Solar Probe Plus
Mission Concept Review

September 29, 2009—October 1, 2009

This chart package has been approved for public release. Some charts have been removed due to the sensitive nature of the information they contain.

The Johns Hopkins University
APPLIED PHYSICS LABORATORY

Solar Probe Plus
Mission Concept Review

Introduction and Guidance

Andrew Dantzler
Project Manager

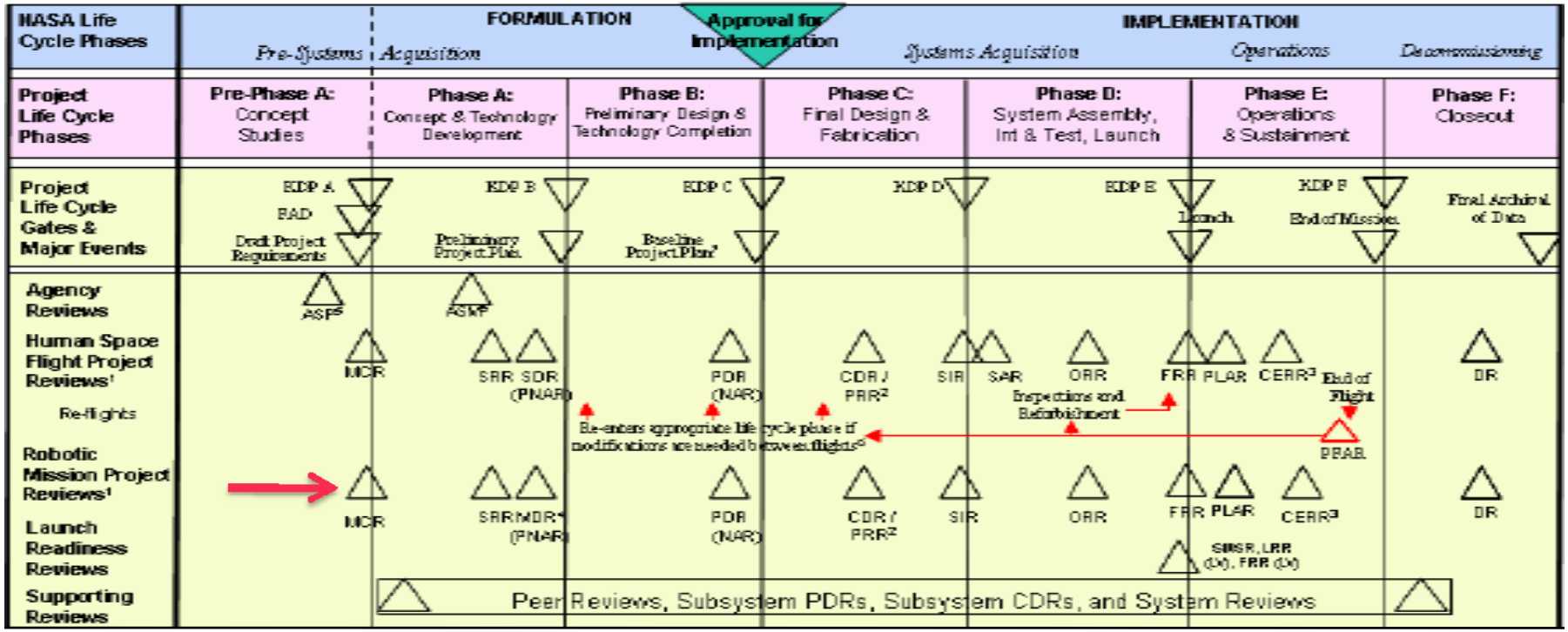
The Johns Hopkins University
APPLIED PHYSICS LABORATORY

General MCR Assumptions and Guidance

- **The MCR is required per NPR 7120.5D (NASA Space Flight Program and Project Management Requirements)**
  - **Guidance for the MCR is provided in NPR 7123.1 (Systems Engineering Procedural Requirements)**

MCR in the Project Life Cycle

Leads to KDP-A and the beginning of Phase A



SPP is in Pre-Phase A

"Purpose: During Pre-Phase A, a pre-project team studies a broad range of mission concepts that contribute to program and Mission Directorate goals and objectives. These advanced studies, along with interactions with customers and other potential stakeholders, help the team to identify promising mission concept(s) and draft project-level requirements. The team also identifies potential technology needs (based on the best mission concepts) and assesses the gaps between such needs and current and planned technology readiness levels. These activities are focused toward a Mission Concept Review and KDP A."

---NPR 7120.5D

Entrance Criteria - NPR 7123.1

- A Project is ready for MCR when the following is developed:
  - Mission goals and objectives
  - Analysis of alternative concepts to show at least one is feasible
  - Concept of operations
  - Preliminary mission descope options
  - Preliminary risk assessment including technologies and associated risk management/mitigation strategy
  - Conceptual test and evaluation strategy
  - Preliminary technical plans to achieve next phase (preliminary SEMP)
  - Measures of effectiveness and measures of performance
  - Conceptual life-cycle support strategies (logistics, manufacturing, operation, etc)

All criteria are met with the March 2008 Mission Study Report and delivered Pre-Phase A technical reports

Success Criteria - NPR 7123.1

- Mission objectives are clearly defined and stated, and are unambiguous and internally consistent.
- The preliminary set of requirements satisfactorily provides a system which will meet the mission objectives
- The mission is feasible. A solution has been identified which is technically feasible. A rough cost estimate is within acceptable cost range.
- The concept evaluation to be used in candidate systems evaluation have been identified and prioritized.
- The need for the mission has been clearly identified (Formulation Authorization Document--FAD).
- The cost and schedule estimates are credible.
- A technical search was done to identify existing assets or products that could satisfy the mission or parts of the mission.
- Technical planning is sufficient to proceed to the next phase.
- Risk and mitigation strategies have been identified and are acceptable.

MCR Strategy

As per the intent of Pre-Phase A, we will not be presenting a “point design” or “baseline design”.

The design will necessarily remain as flexible as possible (when mission constraints allow) until payload selection is final.

For the MCR, we will present a “Reference Design” used in trades, along with options and trades to be done in Phase A/B.

Review Ground Rules

- ***<u>THIS IS THE MISSION CONCEPT REVIEW, not a detailed design or cost review.</u>***
  - ***While we have substantial amounts of data to back up all presentations, it is not appropriate for this review to get into detailed design or cost discussions during the presentations***
- **Candidate Requests for Action can be submitted by anyone present**
  - **Please seek out the presenter for clarification or additional information before submitting a candidate RFA**
  - **For this Review, the Review Chair and Project Manager will meet to determine which submittals are to be elevated to action items (with due dates) and which are for information only**
- **Review Chair and Project Manager can table a discussion if necessary**

Standing Review Board Membership

Chair	Paul Gilbert	MSFC
Review Manager	Charles Taylor	IPAO
Software/Fault Management/I&T	Steve Scott	GSFC
Solar Arrays /Power System	Tom Kerslake	GRC
Mission Design	Dennis Brynes	(JPL)
Thermal	Ted Michalek	GSFC
S&MA/Reliability	Mike Delmont	GSFC
Thermal Shield TPS	Keith Bowman	Civ USAF AFMCAFRL/RB
Mechanical Systems/Structures	Ken Hinkle	GSFC
Cost and Schedule Analysis	Paul G. Morris, Ph.D	IPAO
Mission System Management (Flight Ops /Ground Segment)	Dave Linick	(JPL)
Avionics/Science Instruments	Steve Battel	Battel Engineering
Science and Solar Environments	Charles Kankelborg	(MSU)

Agenda

Day 1: September 29, 2009

08:30	Intro/Kickoff - Dantzler (30 min)
09:00	Science - Decker (30 min)
09:30	Mission Overview - Kinnison (30 min)
10:00	Mission Design – Guo (30 min)
10:40	Break
10:55	Environments - Decker (30 min)
11:25	Flight Systems Overview 1 – Lockwood (45 min)
12:10	Lunch
01:10	Flight Systems Overview 2 – Lockwood (45 min)
01:55	TPS - Mehoke (45 min)
02:50	Break
03:05	Solar Array Cooling System - Ercol (45 min)
03:50	Power – Roufberg 45 min)
04:35	Adjourn

Agenda

Day 2: September 30, 2009

08:30	Mechanical - Hartka (25 min)
08:55	Dust – Mehoke (20 min)
09:25	Avionics - Eisenreich (20 min)
09:45	Software - Furrow (30 min)
10:15	Break
10:30	Data Systems - Koontz (15 min)
10:45	Ground Systems - Furrow (15 min)
11:00	I&T - Colby (30 min)
11:30	Fault Management – Trela (20 min)
12:00	Lunch
01:00	Concept of Operations - Pinkine (20 min)
01:20	Launch Vehicle/3rd Stage – Vernon (15 min)
01:35	MA & Safety - Pfisterer (15 min)
01:50	Risks - Kinnison (45 min)
02:35	Programmatics - Dantzler (30 min)
03:05	Break
03:20	IPAO Cost Outbrief

