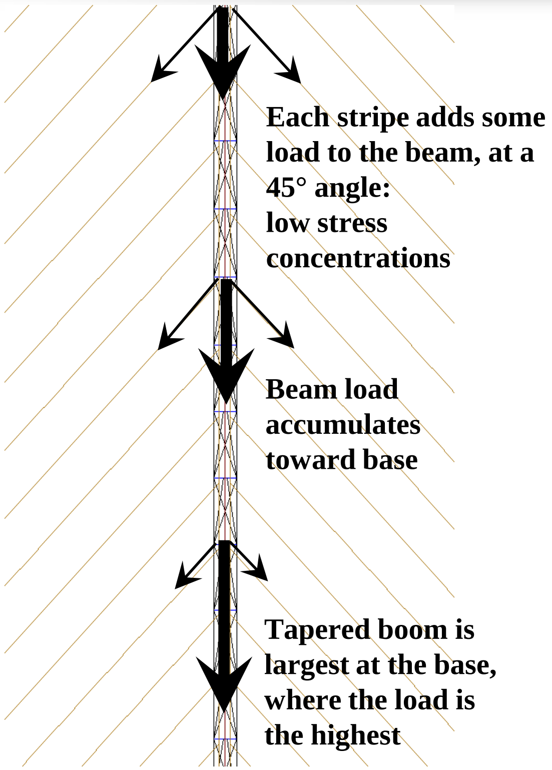
Beam structure

- 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

Deployed 7.5m beam



Net/Membrane Sail Design



Chords are suspended from the boom rings

Sail material is laid over the net allowing billow

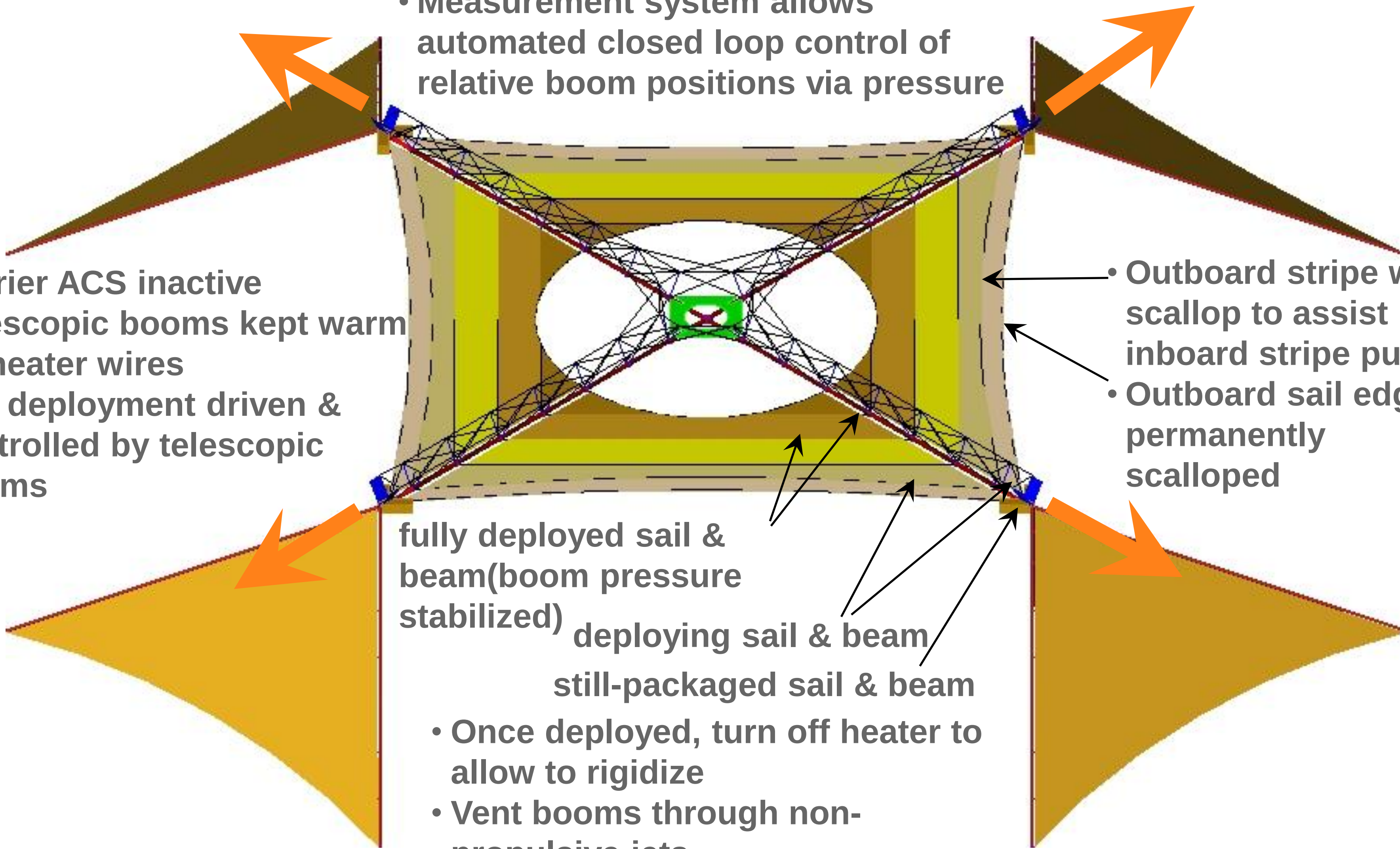
- Net Membrane
  - Sail supported by high modulus and low CTE net
  - Additional membrane materials allows thermal compliance
  - Sail properties effect local billow between net members only
  - Global sail shape is stable
- Advantages
  - Net defines the overall sail shape
  - Stability and geometry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

- Outboard stripe wires scallop to assist inboard stripe pullout
- Outboard sail edge is permanently scalloped



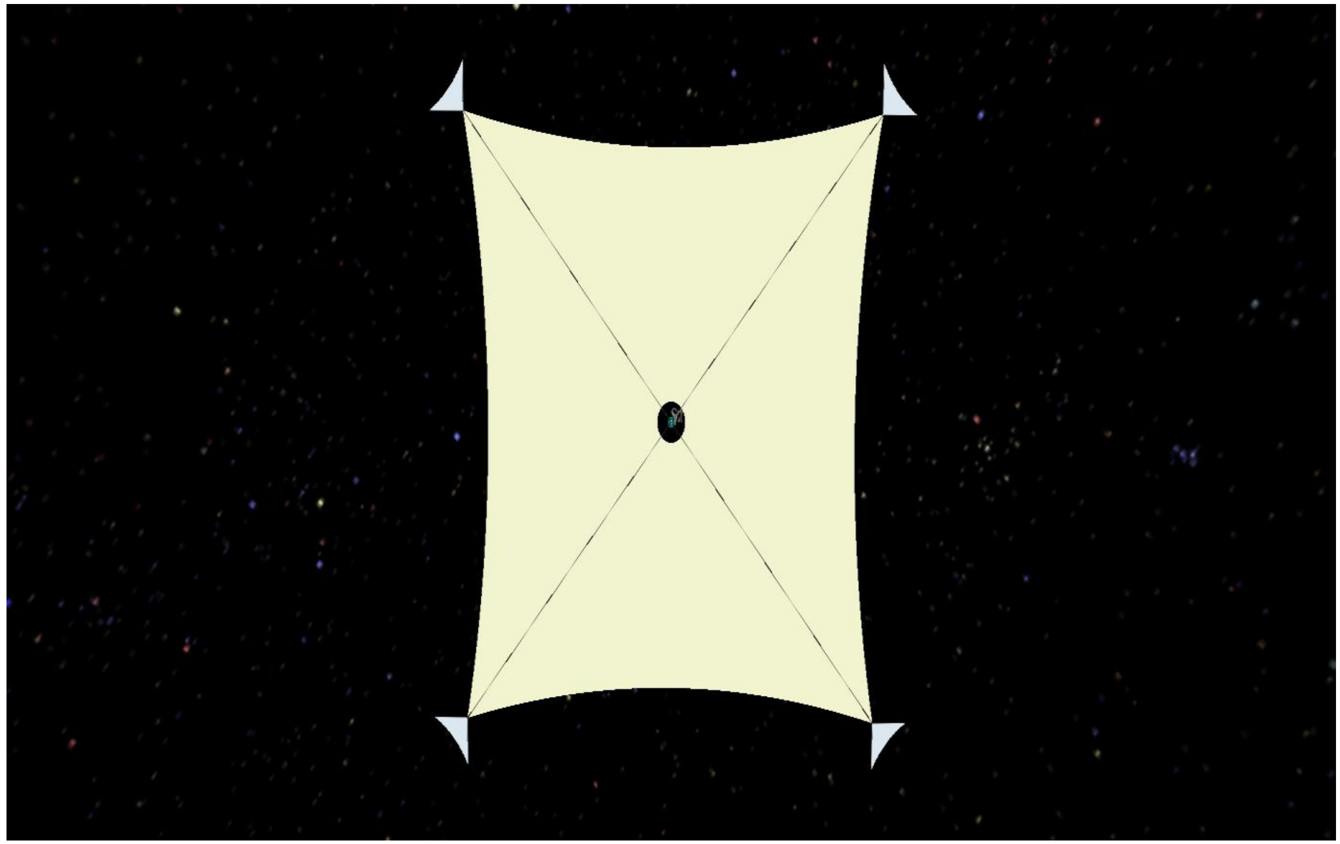
- Once deployed, turn off heater to allow to rigidize
- Vent booms through non-propulsive jets

Solar Sail Propulsion System Mass

	Solar Sail	Basic Mass (kg)	Contingency (%)	Predicted Mass (kg)
2.0	**ERS Solar Sail**	125.08	12	**139.59**
2.1	CP1 Sail Material, coatings, ripestop, and stripe net	101.81	8	110.21
2.2	Solar Sail Beam	14.90	26	18.73
2.3	Attitude Control System. Tip vanes	8.37	27	10.65

Solar Sail Characteristics

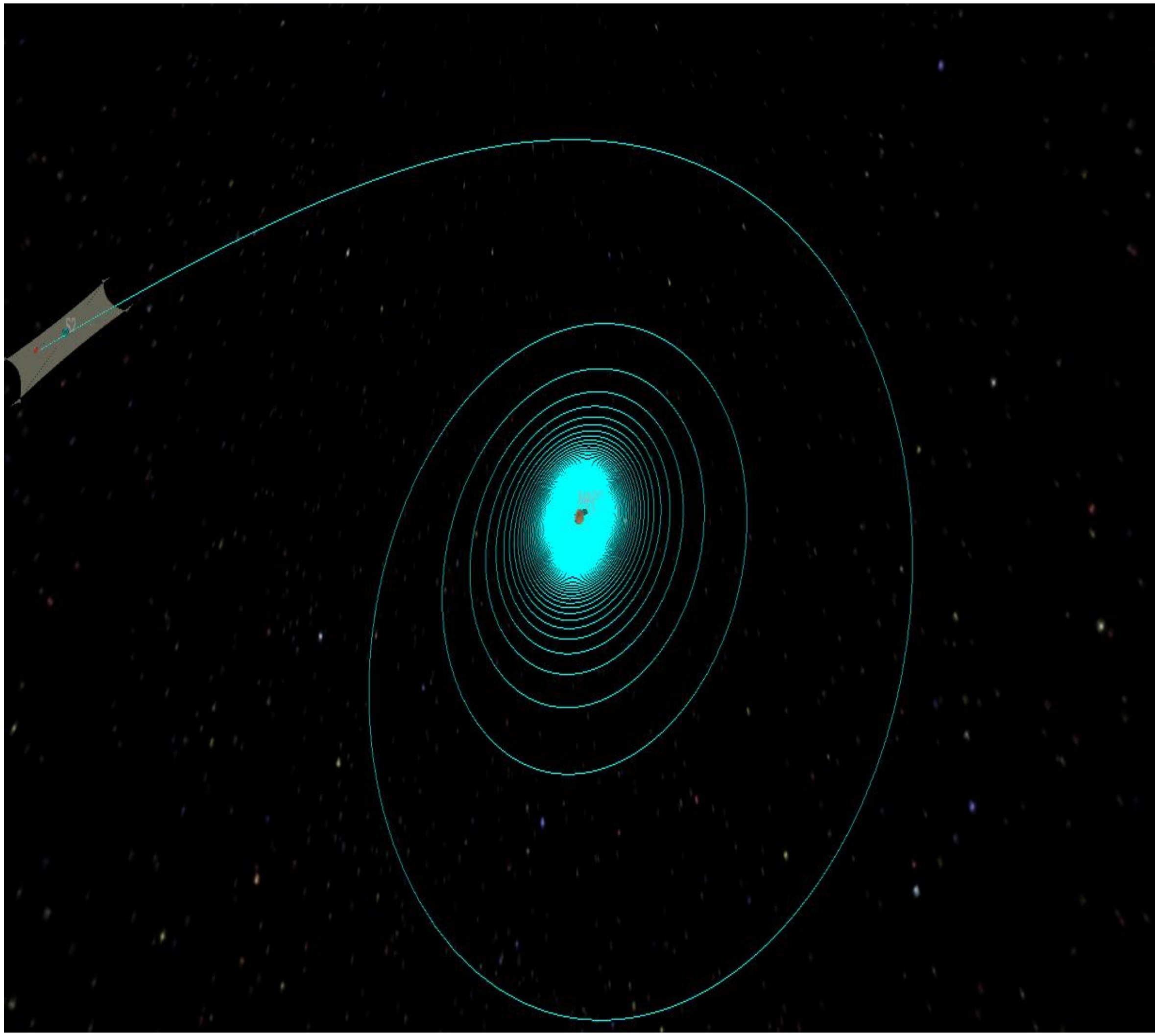
- Assumptions
  - Area = 18,887 m2
  - Payload Mass = 200 kg
  - Sail Mass = 115 kg
  - 2.5 µm CP1



- Characteristic Acceleration = 0.50 mm/s2

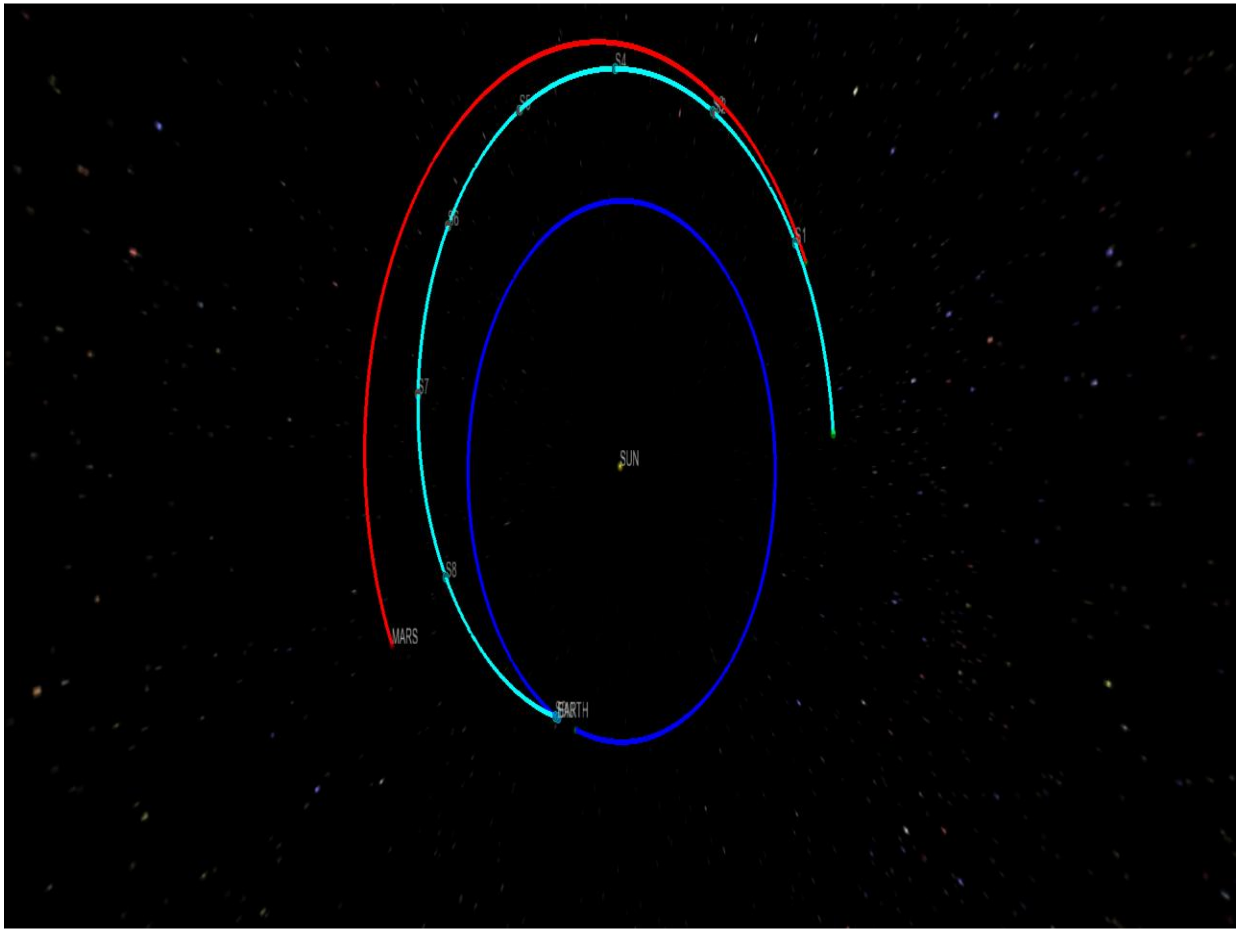
Mars Spiral Out

- 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



Mars-to-Earth Transfer

- 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)