Day 3: October 1, 2009

8:30	SRB Outbrief

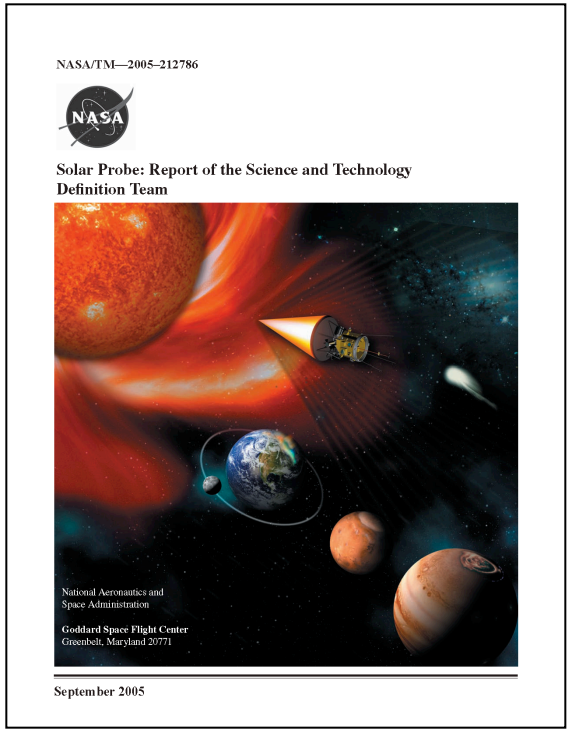
Solar Probe Plus
Mission Concept Review

Science

R. B. Decker
Acting Project Scientist

The Johns Hopkins University
APPLIED PHYSICS LABORATORY

Source of All Material

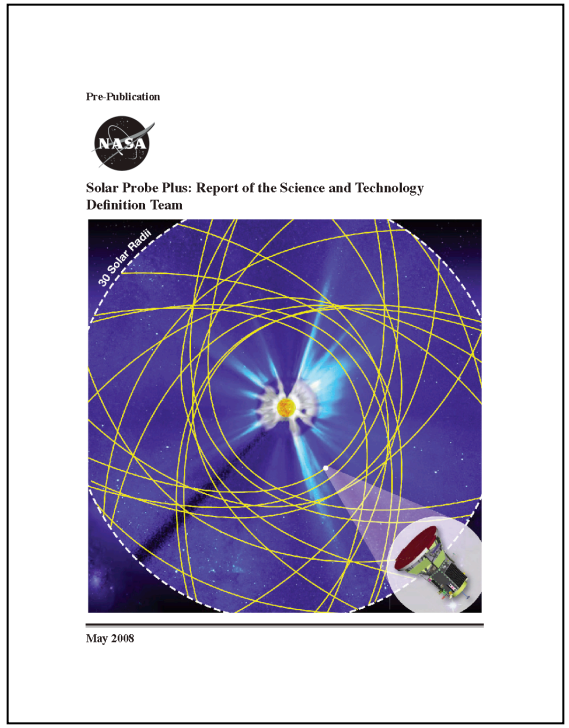

NASA/TM—2005–212786

Solar Probe: Report of the Science and Technology Definition Team

National Aeronautics and Space Administration

Goddard Space Flight Center
Greenbelt, Maryland 20771

September 2005


Pre-Publication

Solar Probe Plus: Report of the Science and Technology Definition Team

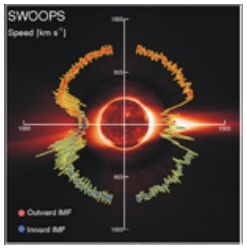
May 2008

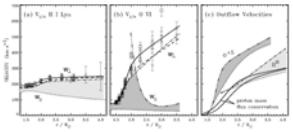
All material (text, graphics) in this presentation is from NASA/GSFC reports:

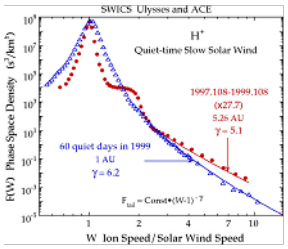
1. Solar Probe: Report of the Science and Technology Definition Team, NASA/TM—2005–212786, September 2005. http://solarprobe.gsfc.nasa.gov/SolarProbe_STDT2005.pdf
2. Solar Probe Plus: Report of the Science and Technology Definition Team, NASA/TM—2008–214161, July 2008. http://solarprobe.gsfc.nasa.gov/SolarProbe+Web.pdf

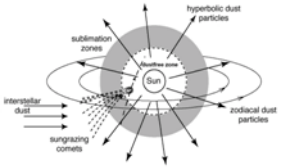
Science Objectives: Solar Probe Plus

(Reference: 2005 and 2008 Solar Probe/Solar Probe Plus STDT reports)



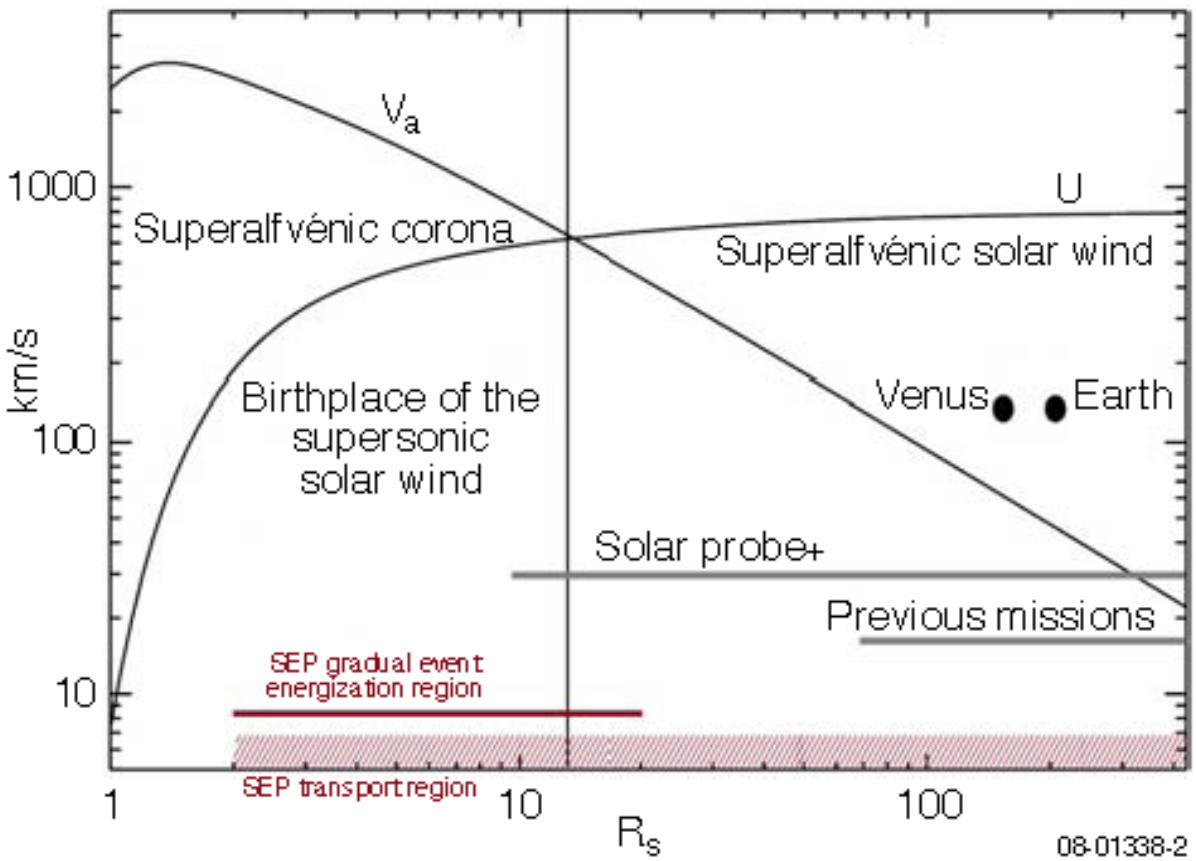






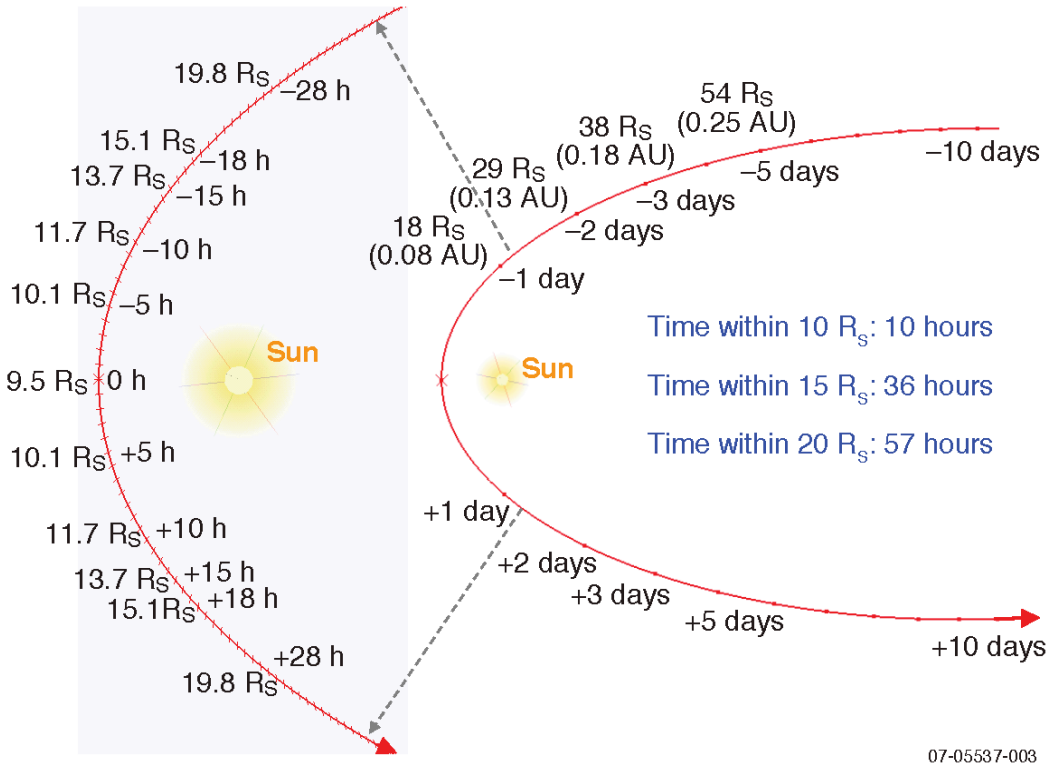
- Determine the structure and dynamics of the magnetic fields at the sources of the fast and slow solar wind.
- Trace the flow and elucidate the thermodynamics of the energy that heats the solar corona and accelerates the solar wind.
- What mechanisms accelerate and transport energetic charged particles?
- Explore dusty plasma phenomena and their influence on the solar wind and energetic particle formation.

SPP will approach sun to within 9.5 R_S



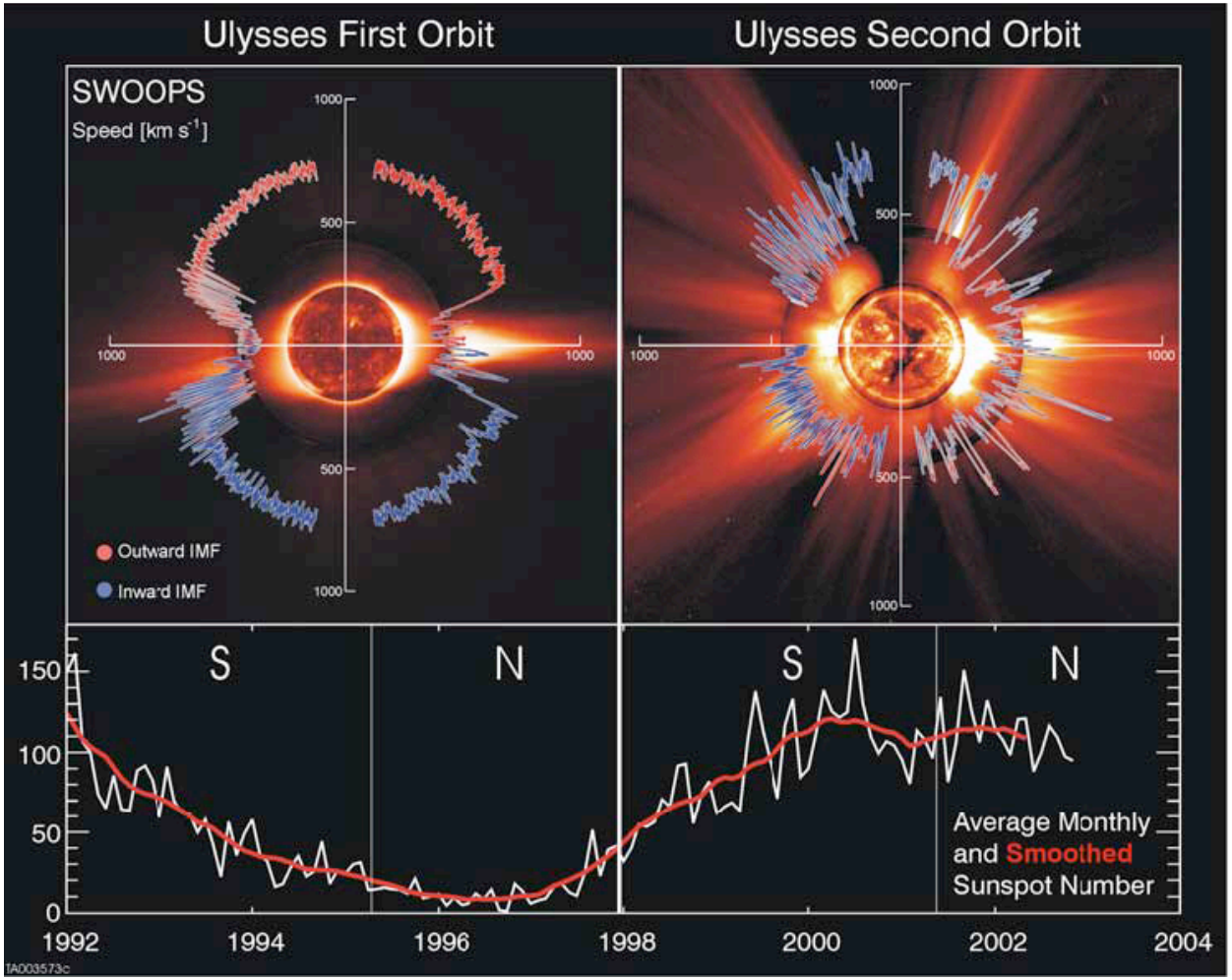
- Model profiles of solar wind speed and Alfvén wave speed with distance from the Sun.
- The vertical bar separates the source, or sub-Alfvénic, region of the wind from the supersonic solar wind flow.
- Solar Probe+ will be the first mission to fly inside the solar wind source region and to sample directly the critical region of the outer corona where solar energetic particles (SEPs) are generated.

SPP will take 30 hrs of data inside 10 R_S



- The trajectory of the Solar Probe+ during one of its minimum perihelion passes.
- With three such passes during its prime mission, Solar Probe+ will gather a total of 30 hours worth of data inside 10 R_S.

SPP will measure near-Sun solar wind during activity minimum (left) and maximum (right)



SPP will measure fast, slow, and transient solar wind near the Sun

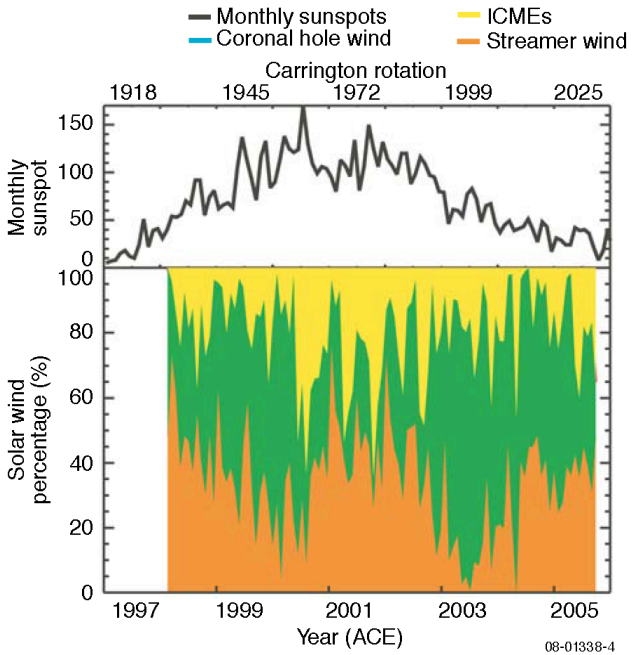


Figure ES-5. Observations from the Advanced Composition Explorer (ACE) during the rising phase, peak, and declining phase of Solar Cycle 23 show how the percentage of solar wind from different sources observed in the ecliptic varies with solar activity (yellow = transient wind; green = fast wind from coronal holes; red = wind associated with coronal streamers). From its near-ecliptic orbit, Solar Probe+ will, over its numerous near-Sun perihelia, repeatedly sample the wind from all solar sources, producing a extensive set of observations of the wind in both its quasi-stationary and transient modes (*Zhao and Zurbuchen*, 2008).

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Science Issue: Determine the structure and dynamics of the magnetic fields at the sources of the fast and slow solar wind.

Specific questions:

1. How does the magnetic field in the solar wind source regions connect to the photosphere and the heliosphere?
2. How do the observed structures in the corona evolve into the solar wind?
3. Is the source of the solar wind steady or intermittent?

Q3: Is the source of the solar wind steady or intermittent?

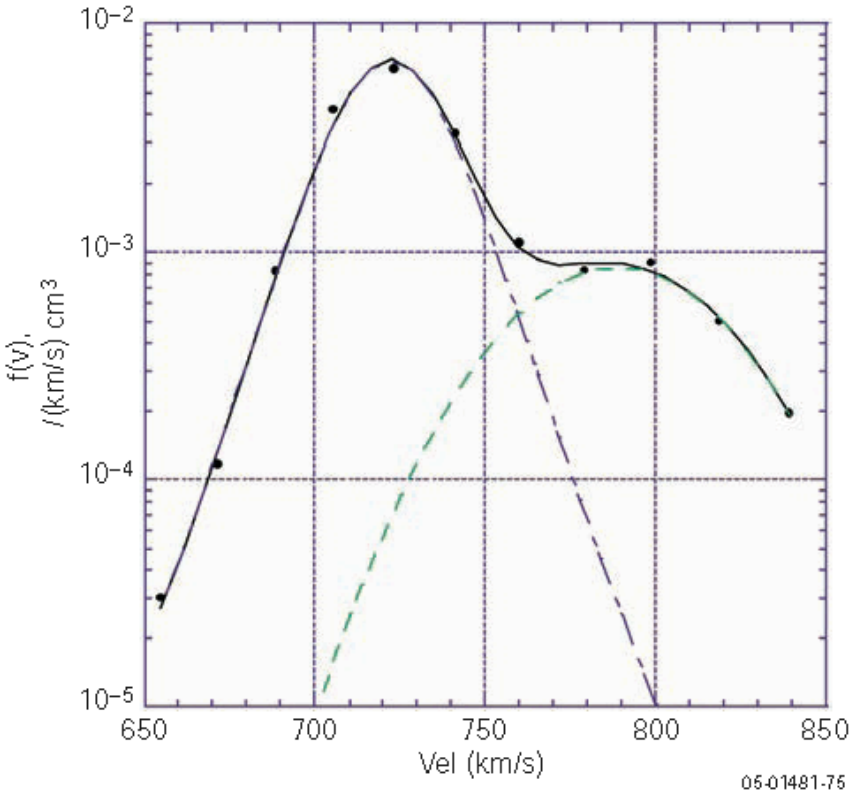


Figure 2-9. Two-beam model fit to logarithm of phase-space density for the Ulysses proton spectrum in the high-speed solar wind (*Goldstein et al.*, 2000). A proton beam with a drift speed of about 50 km/s, i.e., the Alfvén speed, gives best fit. Continuous plasma measurements inside 65 R_S can be used to determine where this beam forms and whether it is the direct remnant of the acceleration mechanism or is produced *in situ* by wave–particle interactions.

Measurement requirements

- In-situ magnetic field and plasma velocity at high cadence in inner heliospheric regions (below 20 R_S)
- Solar wind density, temperature, and composition
- Electron distribution function (bidirectional streaming as evidence of closed magnetic field line topology and correlations with composition and wind speed and magnetic field); strahl;and high-energy tails of proton and helium distribution functions at high cadence
- Neutron and gamma-ray detection
- Energetic electrons and ions

Science Issue: Trace the flow and elucidate the thermodynamics of the energy that heats the solar corona and accelerates the solar wind.