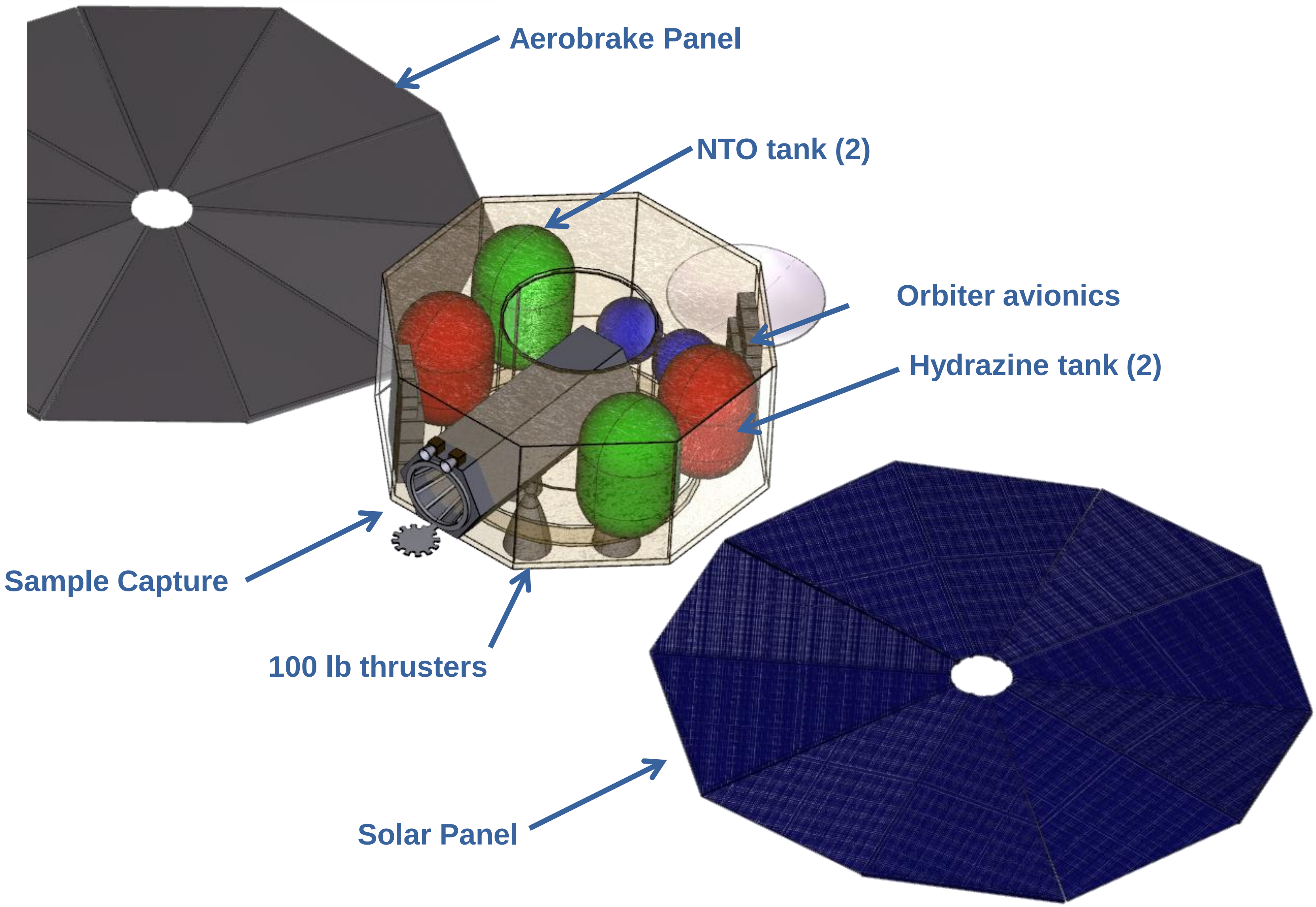
Forward Work

- 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

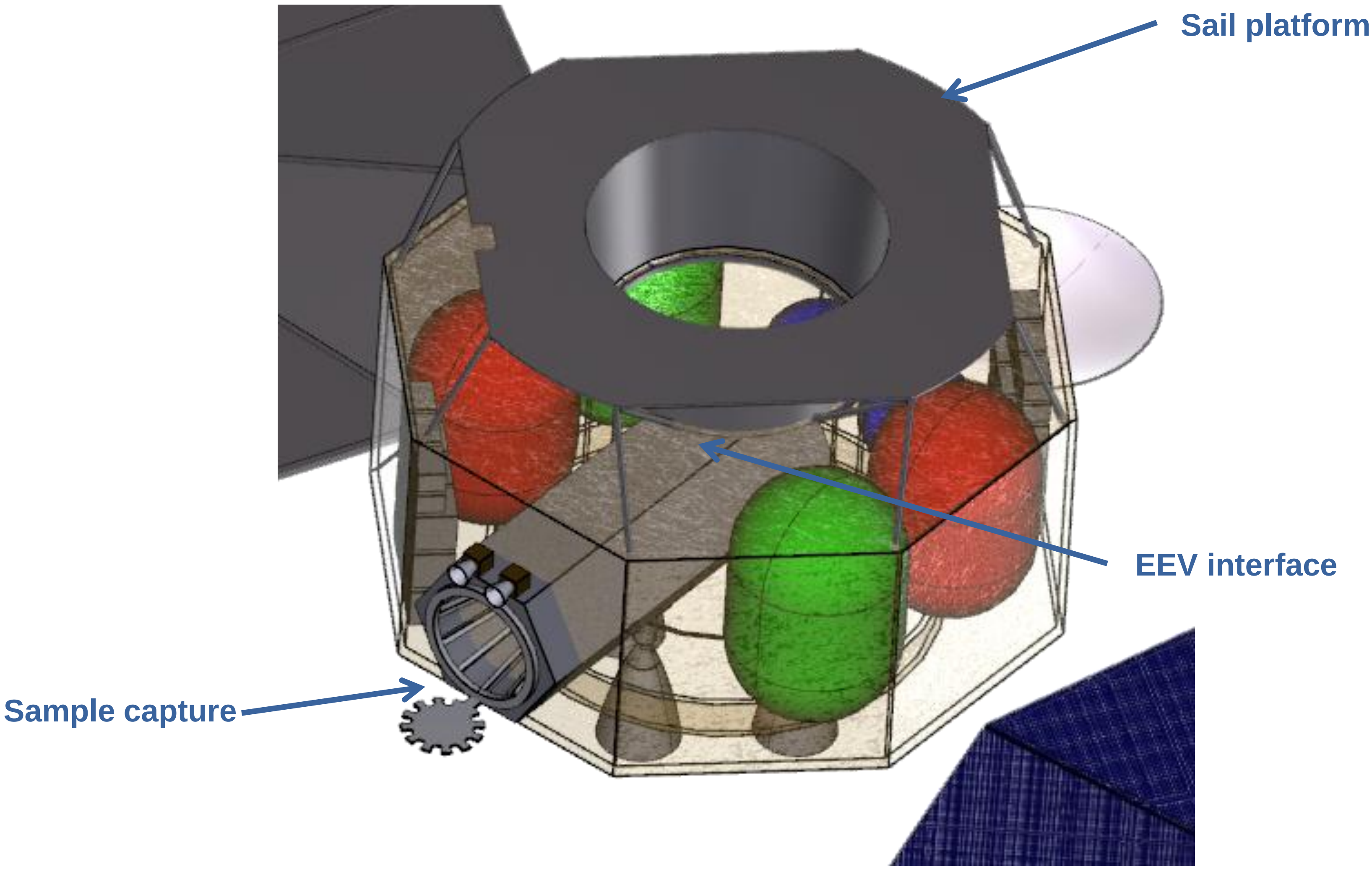
CONFIGURATION

Mike Baysinger

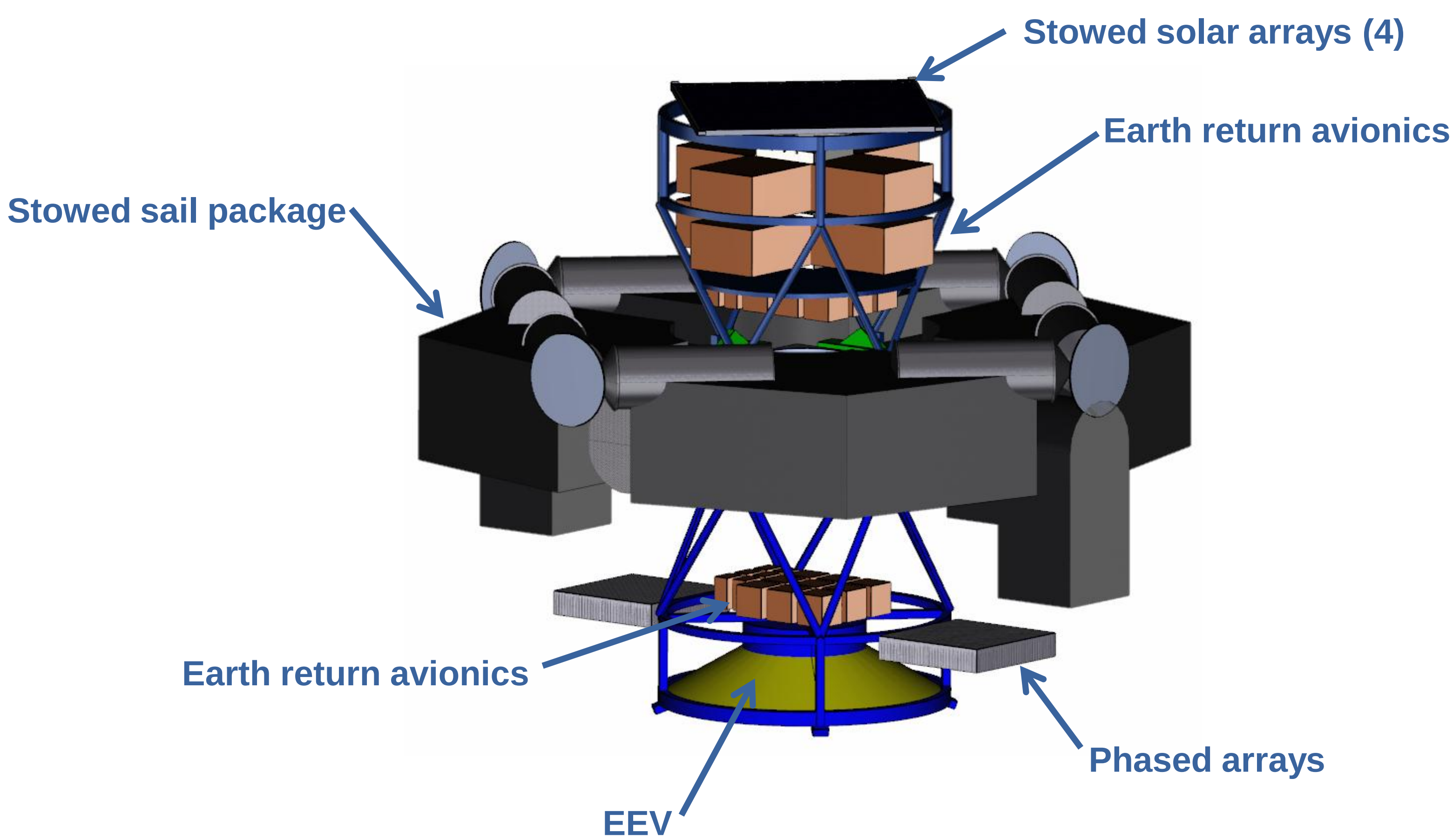
Orbiter


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

Orbiter

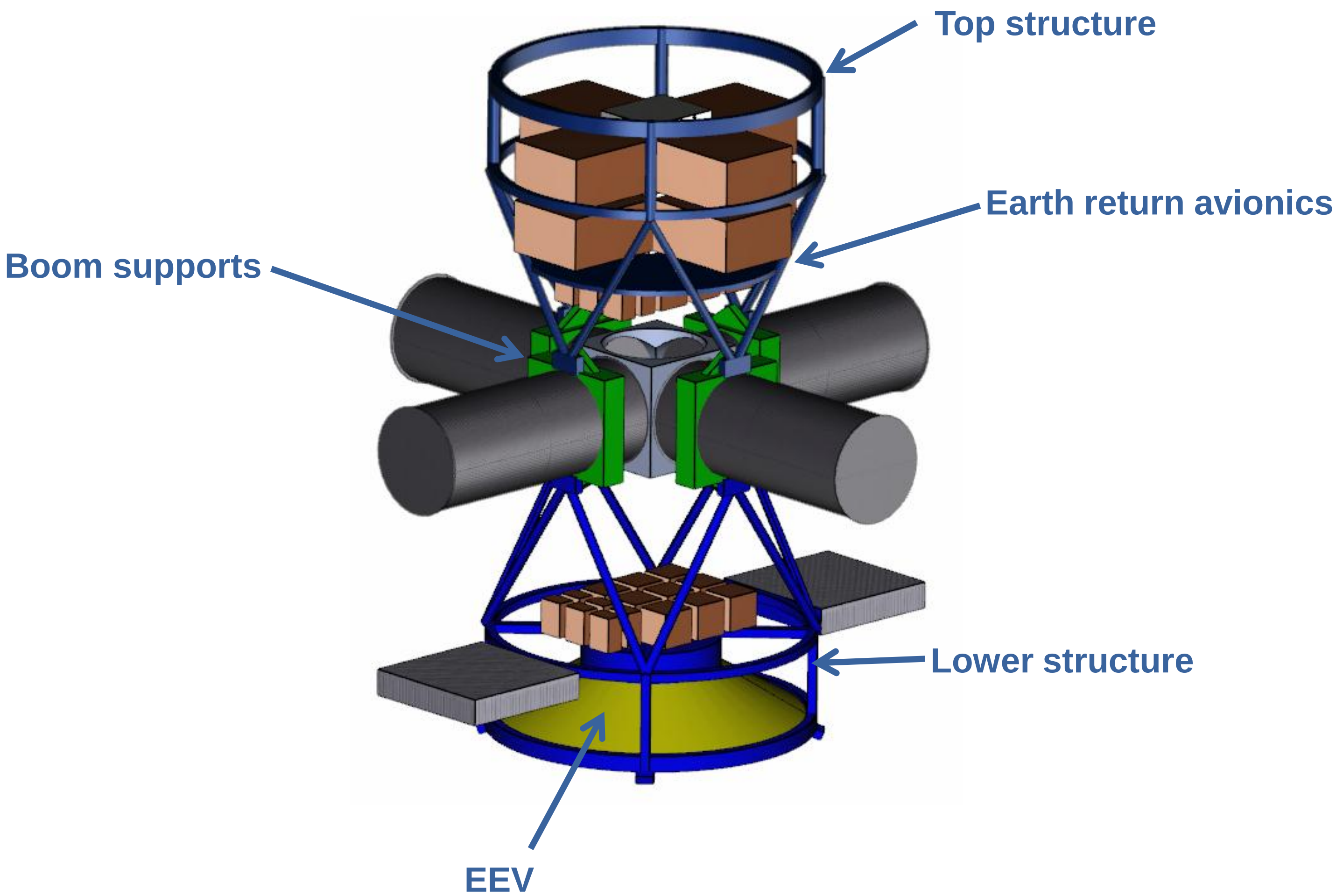


ERS



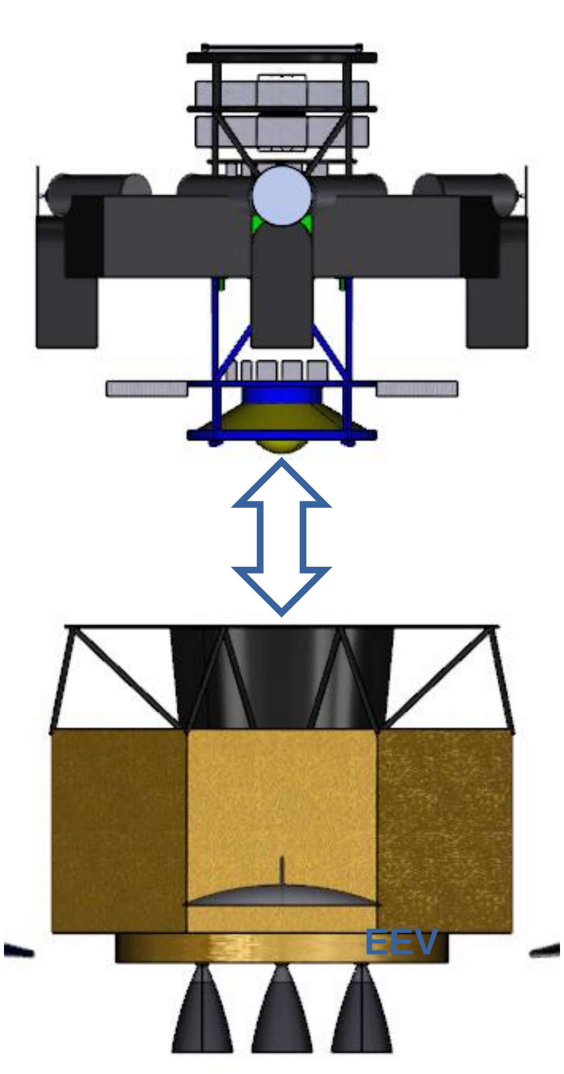
Stowed solar arrays (4)
Earth return avionics
Stowed sail package
Earth return avionics
Phased arrays
EEV