Specific questions:

1. How is energy from the lower solar atmosphere transferred to, and dissipated in, the corona?
2. What coronal processes shape the non-equilibrium velocity distributions observed throughout the heliosphere?
3. How do the processes in the corona affect the properties of the solar wind in the heliosphere?

Q1: How is energy from the lower solar atmosphere transferred to, and dissipated in, the corona?

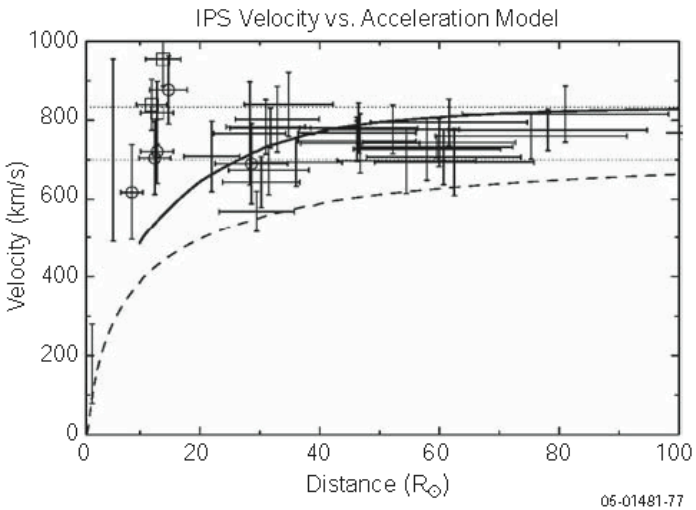


Figure 2-11. Apparent flow speed of the fast wind vs. distance determined from radio scintillation measurements by EISCAT, VLBA, and Spartan 201-01, showing rapid acceleration of the solar wind at distances $\leq 10\ R_S$ (*Grall et al.*, 1996). These measurements rely on the phase shifts in radio signals caused by the movement of density structures across the line of sight, and therefore contain the effects of compressive waves.

Measurement Requirements

- Magnetic field, velocity field, and density fluctuations and their spectra
- Particle distribution functions of protons, electrons, alpha particles, and possibly minor ion species; suprathermal populations
- Compositional variation of the solar wind across structures (FIP effect)
- High-cadence 3-axis electric and magnetic field plasma wave measurements with sufficient temporal resolution to resolve the expected proton gyroperiod at 9.5 R_S (~100 Hz).
- Coherent structure identification using plasma wave electric and magnetic field wave-form data

Science issue: What mechanisms accelerate and transport energetic charged particles?

Specific questions:

1. What are the roles of shocks, reconnection, waves, and turbulence in the acceleration of energetic particles?
2. What are the seed populations and physical conditions necessary for energetic particle acceleration?
3. How are energetic particles transported radially and across magnetic field lines from the corona to the heliosphere?

Q1: What are the roles of shocks, reconnection, waves, and turbulence in the acceleration of energetic particles?

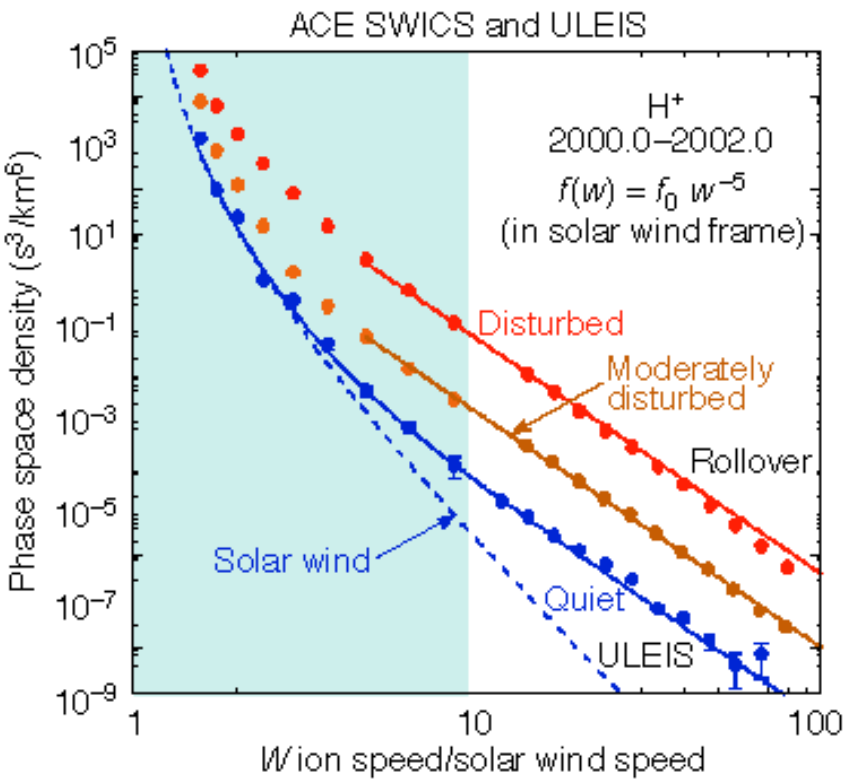


Figure 2-17. Velocity distributions of H measured with the ULEIS and SWICS instruments on ACE during quiet, moderately disturbed, and disturbed times. Power law tails (in the solar wind frame), with spectral index –5, extend to W ≈ 50 (2.5 MeV).

Measurement Requirements

- Composition and energy spectra of energetic ions and electrons
- Suprathermal ions and electrons
- Neutron and gamma-ray energy spectra
- In-situ magnetic field and solar wind properties
- Photospheric magnetic field measurements and, if available, EUV observations from Solar Orbiter

Science Issue: Explore dusty plasma phenomena and their influence on the solar wind and energetic particle formation

Specific questions:

1. What is the dust environment in the inner heliosphere?
2. What is the origin and composition of dust in the inner heliosphere?
3. What is the nature of dust–plasma interactions and how does dust modify the spacecraft environment close to the Sun?
4. What are the physical and chemical characteristics of dust-generated species?

Dust measurement requirements

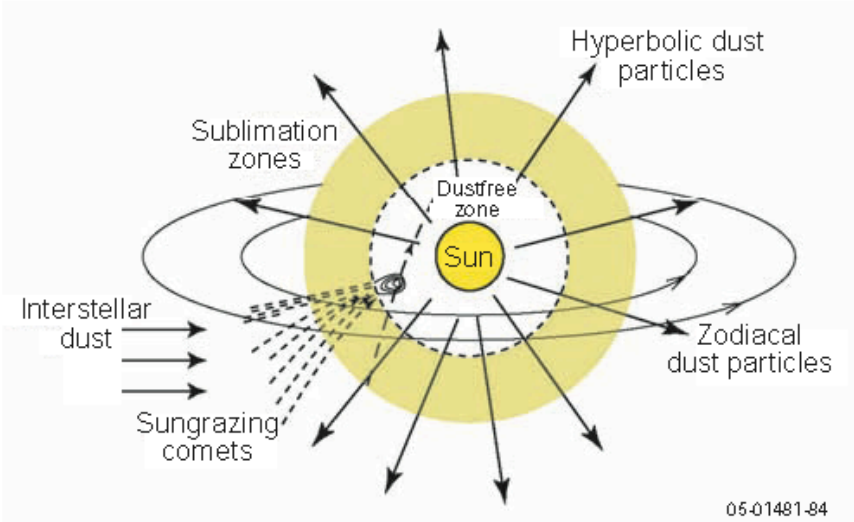


Figure 2-19. Sketch illustrating the dust environment near the Sun. Most of the dust near the Sun is bound in Keplerian, roughly circular orbits near the ecliptic, although some dust particles are ejected by radiation pressure on hyperbolic trajectories as b-meteoroids. Comets and asteroids are the principal sources of the solar dust cloud, with smaller contributions from sungrazing comets and interstellar dust. (*Mann et al.*, 2004)

Measurement Requirements

- Spatial variation of dust flux as a function of radial distance and latitude from 9.5 R_S to 1.02 AU
- Distribution functions and composition of inner source pickup ions
- Solar wind bulk parameters
- Solar wind ion composition
- Plasma wave electric field measurements
- Energetic particle spectra and composition
- Magnetic field orientation and strength

Table 3-1. Traceability matrix

(From July 2008 STDT Report)

Table 3-1. Traceability Matrix. All measurements are includd in the Solar Probe+ Baseline mission. "R" denotes measurements required to address the question in whose row they appear; "S" denotes supporting measurements.

KEY

R: Required

S: Supporting

CD: Coronal Dust Detector
EPI: Energetic Particle Instrument
FEA: Fast Electron Analyzer
FIA: Fast Ion Analyzer
ICA: Ion Composition Analyzer
HI: Hemispheric Imager
MAG: Magnetometer
NGS: Neutron/Gamma Ray Spectrometer
PWI: Plasma Wave Instrument

			Measurements/Instruments						
Science Objectives	Objective Questions	Strategy	PLASMA DISTRIBUTION; COMPOSITION; SUPRATHERMALS (FIA, FEA, ICA)	ENERGETIC PARTICLES (EPI)	PLASMA WAVES (PWI)	VECTOR MAGNETIC FIELD (MAG)	NEUTRON/GAMMA-RAY EMISSIONS (NGS)	CORONAL DUST (CD)	WHITE LIGHT IMAGING (HI)
2.1 Determine the structure and dynamics of the magnetic fields at the sources of the solar wind.	a. How does the magnetic field in the solar wind source regions connect to the photosphere and the heliosphere?	Coordinate in-situ observations and remote-sensing of corona (high-cadence white-light imaging < 20 $R_\odot$); intercompare in-situ measurements along the orbits.	R	S	S	R			R
	b. How do the observed structures in the corona evolve into the solar wind?	Identify in-situ signatures of density and magnetic field structures or boundaries and compare with coronal structure as observed with remote sensing.	R	S	S	R			R
	c. Is the source of the solar wind steady or intermittent?	Perform statistical study of temporal and radial variations of composition and plasma and field data of various types of solar wind; relate to filamentary structure and filling factors.	R	S	S	R	S		R
2.2 Trace the flow of the energy that heats the solar corona and accelerates the solar wind.	a. How is energy from the lower solar atmosphere transferred to and dissipated in the corona?	Characterize the turbulence and energy budgets and their evolution along the orbits; compare with sources at the Sun and along the trajectory; search for signatures of candidate dissipation mechanisms at kinetic scales.	R	S	R	R	R		S
	b. What coronal processes shape the non-equilibrium velocity distributions observed throughout the heliosphere?	Compare measured electromagnetic fields and distribution functions of protons, electrons,and heavy ions and their variability	R	R	R	R			
	c. How do the processes in the corona affect the properties of the solar wind in the heliosphere?	Use measured amplitude and spectra of electromagnetic and plasma fluctuations to assess role of Alfvénic turbulence in coronal heating and solar wind acceleration; with modeling, determine origin of spectra (photospheric or coronal).	R	S	R	R			S
2.3 Determine what mechanisms accelerate and transport energetic particles.	a. What are the roles of shocks, reconnection, waves, and turbulence in the acceleration of energetic particles?	Correlate energetic particle distributions and their temporal variations with shocks, signatures of reconnection, and properties of turbulence properties.	R	R	R	R	S		S
	b. What are the seed populations and physical conditions necessary for energetic particle acceleration?	Correlate suprathermal and energetic particle distribution functions with measured plasma and electromagnetic field properties.	R	R	R	R			
	c. How are energetic particles transported from the corona to the heliosphere?	Determine energetic particle distribution functions and composition and turbulence properties along the trajectory, along with measured and modeled large-scale magnetic field.	R	R	R	R	S		S
2.4 Explore dusty plasma phenomena and their influence on the solar wind and energetic particle formation.	a. What is the dust environment of the inner heliosphere?	Compare dust measurements with brightness patterns from coronagraph and models to characterize size distribution.			S	S		R	S
	b. What is the origin and composition of dust in the inner heliosphere?	Derive information about dust composition from measurements of dust-generated species and from sublimation sequence determined from size distribution as function of distance from the Sun.	R			S		R	
	c. What is the nature of dust–plasma interactions in the near-Sun environment?	Correlate plasma distribution functions, composition measurements, and dust properties.	R		S	S		R	S
	d. What are the physical and chemical characteristics of dust-generated species?	Measure composition of dust-generated species; correlate with dust, solar wind, and magnetic field data.	R			S		R	

Table 3.2. Solar Probe+ baseline instrument payload (From July 2008 STDT Report)

Table 3-2. Solar Probe+ baseline instrument payload

Instrument	Parameter(s) or Quantity(ies) Measured	Sensitivity Dynamic Range	Spectral Range Resolution	Angular Range Resolution	Time Integration Cadence
Fast Ion Analyzer (FIA)	H, He distribution functions	2D distributions in 0.1 s @20 R_S	50 eV–20 keV $\Delta E/E \sim 0.05$	330° × 90° (2 orthogonal) 10° × 10° 90° back from heat shield (ϕ) × centered on ram direction; 80° perp to symmetry axis; 5° toward beam to 20° away from beam	3 s for 3D; 0.1 s for 2D
Fast Electron Analyzer (FEA)	Electron distribution function	2D distributions in 0.1 s @20 R_S	1 eV–5 keV $\Delta E/E \sim 0.1$	As close to 4π less S/C obscuration ~3° (1D) near strahl 30°× 30° otherwise	3 s for 3D; 0.1 s for 2D
Ion Composition Analyzer (ICA)	Distribution functions of ^{3}He, ^{4}He, C, O, Mg, Si, Fe; composition of dust-related pickup ions from 2 to ~60 amu	Enable He/O in 10 s at 20 R_S	100 eV/q–60 keV/q $\Delta E/E \sim 0.04$–0.05	120° × 120° 10° × 10°	10 s for He/O ratio at 20 R_S 60 s for heavy ions (e.g., Fe/O, charge states)
Magnetometer (MAG)	DC vector magnetic field	10 nT–8 G 3-axis >10^4	n/a	3° knowledge inside 20 R_S	20 samples/s
Plasma Wave Instrument (PWI)	3-axis AC E- and B-fields	2×10^{-17} V/m^2/Hz	**E**: 10 Hz–10 MHz **B**:10Hz–80 kHz 40 samples/decade	n/a	0.1 s for spectral data; 60 s for waveform data
Energetic Particle Instrument (EPI)	Differential fluxes of H, ^{3}He, ^{4}He, C, O, Ne, Mg, Si, Fe, and e^-	10^6 protons/cm^2 s sr with 0.2–1 MeV 5×10^5 protons / cm^2 s sr with >1 MeV	0.02–100 MeV/nuc for ions 0.03–3 MeV for e^- 6 energy bins/decade	Capability to measure pitch-angle distributions and anisotropies. Minimum coverage: 30° to 150° wrt nominal magnetic field at 4 R_S with 30° resolution	5 s for H; 30 s for Z ≥ 2; 1 s for e^-
Neutron/γ-ray Spectrometer	Neutron, hard X-ray, and gamma-ray spectra	Factor of 10^4	0–20 MeV < 0.5 $\Delta E/E$ 0.02–10 MeV (power-law spectrum)	4π field of view for both energy ranges	10 s integration both ranges 10 s cadence both ranges
Coronal Dust (CD)	Flux and mass spectrum; silicate vs. organic	m > 10–15 g for ΔV = 5 km/s (smaller particles detected for larger ΔV)	0.05–50 μm (1 fg–1 μg)	As close to 2π less S/C obscuration	10 s 30 s
Hemispheric Imager (HI)	White light intensity	10^{-9} erg/cm^2/deg/s	White light n/a	160° × 160° less S/C obscurations ~0.2°	1–10 s 90 s

3-6

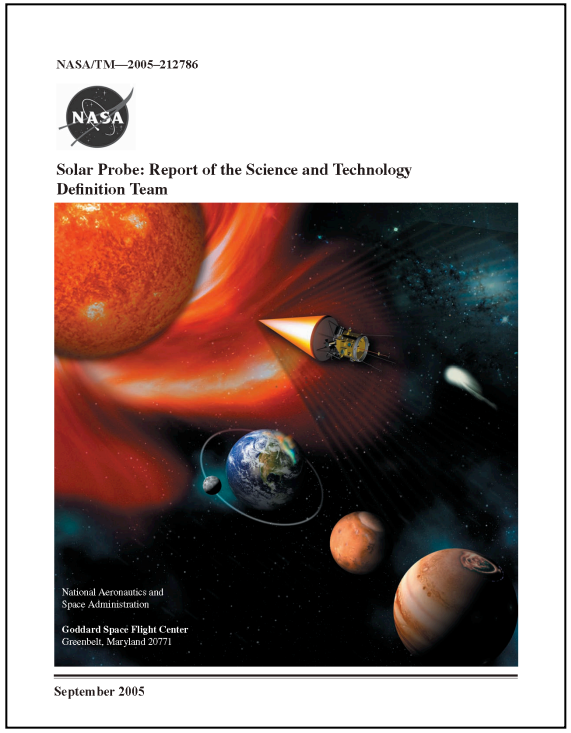
Solar Probe+: Report of the Science and Technology Definition Team

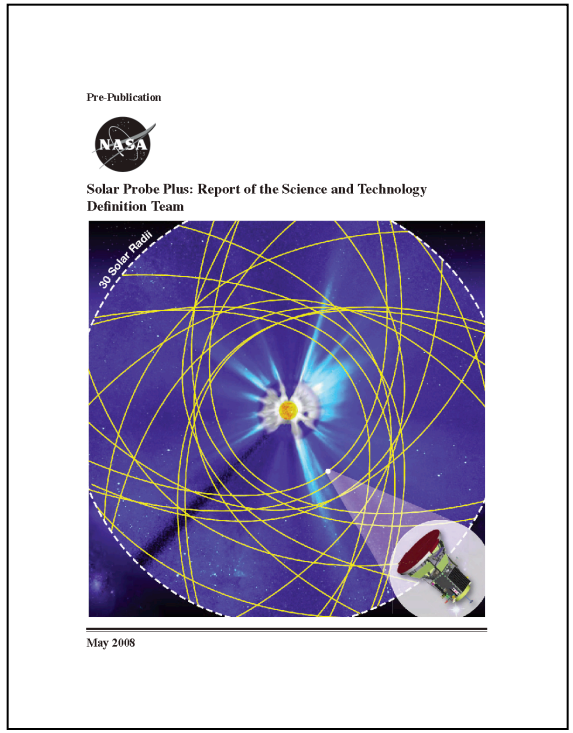
Table 3-3. Instrument resource requirements (From July 2008 STDT Report)

Table 3-3. Instrument resource requirements.