ERS

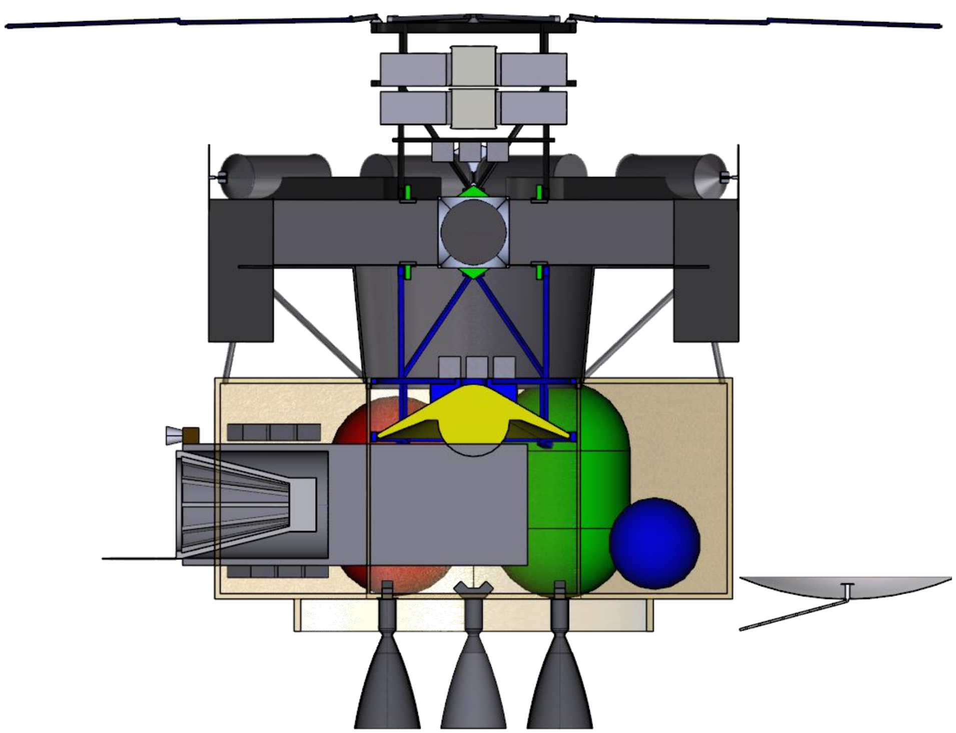


Top structure
Earth return avionics
Boom supports
Lower structure
EEV

Configuration

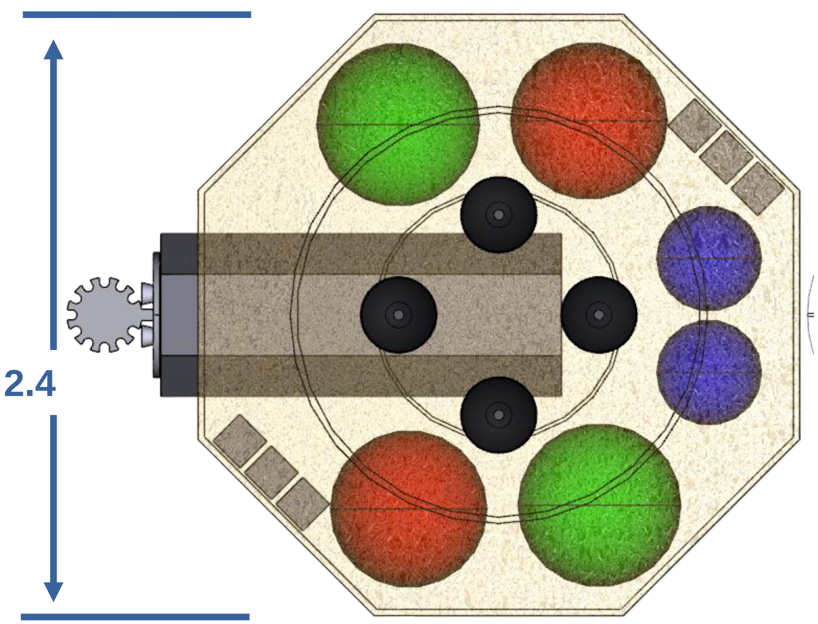


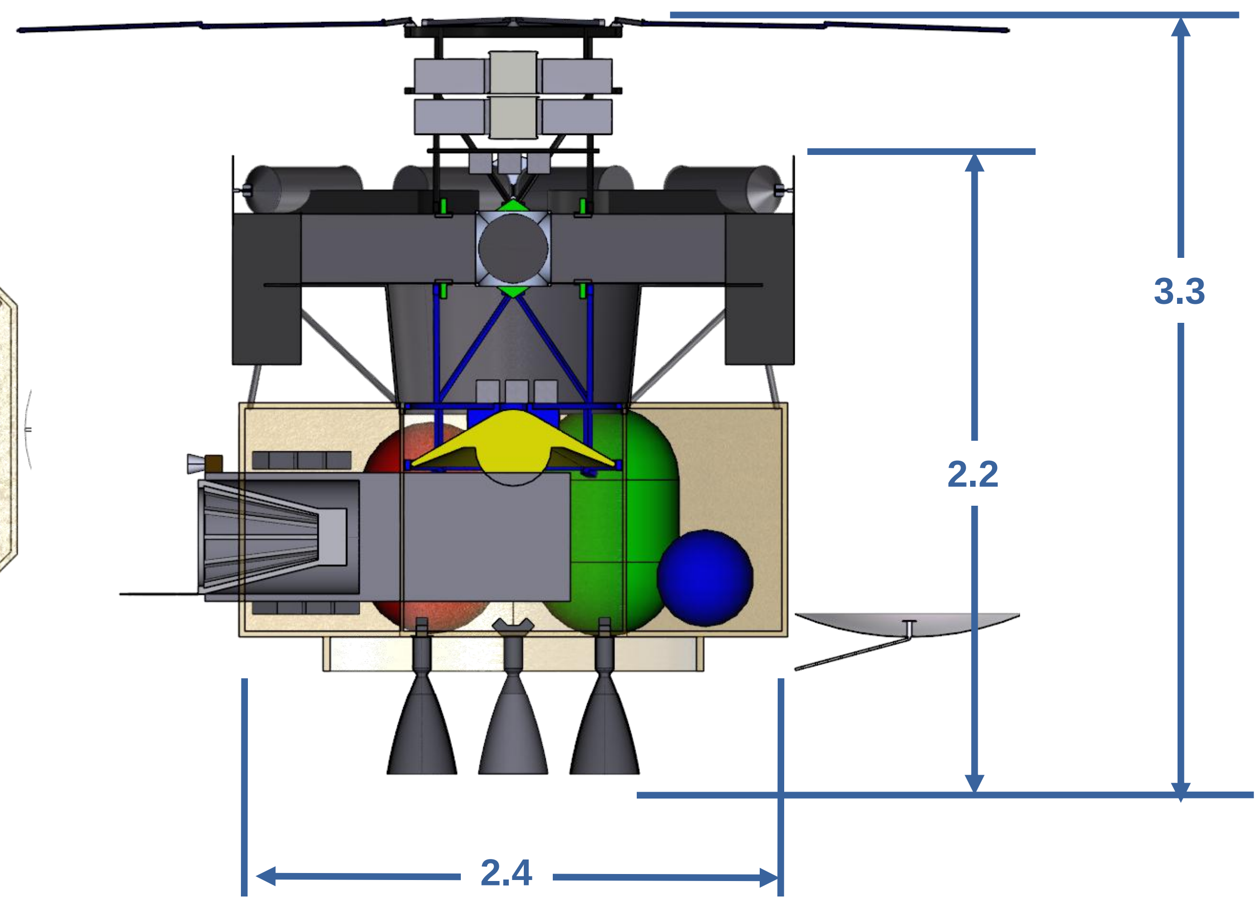
Sail with EEV



Orbiter

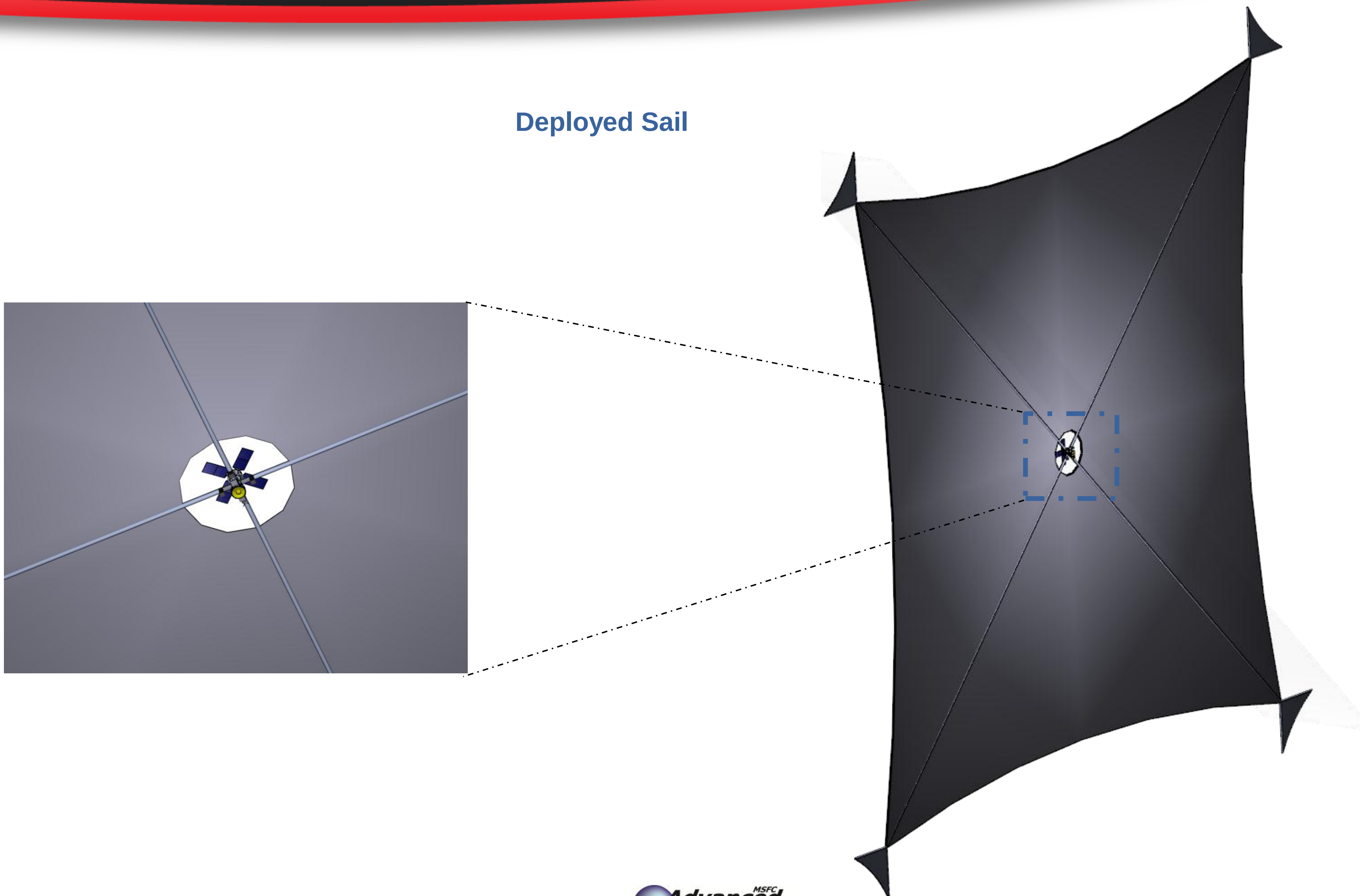
Configuration



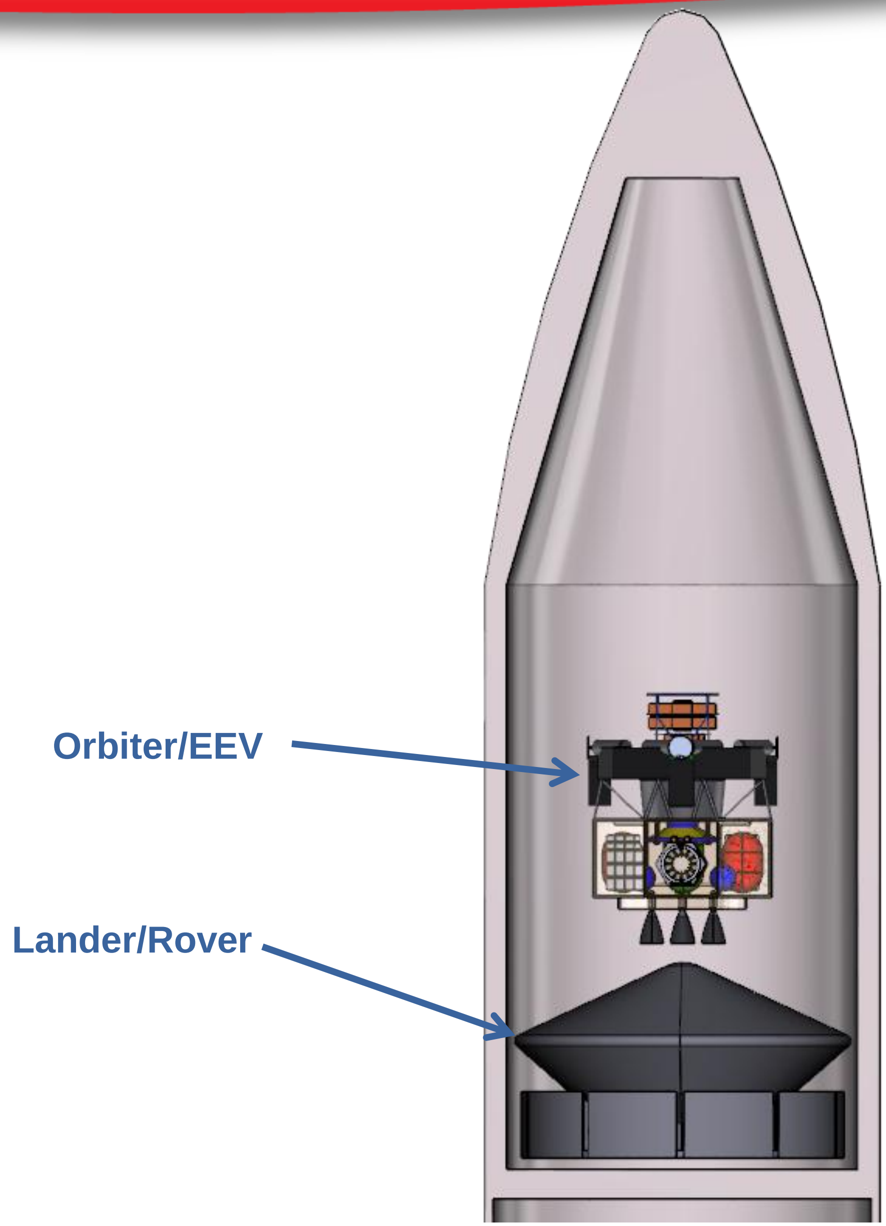


Deployed Sail

Deployed Sail



Atlas Shroud



STRUCTURES

Janie Miernik
Andy Wayne
Mark Davis

Ground Rules and Assumptions

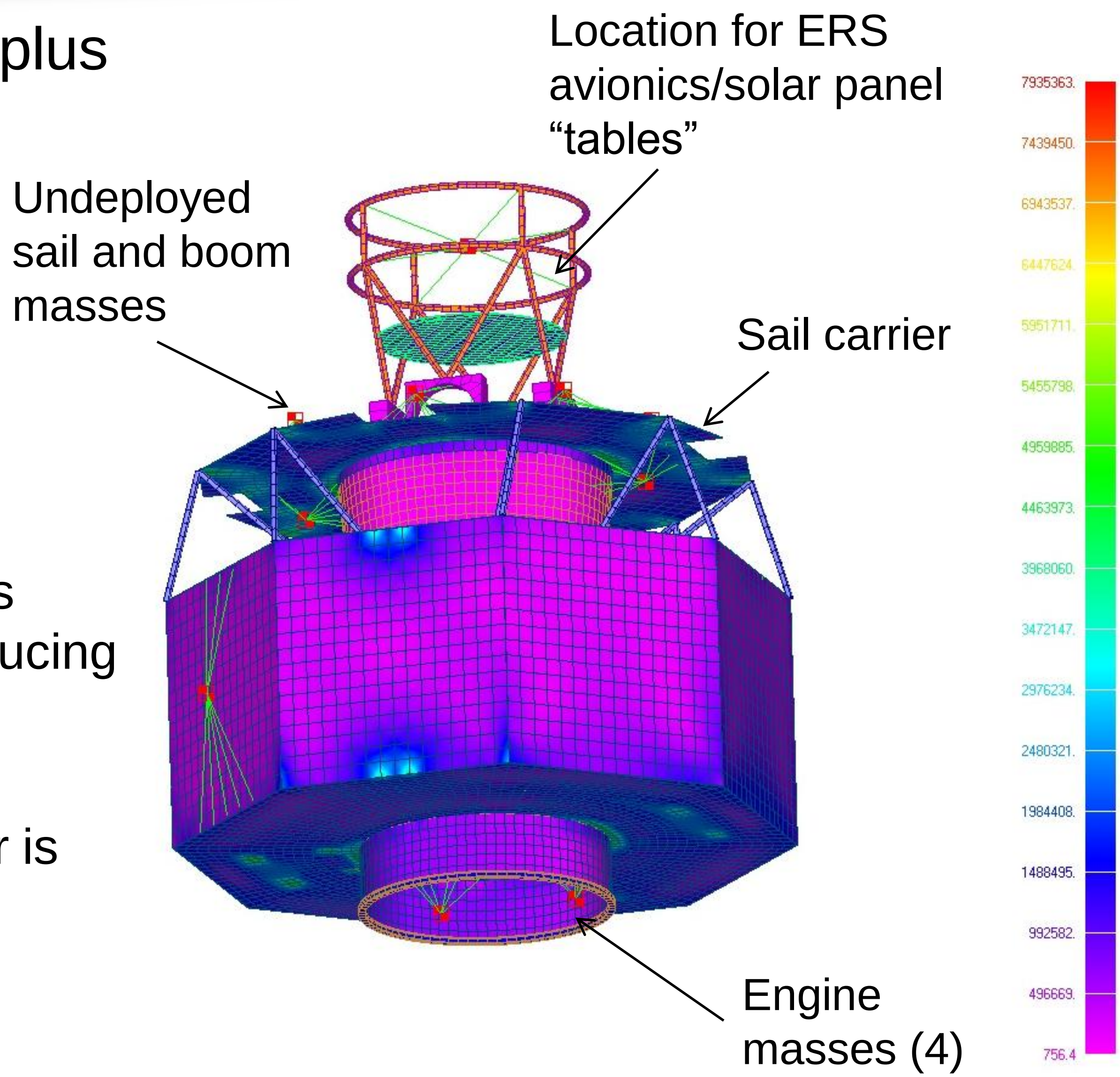
- Vehicle structure modeled and analyzed with Finite Element Model Analysis and Post-processing (FEMAP) software to produce the Finite Element Analysis (FEA) mesh.
- Hypersizer® FEA software optimized the structure for minimum mass.
- Load Cases: Atlas V static launch loads:
  - Axial: +6, -2g
  - Lateral: +/- 2g
- Factors of Safety:
  - Al 2024 Metallic Material: 1.4
- Deployed sail structural analysis not performed.
- Launch adapter not included in mass.

Design Methodology

- 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 retainment doors.
- Separation systems for ERS and EEV are Planetary Systems Corp (PSC) motorized Lightbands.

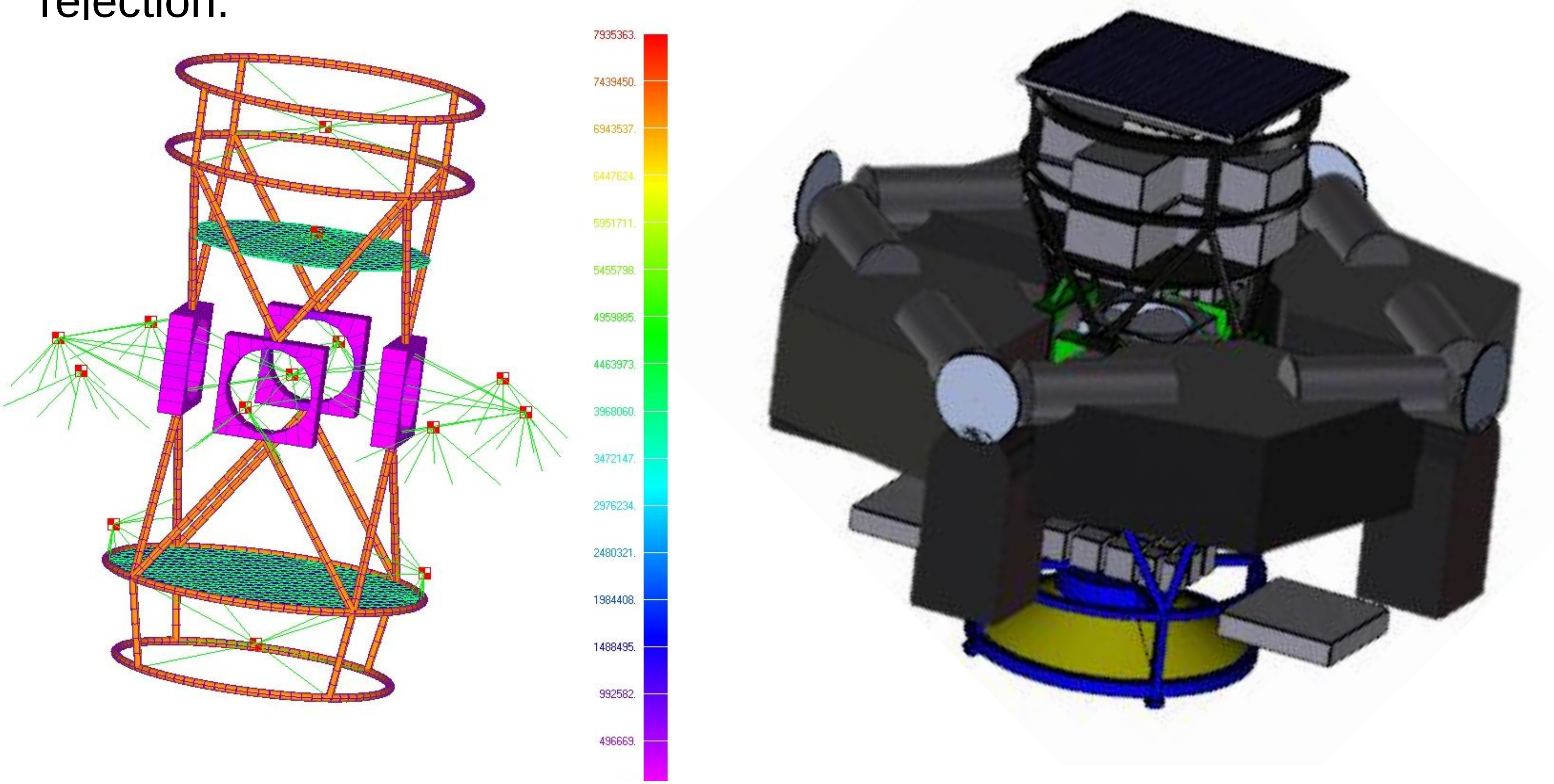
Analysis Results

- 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



Analysis Results

- FEA Results: 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	Basic Mass (kg)	Contingency (%)	Predicted Mass (kg)
1.0	**Structures**	139.45	23	**171.78**
1.1	Primary structure	94.00	25	117.50