Instrument	Mass (kg)	Power (W)	Peak Data Rate (kbps)
Fast Ion Analyzer (FIA)	2.8	3.7	10
Fast Electron Analyzer (FEA) (2X)	5.0	7.2	20
Ion Composition Analyzer (ICA)	7.0	6.0	10
Magnetometer (MAG)	2.5	2.5	1.1
Plasma Wave Instrument (PWI)	5.0	5.0	3.5
Energetic Particle Instrument (Low Energy)	1.4	2.3	5
Energetic Particle Instrument (High Energy)	2.7	1.7	3
Neutron/Gamma Ray Spectrometer (NGS)	2.0	3.0	0.5
Coronal Dust Detector (CD)	1.5	3.8	0.1
Hemispheric Imager (HI)	1.5	4.0	70
TBD	5.0	4.0	70
Common DPU/LVPS	10.8	14.0	N/A
Total	47.2	57.2	123.2

Summary: See 2005 SP and 2008 SP+ STDT reports





All material (text, graphics) in this presentation is from NASA/GSFC reports:

1. **Solar Probe: Report of the Science and Technology Definition Team, NASA/TM—2005–212786, September 2005.** (http://solarprobe.gsfc.nasa.gov/SolarProbe_STDT2005.pdf)
2. **Solar Probe Plus: Report of the Science and Technology Definition Team, NASA/TM—2008–214161, July 2008.** (http://solarprobe.gsfc.nasa.gov/SolarProbe+Web.pdf)

Solar Probe Plus
Mission Concept Review

Mission Overview

Jim Kinnison
Mission System Engineer

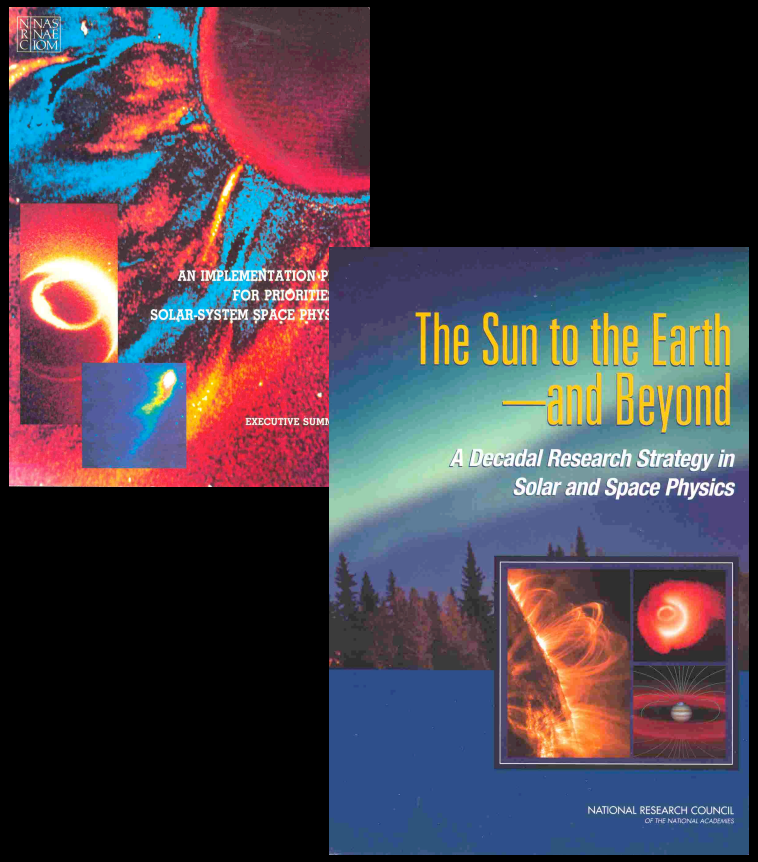
The Johns Hopkins University
APPLIED PHYSICS LABORATORY

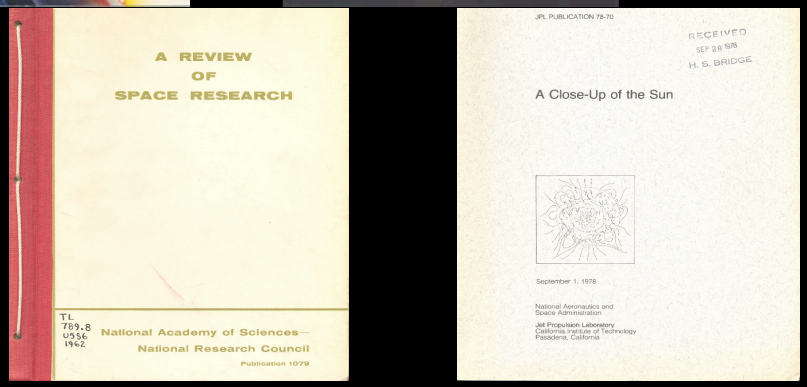
Outline

- Overview of Solar Probe History
- 2008 Engineering Study Summary
- Pre-Phase A Summary
  - Task Summary
  - Requirements
  - Top-Level Philosophy
- System Engineering Approach

Solar Probe History (1958 - present)

New STDT in line of studies dating back to "Simpson's Committee" of the Space Science Board (National Academy of Sciences) (24 October 1958)





Solar Probe studies, reports; NAS: 1962, 1985, 1995, 2003

2008 Engineering Study

1. Preserve Science
2. No RPS
3. Launch 2015
4. $750M Cap (FY07)

Science Evaluation

Feasibility

Mission Design Trade
- Single vs Multiple Passes
- Ecliptic vs Polar
- Perihelion Altitude

Mission Design

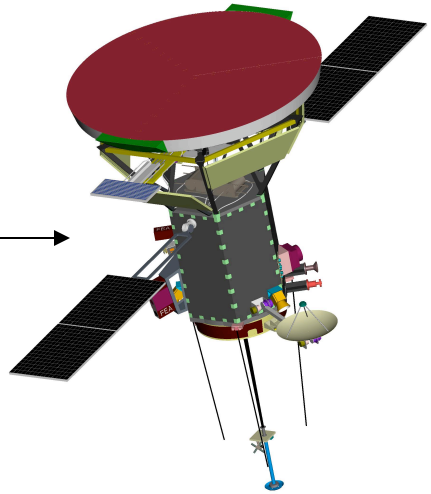
Power Generation
- Solar Encounter
- 0.3 - 1 AU
- > 1 AU

Brainstorming

Tech Assessment

Power Generation Trade
- Energy Storage
- TPS Waste Heat Conv.
- Cooled Solar Arrays

Solar Arrays



Engineering Requirement Allocation Assumptions

- No formal requirements have yet been allocated beyond the 4 Science Objectives and corollary requirements provided in the STDT Report of 2008
  - Our reference design addresses all STDT requirements
- Allocation assumptions made to develop an MCR reference vehicle to show mission feasibility
- The mission concept reflects a current best estimate based on the maturity of trades to show feasibility that requirements can be met
  - While the major trades have been completed, other trades typical of Phase A, most notably those involving the selected payload, are yet to be fully performed.
  - The concept is expected to evolve considerably before the end of Phase A

Draft Top-Level Requirements Identified

Mission Requirements

Requirement	Justification
At least three orbits with minimum perihelion less than 10 Rs	Required to separate fast and slow solar winds
A Sun-spacecraft-Earth geometry at encounter that allows simultaneous Earth-based observations to support Solar Probe observtions.	Connect in situ measrements with global Sun activity and derive context for measurements
Minimum perihelion orbits achieved within 10 years	Cost cap
Return of full data collected in a solar encounter	Accommation of payload and science

Working Spacecraft Requirements

- Operate at solar intensity consistent with spacecraft-Sun distances between 9.5 Rs and 1.02 AU
- Provide power over the distance range 0.0442 - 1.02 AU
- Protect instruments and spacecraft from dust environment near the Sun.
- Provide large total science data return (not to exceed 128 Gbits per orbit, including housekeeping and margin)
- Accommodate significant payload mass (not to exceed 51.9 kg*) and average power (not to exceed 62.9 W**)
  - Provide science boom for magnetometer and Plasma Wave Instrument search coils.

* 51.9kg NTE = 47.2kg CBE + 10% ** 62.9W NTE = 57.2W CBE + 10%

Overall Mission Concept Development Philosophy

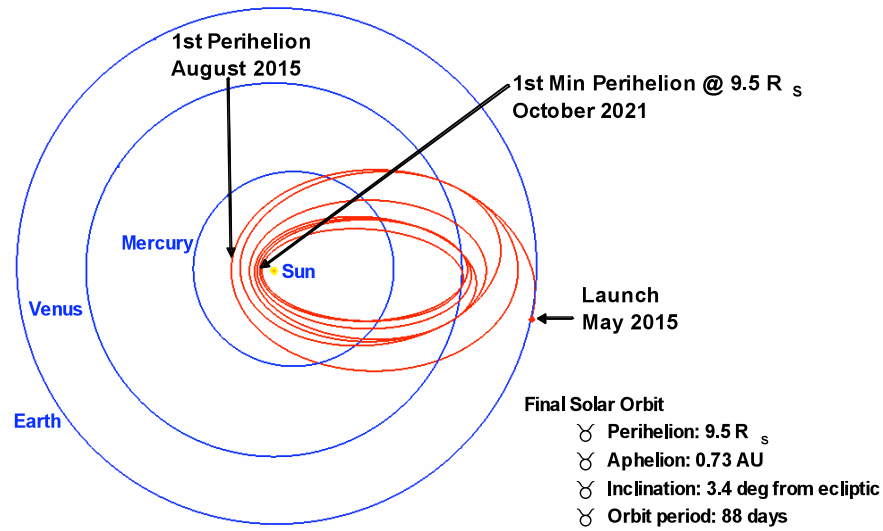
- Balance mission performance with resources and risk
  - Early identification of risks and mitigations
  - Identify options that provide the best performance in priority areas compared to resources
  - High consideration to margin management and preservation

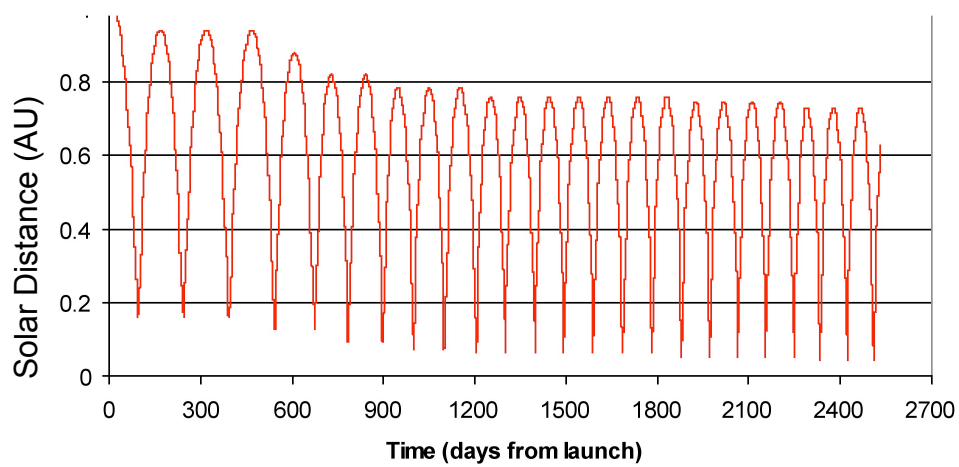
- Try to maintain a simple design concept that minimizes risk and cost
  - Maintain flexible mission design allowing step-by-step approach to minimum perihelion
  - Simplify spacecraft systems as possible
  - Maintain simple operating modes
    - STEREO and TIMED heritage operational approach
  - Minimize new technology developments

- Maintain simple and clean interfaces
  - Instruments perform de-coupled operations through SOCs

Mission Design Summary

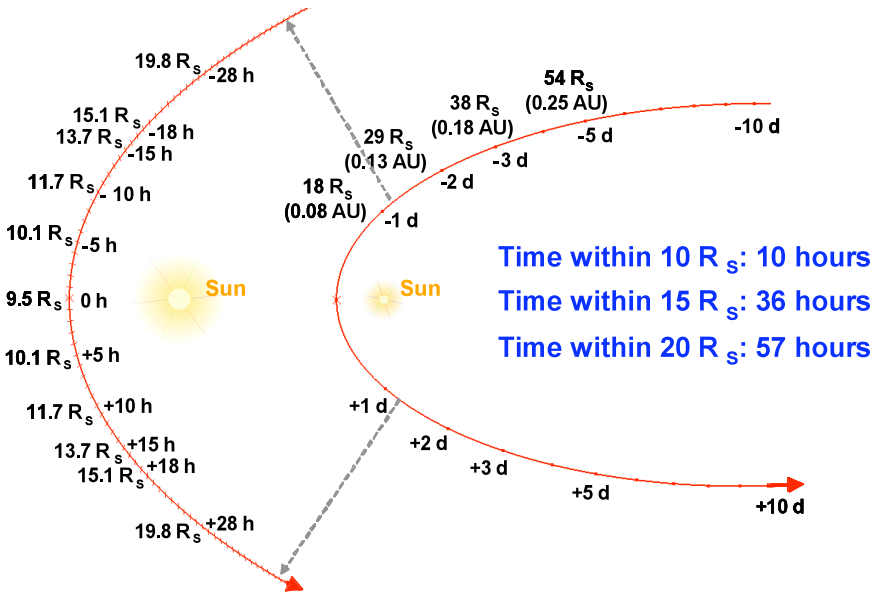
- Launch: May 21-June 9, 2015 (20 days)
- Max launch C3: 158 km^2/s^2
- Flybys: 7 Venus flybys
- Final solar orbit
  - Perihelion: 9.5 Rs
  - Aphelion: 0.73 AU
  - Inclination: 3.4 deg from ecliptic
  - Orbit period: 88 days
- Launch to 1st perihelion < 0.25 AU: ~3 months
- Launch to min perihelion: 6.39 years
- Mission duration (3 passes): 6.88 years
- Max Solar Distance: 1.02 AU
- No deep space maneuvers
- Launch mass: 610 kg (Atlas V 551 + Star 48BV)
- Delta-V budget: 230 m/s (propellant: 68 kg)





Solar Encounters

- Repeated passes give coverage over more than half the solar cycle
- Perihelion walk-in controlled by Venus flybys - depth of final perihelion adjustable

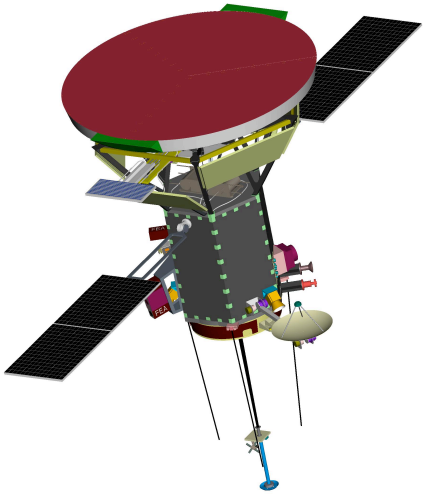


Solar Pass #	Time within 30 Rs (hr)	Time within 20 Rs (hr)	Time within 15 Rs (hr)	Time within 10 Rs (hr)
4	67			
5	67			
6	105	10		
7	105	10		
8	109	50		
9	109	50		
10	108	55	23	
11	108	55	23	
12	108	55	23	
13	108	55	23	
14	108	55	23	
15	108	55	23	
16	108	55	23	
17	105	57	33	
18	105	57	33	
19	105	57	33	
20	105	57	33	
21	105	57	33	
22	102	57	36	10
23	102	57	36	10
24	102	57	36	10
Total	2149	961	434	30

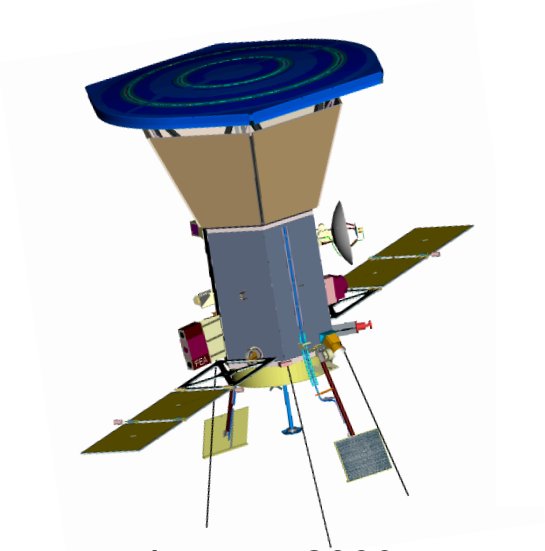
10x time spent below 20 Rs over previous mission designs

Spacecraft bus concept design has evolved as trade studies are conducted

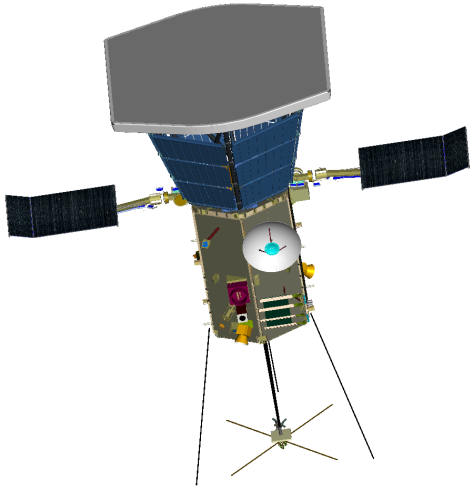
But requirements remain constant



March 2008
Study Report concept



January 2009
Sliding arrays replaced



August 2009
Combined Array concept

Key design drivers:

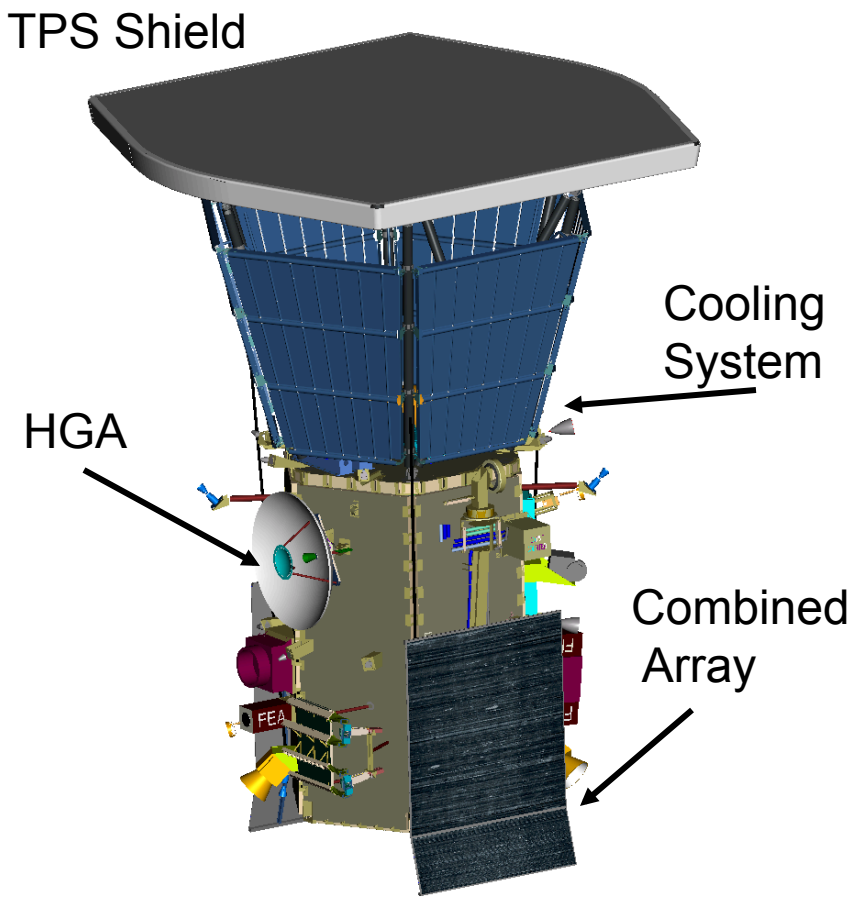
- Protect payload from Solar Intensity
- Provide power over 0.0442 -1.02 AU
- Control temperature of arrays
- High C3/limited lift mass

Key enabling subsystems:

- Thermal Protection System
- Solar Arrays
- Array Cooling system
- Third Stage

Near-Sun environment leads to coupled spacecraft design

- Sun-facing TPS casts shadow to protect spacecraft systems. Bus is packaged so that everything fits within the umbra.
- Solar arrays peek past umbra. Exposure designed to keep power generation and thermal load roughly constant throughout the orbit.
- Cooling system sized to keep array temperature below design limits for solar cells. Maximum heat capacity determined by expected max thermal load design point.
- Spacecraft slew allowed near aphelion for increased communications coverage. Bus layout optimized for thermal load in slewed orientation.
- G&C system keeps TPS sun pointing in the presence of environmental disturbances.