1.2	Secondary structures	16.00	30	20.80
1.3	Solar array/aerobrake booms, drive assys	7.20	25	9.00
1.4	Aerobrake panel, balanced	15.20	10	16.72
1.5	ERS Separation adapter, Orbiter side	7.05	10	7.76

	Deploytech ERS	Basic Mass (kg)	Contingency (%)	Predicted Mass (kg)
1.0	**Structures**	38.02	26	**48.07**
1.1	Primary structure	19.00	25	23.75
1.2	Secondary structures	9.00	30	11.70
1.3	Solar array booms (4)	8.00	30	10.40
1.4	ERS Separation adapter	2.02	10	2.22
	Total Structures	**177.47**		**219.85**

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

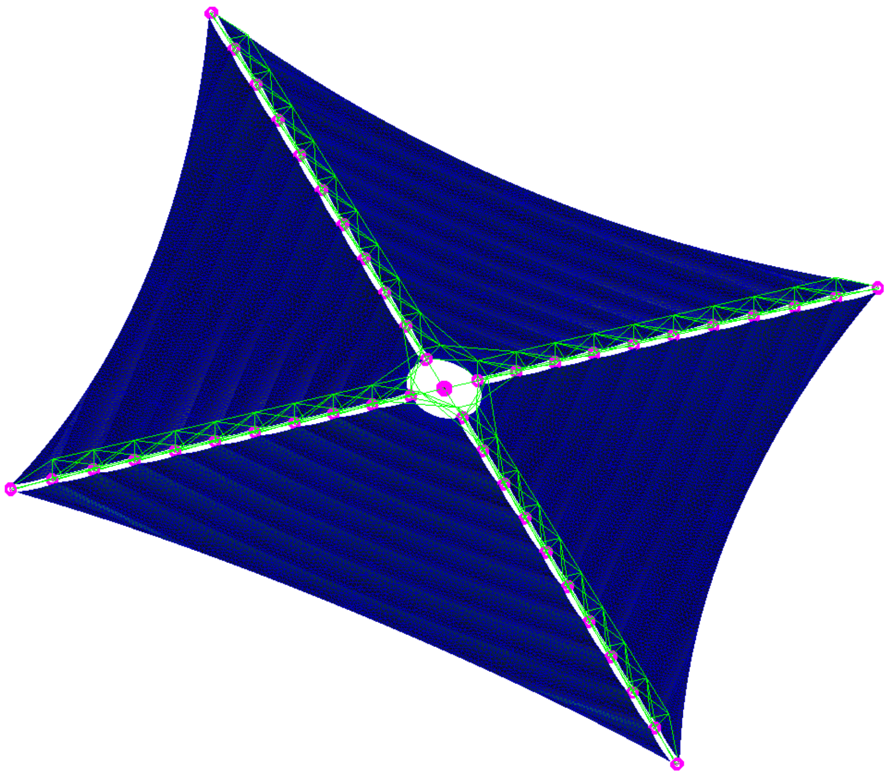
- Develop sail structural analysis to verify vendor results and that structural supports and vehicle attachments are sufficient.
- Size the ERS sail carrier and retainment doors in light composite.
- Divide orbiter structural panels into more properties to be able to optimize specific areas for component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

SAIL STRUCTURES

Mark Davis

Sail Finite Element Model

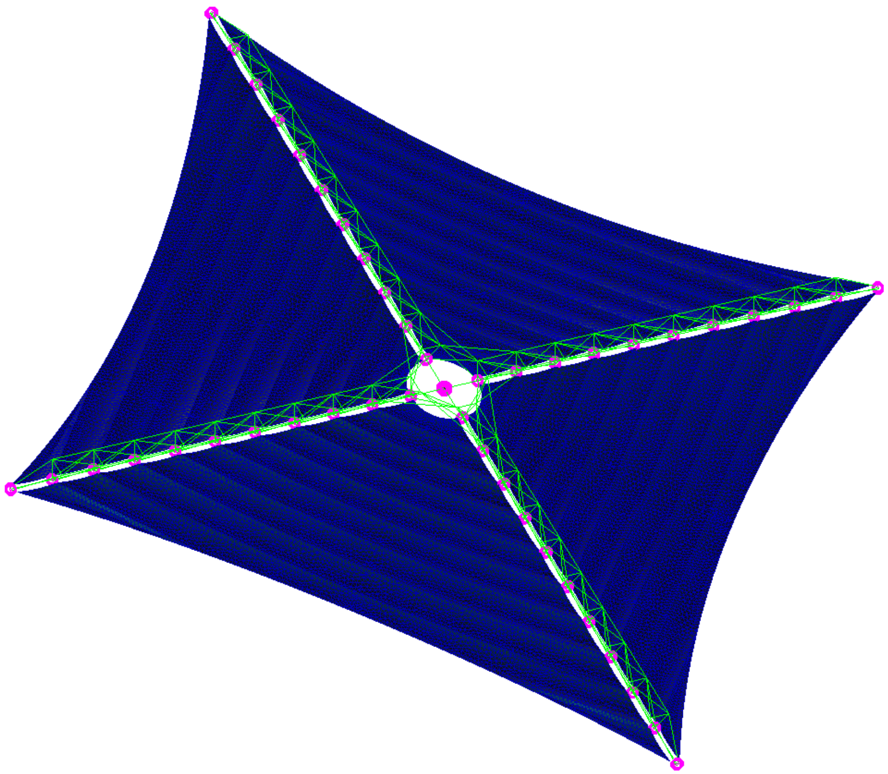
- The 20m ABAQUS finite element model was obtained from David Sleight (LaRC).
- Unfortunately the model in it's current form does not run due to some changes in ABAQUS version that prevents proper convergence of the solution.
- The modification of the 20m model to change the convergence criteria is ongoing, 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.



Sail Finite Element Model

- 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.



Sail Finite Element Model

- 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

Assumption	Value
Power Provision	Separate Solar Conversion Systems, Common Power Electronics
Solar Irradiance	1361 W / m^2
Max Distance from Sun	1.67 AU
Bus Voltage	28 V Nominal
Max Solar Incidence Angle	35°
Ground Reference	Common Ground Reference while connected

Power Required

Orbiter

Power Use Profile	Power (W)
Stand-By	183
Propulsive Maneuver	343
Rendezvous and Capture	489
Flight with Attitude Control	501

+ 30% Design Margin =
651 W

Earth Return Stage

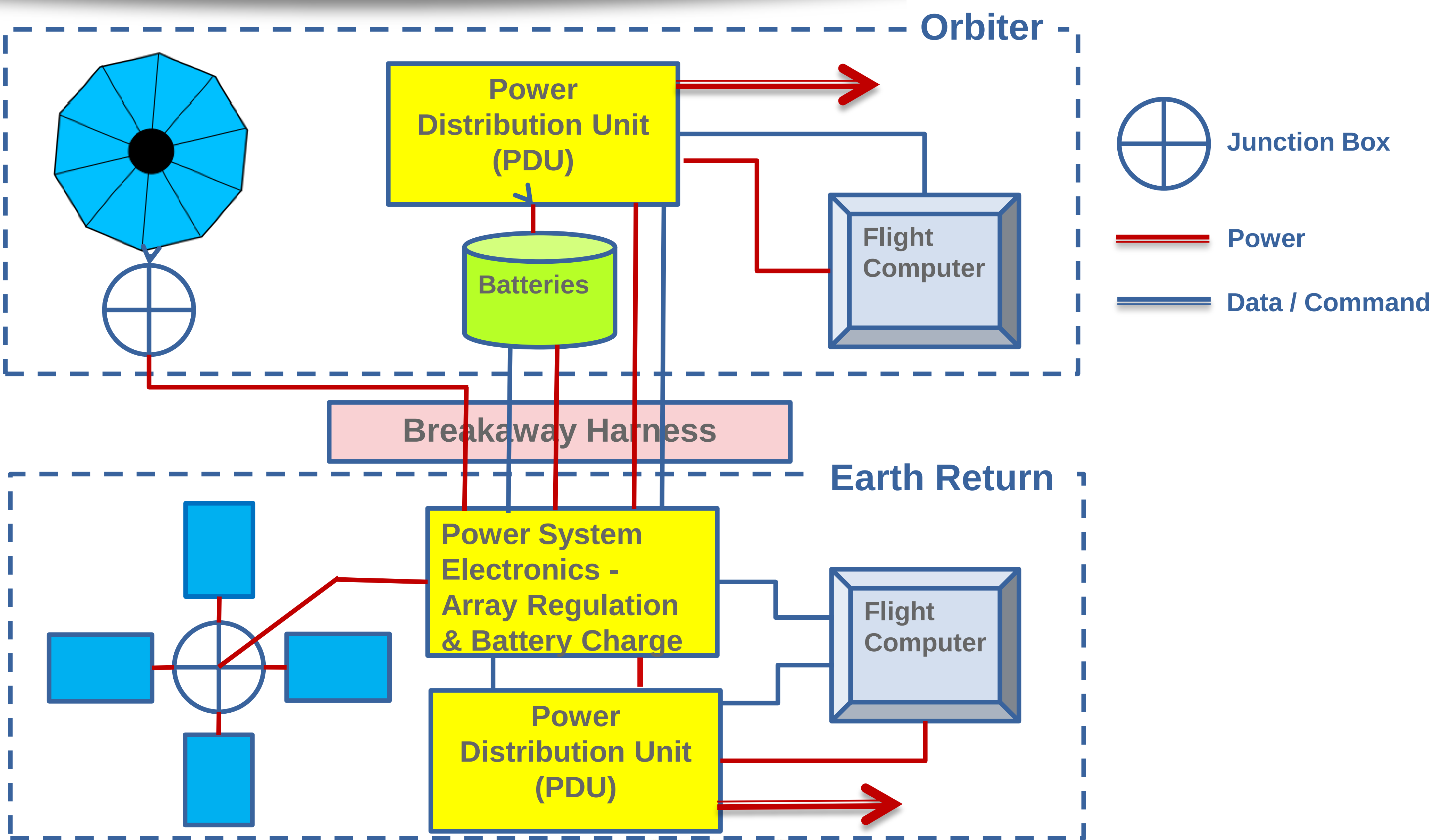
Power Use Profile	Power (W)
Keep-Alive	65
Stand-By	89
Normal Sail Flight	246

+ 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



System Breakdown and Mass Summary

Orbiter

Item	Unit Mass (kg)	Qty	Total Mass (kg)	MGA (%)	Predicted Mass (kg)
UltraFlex Solar Array Assembly	20.2	1	20.2	10%	22.2
Power Distribution Unit	11.5	1	11.5	5%	12.1
Solar Array Junction Box	1.3	1	1.3	5%	1.4
Secondary Battery	19.2	3	57.6	10%	63.4
Cables, Breakaway Harness	16.0	1	16.0	30%	20.8
Total Mass			**106.6**	**12%**	**119.8**

System Breakdown and Mass Summary

Earth Return Stage

Item	Unit Mass (kg)	Qty	Total Mass (kg)	MGA (%)	Predicted Mass (kg)
Rigid Solar Array Panel	3.2	4	12.8	10%	14.1
Integrated Power Systems Electronics Box (PSE)	6.6	1	6.6	5%	6.9
Power Distribution Unit	11.5	1	11.5	5%	12.1
Solar Array Junction Box	1.3	1	1.3	5%	1.4
Cables, Harness	4.5	1	4.5	30%	5.9
Total Mass			**36.7**	**10%**	**40.4**

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.

Forward Work

- 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 - GR&A

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)

Earth Return Stage - GR&A

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 – Approach and Rationale

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 baslined 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 - Approach and Rationale

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

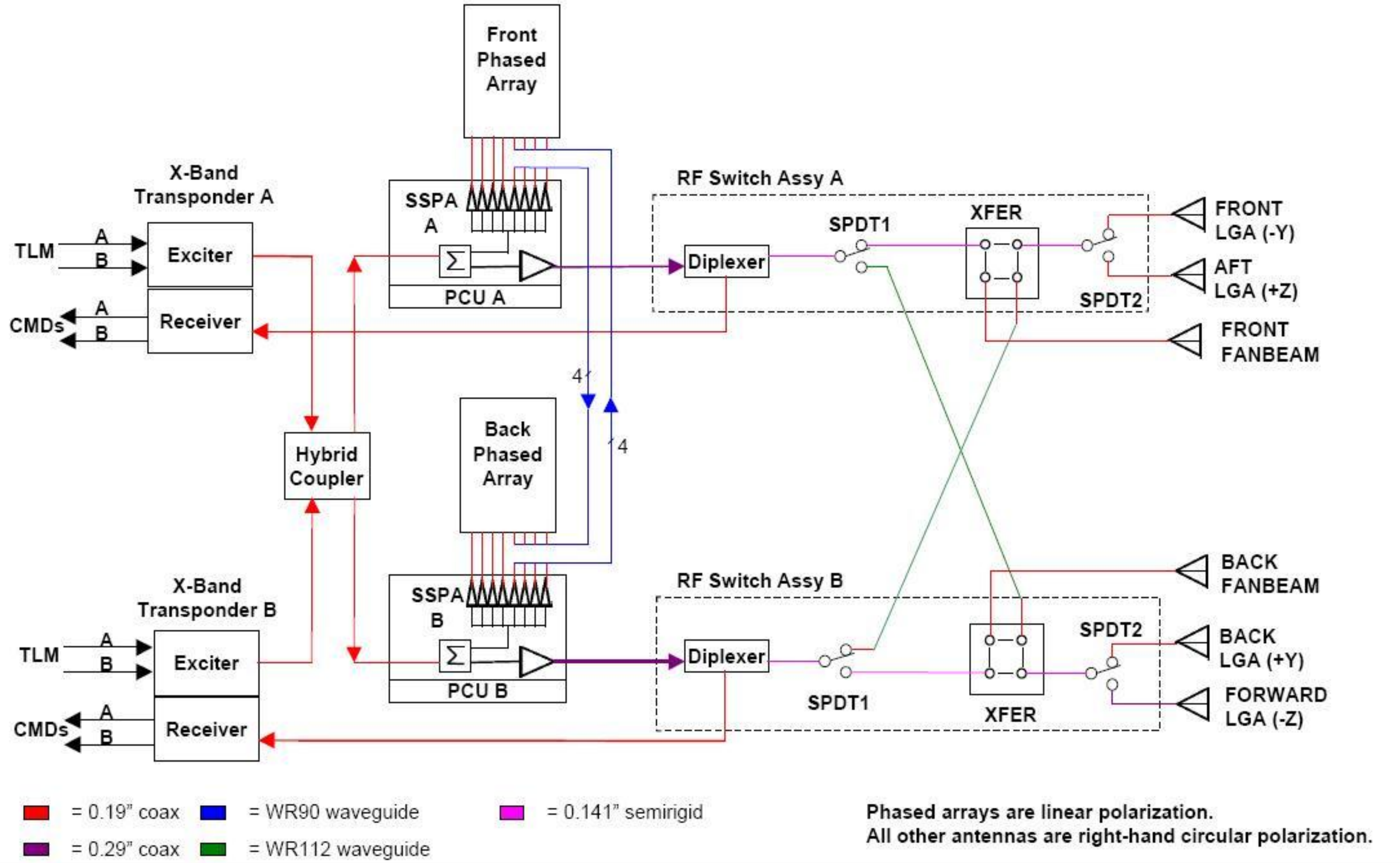
- For the ERS communications, the Messenger spacecraft communication system was baselined
  - This system allows over 2 kbps of data to be transmitted from Mercury to Earth
  - Since the distance from Mars to earth is nearly the same as Mercury to Earth (about 1.5 AU), it can be assumed that this system 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 - Approach and Rationale cont.

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

System Schematic



Phased arrays are linear polarization.
All other antennas are right-hand circular polarization.

Mass Summary

Orbiter Avionics Mass Summary

Subsystem	Basic Mass (kg)	Contingency (%)	Predicted Mass (kg)
Command and Data Handling	48.6	20%	58.44
Guidance Navigation and Control	11.96	3%	12.32
Communications	29.15	6%	30.76
Vehicle Systems	19.70	13%	22.33
Total	109.41		123.85

ERS Avionics Mass Summary

Subsystem	Basic Mass (kg)	Contingency (%)	Predicted Mass (kg)
Command and Data Handling	20.30	11%	22.48
Guidance Navigation and Control	5.10	3%	5.26
Communications	24.61	3%	25.35
Vehicle Systems	1.80	3%	1.85
Total	51.81		54.94

Avionics Conclusion

- 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.

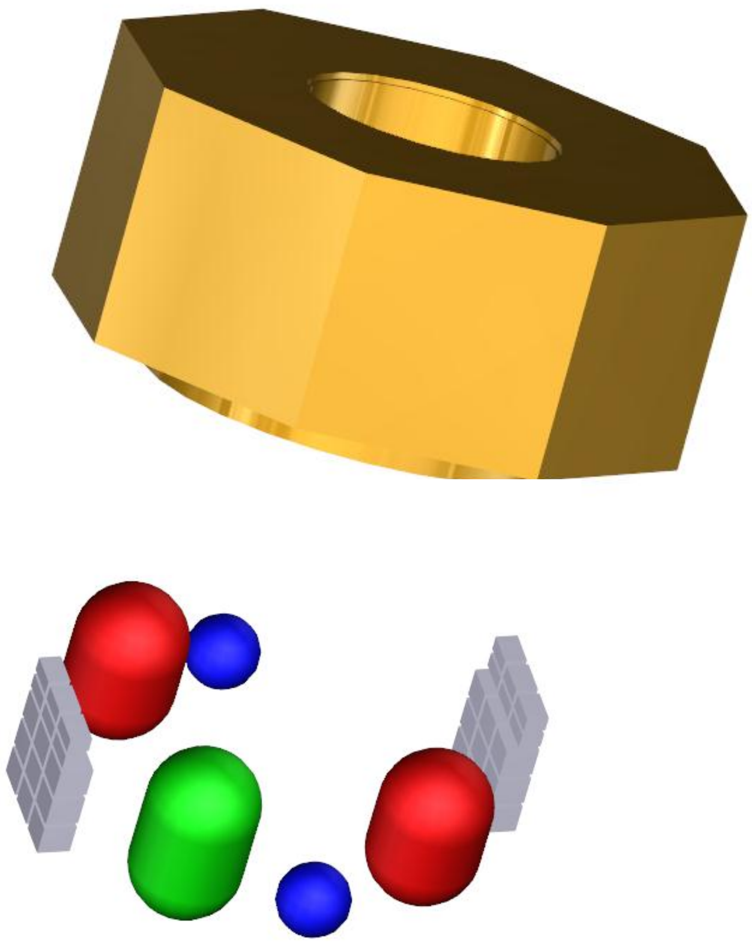
THERMAL

Linda Hornsby

Methodology

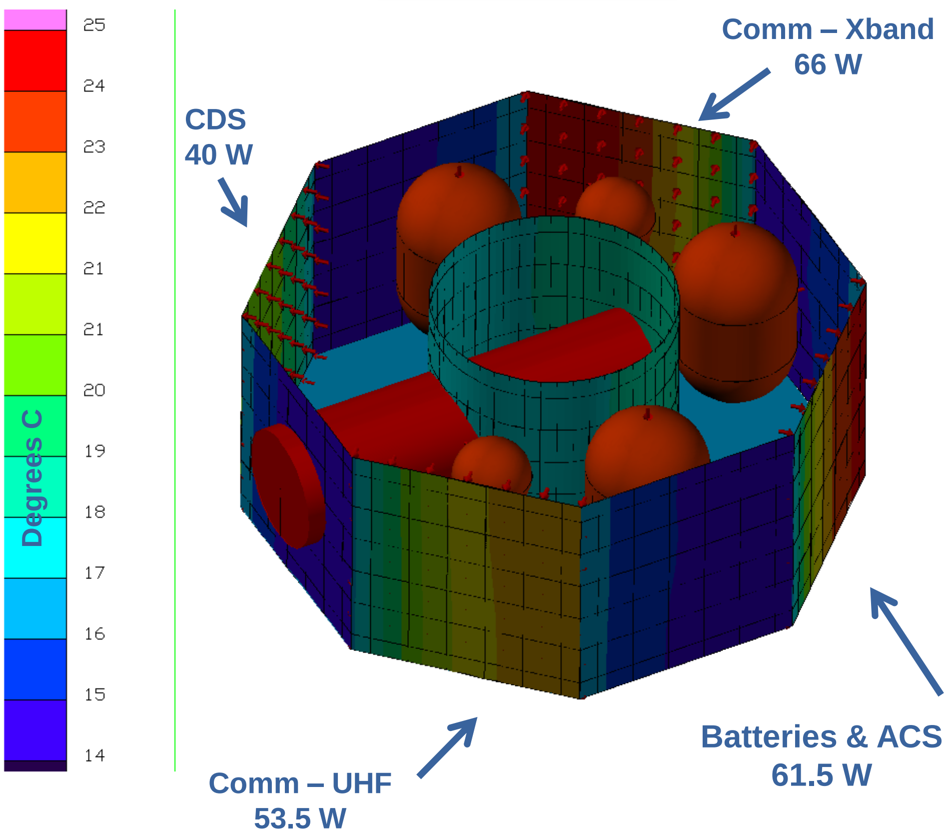
- Thermal Desktop® model of orbiter & ERS bus developed to assess subsystems interface temperatures
- Structural panels double as radiative surfaces.
- Structure modeled as aluminum, panel thickness consistent with structural design
- MLI blankets are assumed to cover all spacecraft surfaces
- Subsystems 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

Orbiter Bus Analysis

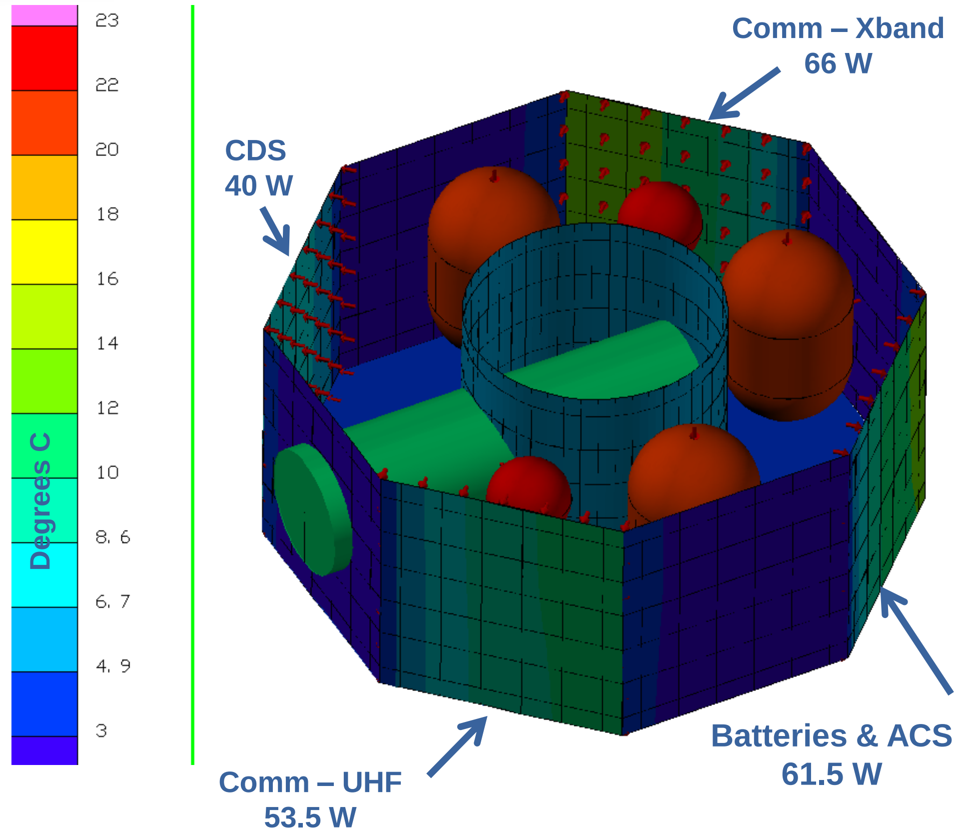


- Subsystems avionics and power systems are mounted to the exterior structural panels of the bus.
- Chemical propulsion tanks are located inside the bus.

Orbiter Analysis Results



Mars Orbit



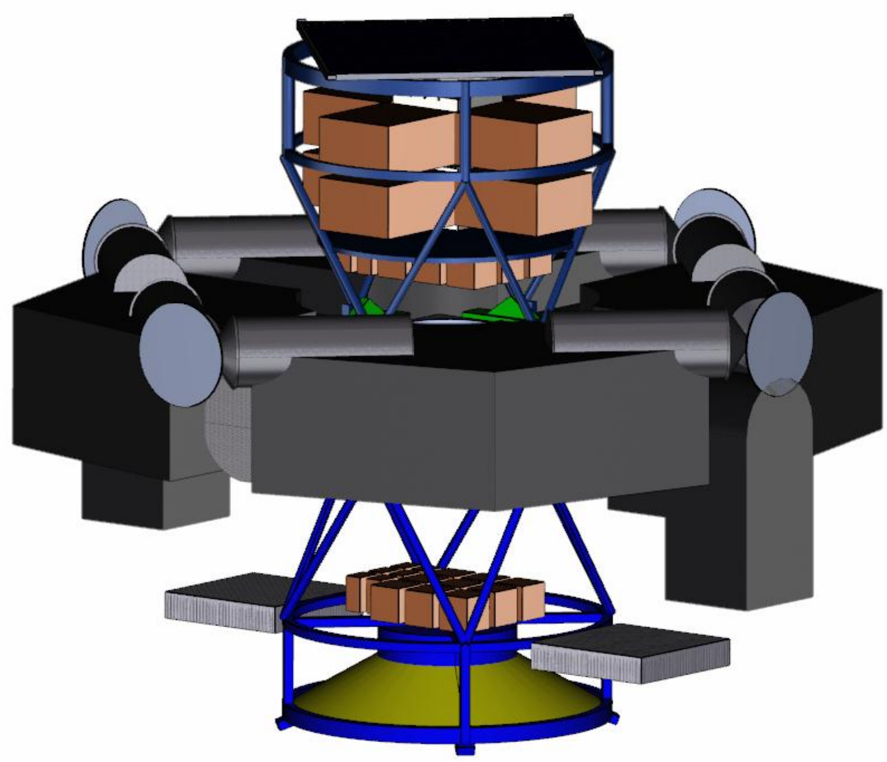
Mars Transit
(11W tank heater power)

Orbiter Heat Dissipation & Operating Temperatures

	Total Heat Dissipation	Operating Temp Range
ACS (IMU Assembly)	24 W	-40 C to 85 C
Command & Data System	40 W	-40 C to 85 C
Communication System (UHF)	53.5 W	-40 C to 85 C
Communication System (X-band)	66 W	-40 C to 85 C
Batteries (3)	37.5 W	0 C to 45 C
Propellant and Pressurant Tanks (Heaters)	11 W	10 C to 30 C
Total	232 W	

All predicted interface temperatures are within acceptable range of 0 to 30 degrees C for the trip to Mars and Mars orbit cases analyzed.

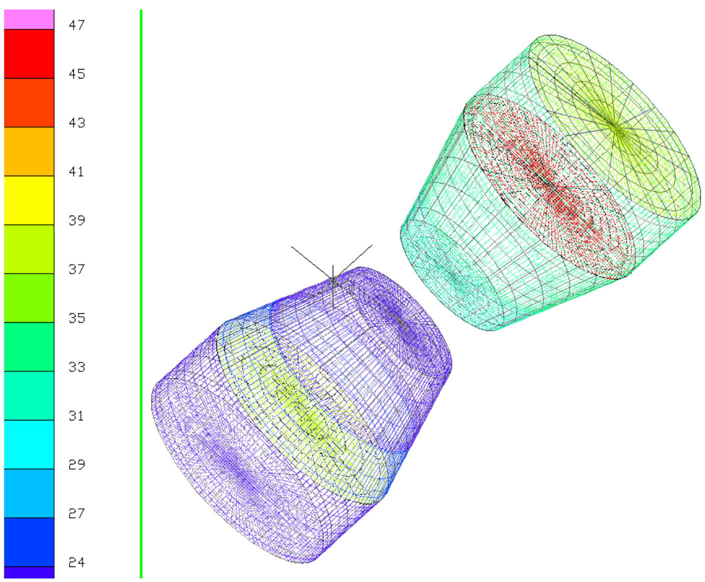
ERS Analysis



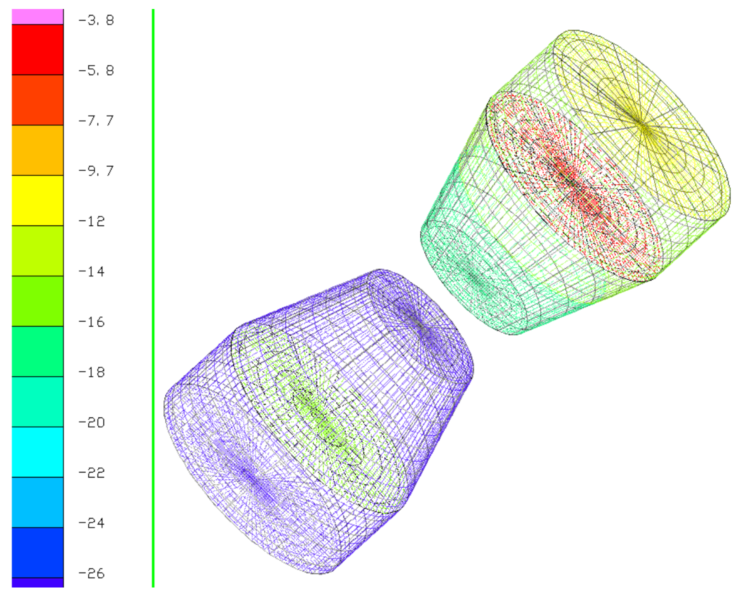
- The Earth Return Stage is designed with multiple avionics shelves.
- Subsystems equipment heat loads are equally distributed on the shelves.
- ERS structure is wrapped with MLI.

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

	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	

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

	Material	Absorptivity	Emissivity
Spacecraft Bus & ERS Internal Surfaces	Black Annodized	.9	.9
Orbiter & ERS External Surfaces (EOL)	EOL Silverized Teflon MLI ε*=.04 to .06	.25	.87
Propulsion & Pressurant Tanks	MLI ε*=.02	.6	.09

Thermal Control Mass Summary

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

	Deploytech ERS	Basic Mass (kg)	Contingency (%)	Predicted Mass (kg)
3.0	Thermal	16.82	30	21.87
	3.1 Insulation	10.40	30	13.52
	3.2 Heaters/Thermistors	0.70	30	0.91
	3.3 Paint/Coatings	3.12	30	4.06
	3.4 Misc	2.60	30	3.38

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

Mass Properties - Orbiter

		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 TOTAL		**495.27**	**19%**	**96.04**	**591.31**
6.0	Non-Propellant Fluids	26.46			26.46
7.0	Payload	149	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**

Mass Properties - ERS

		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**

Mass Properties – Total Launch Mass

	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**

STUDY CONCLUSIONS

Launch Mass Comparison

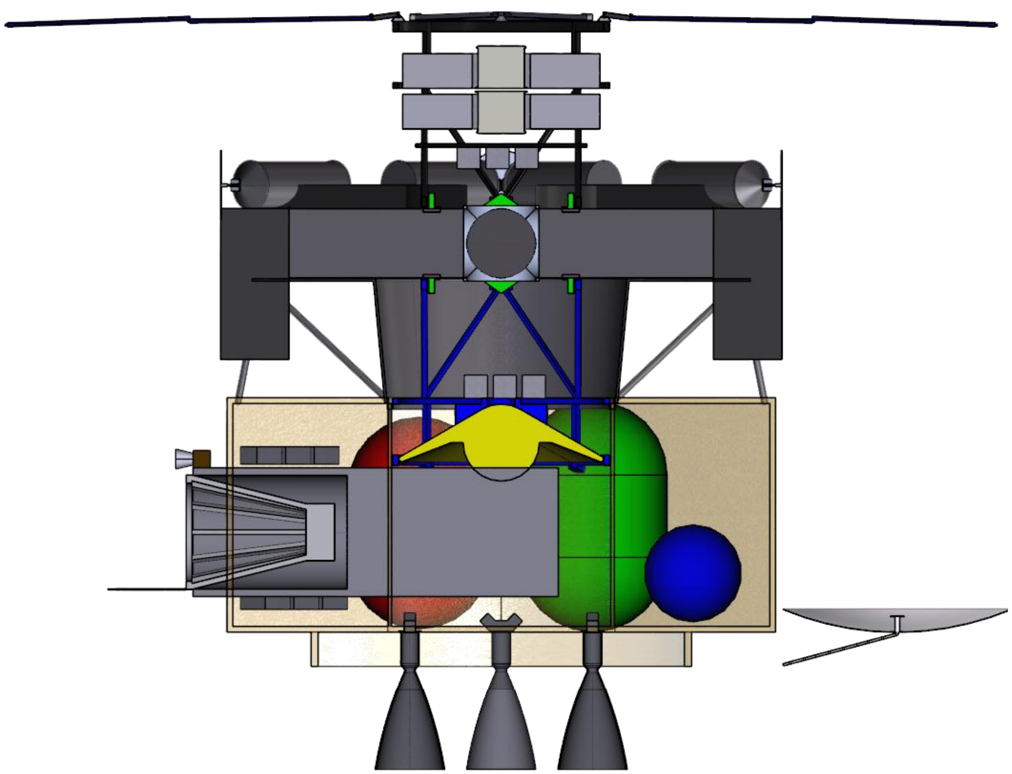
- Primary objective to reduce baseline mission from three to two launches
- Required to maintain baseline mission objectives for all three elements, only redesigning the Earth return propulsion system
- ACO MSR Orbiter and ERS had a launch mass too large to be packaged on the Max-C Rover or Lander launch vehicles

	Predicted Launch Mass (kg)	Launch Vehicle	Launch Capability (kg)	Contingency (kg)
Max-C Rover	4457.4*	Atlas V 531	4980	522.6
MSR Orbiter	3270	Atlas V 551	4770	1500
Lander and Ascent Vehicle	4668	Atlas V 551	5130	462
ACO MSR Orbiter and ERS	1947.49	--	--	--

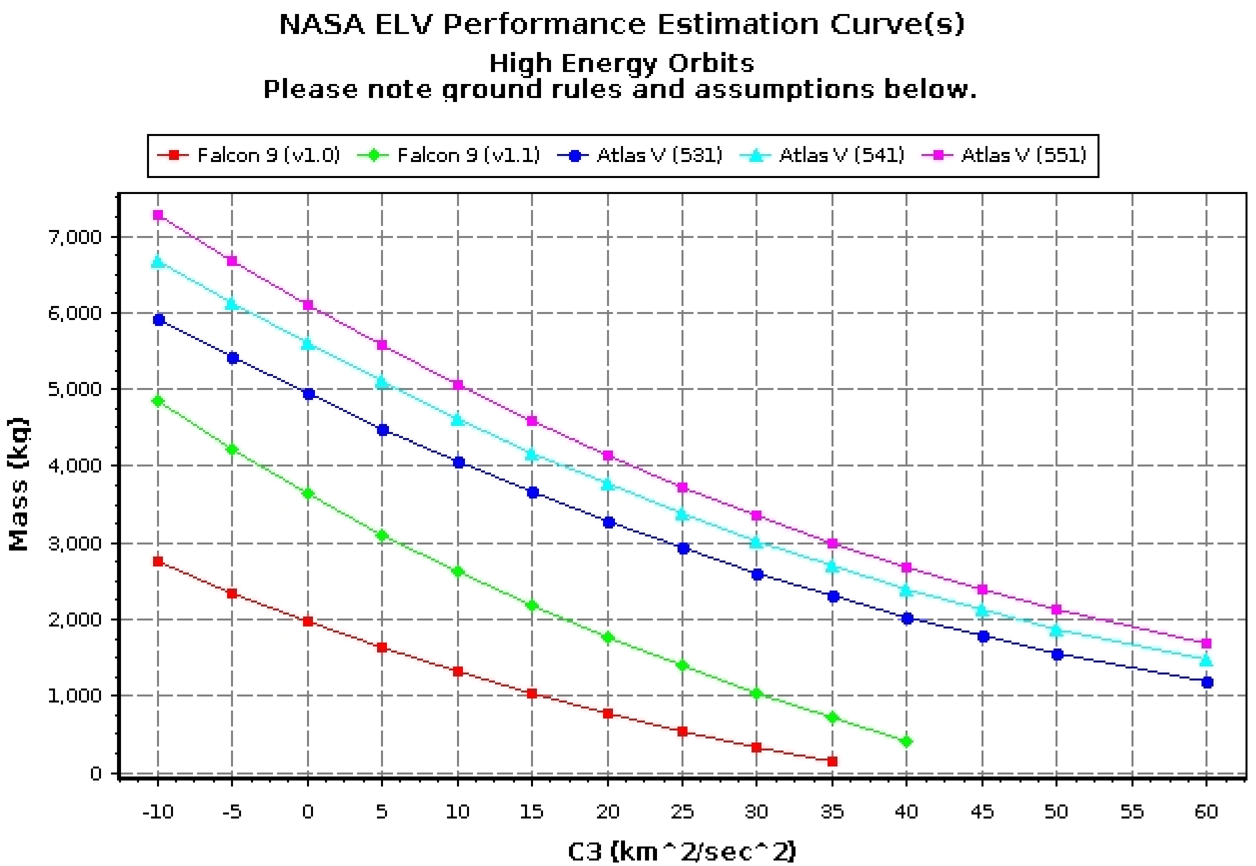
*** 300kg was bookmarked in the baseline architecture as a placeholder for ExoMars**

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



Forward Work



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

Forward Work

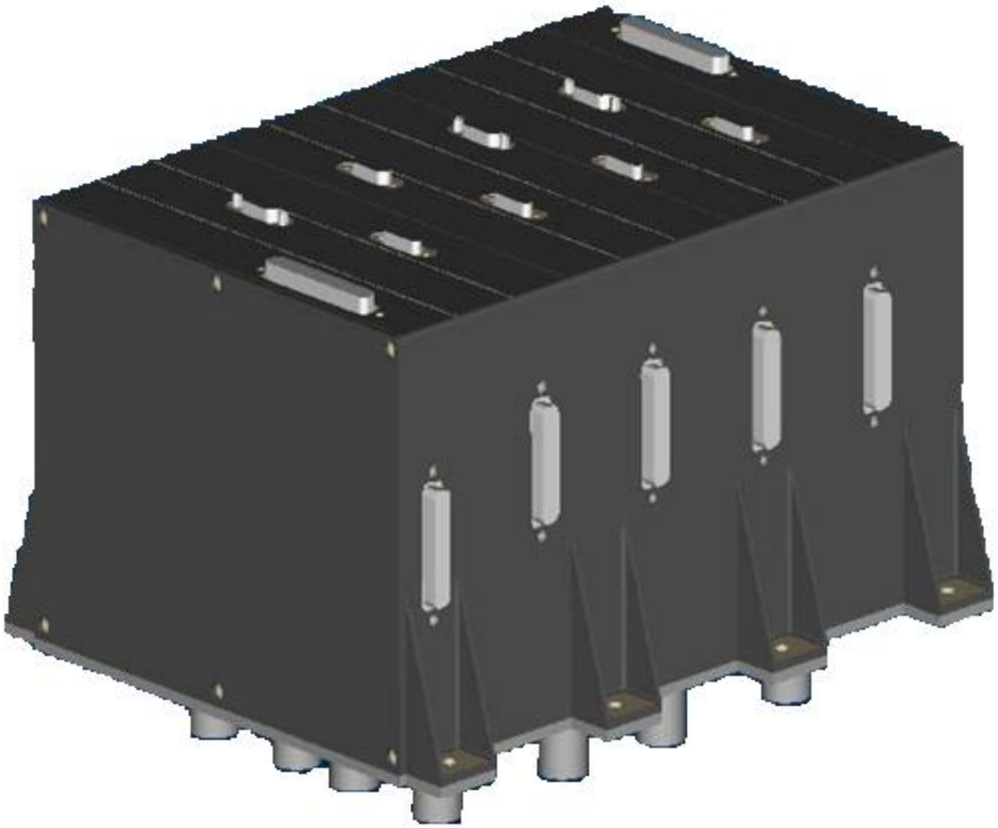
- Revisit 2 launch configuration by adjusting the GR&A
  - Reduce Max-C and Orbiter mass
  - Eliminate Max-C Rover
  - Re-evaluate launch vehicle options
  - Use solar sail for all interplanetary primary propulsion

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

Estimated values from NLSII performance values

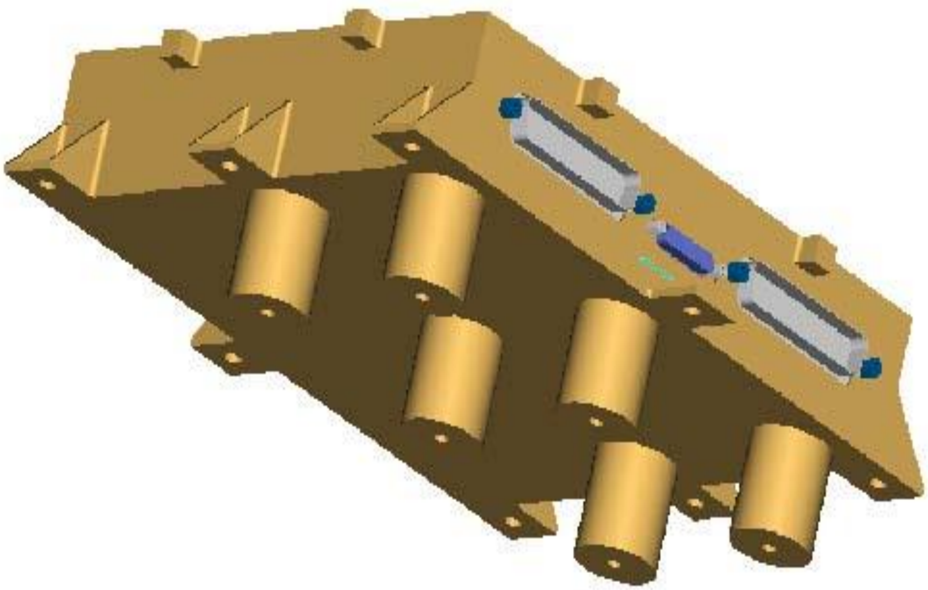
BACKUP

Power System Electronics Box



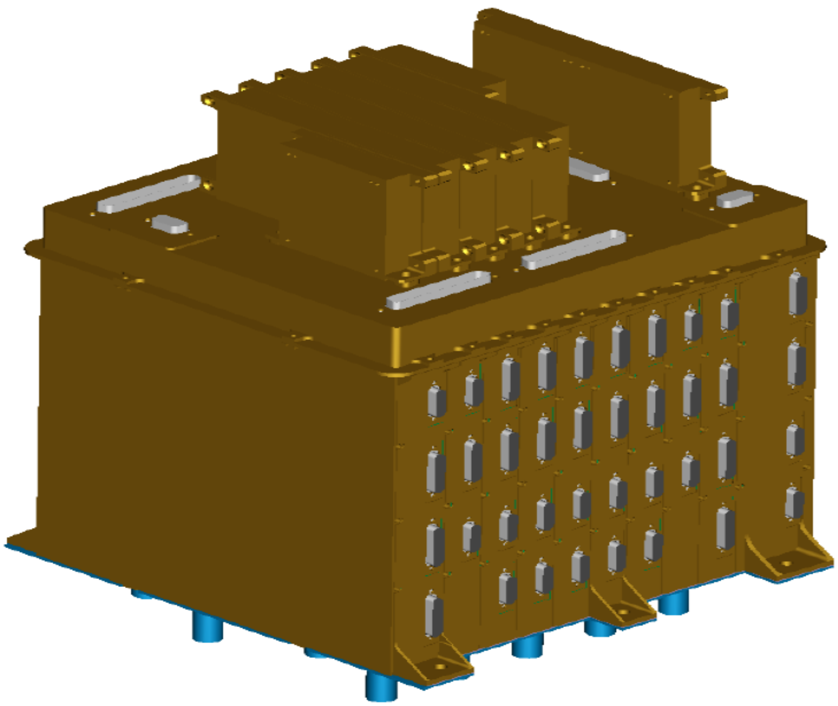
- MESSENGER Heritage
- Solar array regulation
- Peak Power Tracking (PPT)
- Battery Charge Control
- Direct redundant serial interface to Power Distribution Unit

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