Reviews and Reports

Peer reviews

- TPS (April 30)
- Cooling System (May 5)
- Dust (May 13)
- Spectrolab Solar Array (May 21)
- Emcore Solar Array (May 29)
- Avionics (August 11)
- Third Stage (mid-October)

Study Reports

- Environments
- Dust Environment and Effects
- Thermal Protection System
- Transition Structure
- Cooling System
- Solar Arrays
- Mechanisms

Summary

Solar Probe Plus is ready to enter Phase A

- Existing mission requirements and implications for spacecraft design are understood.
- Feasible reference design serves as a basis for trades, analysis and requirements definition.
- Candidate payload is accommodated.
- Technical management is ready for Phase A

Solar Probe Plus
Mission Concept Review

Mission Design

Yanping Guo
Mission Design Lead

The Johns Hopkins University
APPLIED PHYSICS LABORATORY

Original Solar Probe Mission Concept Being Challenged

NASA Guidelines for 2007 Solar Probe "Lite" Study

- Preserve Solar Probe science defined in 2005 Report
- **Power spacecraft with non-nuclear source**
- Launch time around 2015
- Total mission duration less than 10 years
- Total mission cost under $750M

2005 Report Concept
baselined 2 near Sun passes via a Jupiter gravity assist (JGA) trajectory and 3 mini RTGs for power

Technically difficult to implement without RTG
(Oversized solar panels could not be retracted beneath TPS)

A possible solution to solve the power problem is to reduce the distance from the Sun => not to travel as far as to Jupiter

How to get close to the Sun without the gravity assist from Jupiter?
(Jupiter gravity assist was found a necessary for getting close to the Sun in past studies. Is a solar probe mission possible without JGA?)

Re-Design of Solar Probe

A wide range of trajectory designs were investigated

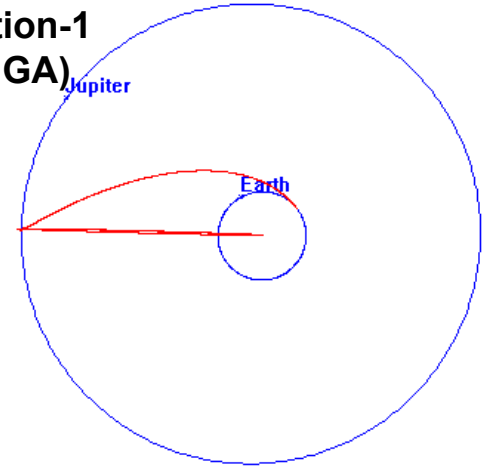
(Gravity assist from Jupiter, Earth, Venus, Mercury, Mars
Single or multiple flybys of same or different planets
Powered and un-powered flybys
With and without deep space maneuvers
High C3 - short duration vs. low C3 - long duration
Ecliptic vs. inclined solar orbit
Going inside vs. going beyond Earth orbit)

- **How close to the Sun is achievable under the program constraints?**
- **What trajectory design is favored by science investigation?**
  (feedbacks from the Science and Technology Definition Team)
- **What trajectory is favored by engineering implementation that simplifies spacecraft design and reduces mission costs and risks?**
  (feedbacks from the Engineering team)

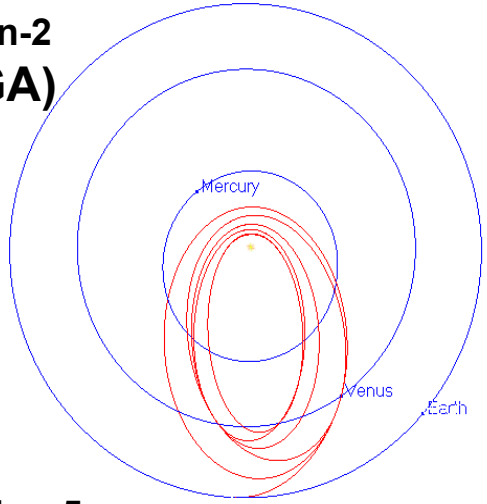
Six trajectory designs representing different mission options developed for detailed evaluation

6 Leading Candidate Trajectory Designs
Representing Various Mission Scenarios

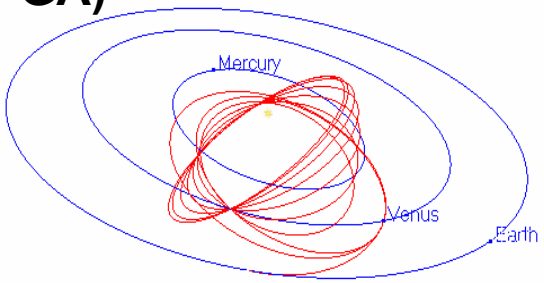
Option-1 (JGA)



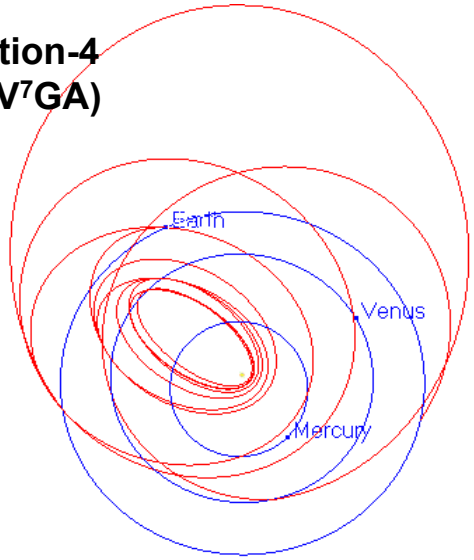
Option-2 (V^5GA)



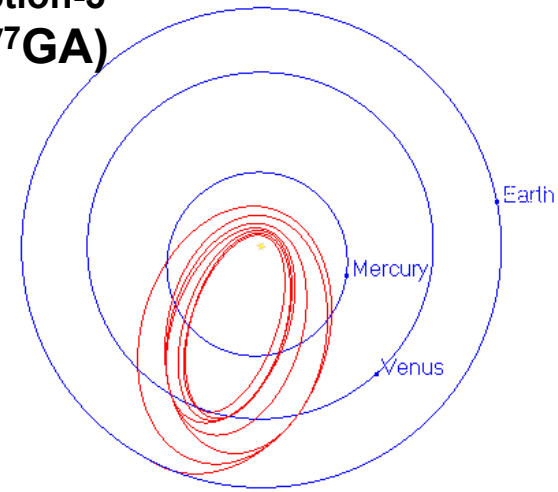
Option-3 (V^9GA)



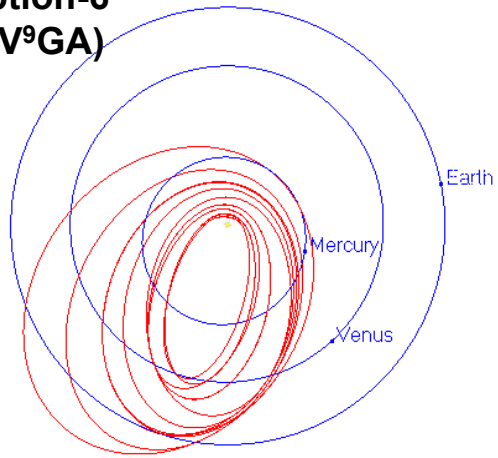
Option-4 (E^3V^7GA)



Option-5 (V^7GA)



Option-6 (EV^9GA)



Comparison of Trajectory Options

Trajectory Option		Option-1	Option-2	Option-3	Option-4	Option5 (V^7GA)	Option-6
Launch	Date	11/20/2015	5/30/2015	5/30/2015	9/6/2014	5/21/2015	5/24/2015
	C3 (km^2/s^2)	114	156	156	108	158	128
Trajectory	Flybys	1 Jupiter	5 Venus	9 Venus	3 Earth, 7 Venus	7 Venus	1 Earth, 9 Venus
	Deep space maneuver (ΔV)	None	None	None	1 (232 m/s)	None	1 (397 m/s)
	Max aphelion (AU)	5.56	1	1	2.29	1	1.19
Final orbit	Perihelion (Rs)	4	11.8	39.8	9.5	9.5	9
	Aphelion (AU)	5.56	0.75	0.725	0.73	0.73	0.73
	Inclination from ecliptic (deg)	90	3.4	37.9	2.9	3.4	3.4
	Orbital period	4.6 yr	94 d	112 d	88 d	88 d	88 d
Timeline	Launch to min. peri.	4.1 yr	3.3 yr	2.1 yr	10 yr	6.4 yr	9.5 yr
	Mission duration	4.5 yr	3.8 yr	5.8 yr	10.5 yr	6.9 yr	9.95 yr
Near Sun Pass	Total # of solar passes (< 0.2 AU)	1	12	18	16	24	21
Pros		Pole-to-pole solar flyby at 4 Rs	Short mission duration; multiple, frequent solar flybys; aphelion ≤ 1 AU	Short mission duration; multiple, frequent solar flybys; aphelion≤ 1 AU	Good perihelion distance; multiple, frequent solar flybys; moderate C3	Good perihelion distance; multiple frequent solar flybys; short mission duration; no deep space maneuver; aphelion ≤ 1 AU	Good perihelion distance; multiple, frequent solar flybys; moderate C3
Cons		Single solar flyby; great aphelion distance; long cruise; long orbit period	Low inclination; high C3	Large perihelion distance; high C3	Long mission duration; requiring deep space maneuver	low inclination; high C3	Long mission duration; requiring deep space maneuver

Findings from Trajectory Study

- It is possible to get closer than 10 R_S of the Sun without JGA but with new trajectory designs
- The mission can be accomplished with less than 10 years under affordable costs
- Significantly more science return is expected from the new trajectory designs, even though it's not a polar orbit
- The engineering issues associated with extremely high dynamic ranges of thermal and solar intensity associated with great solar distance changes can be substantially eased by designing a trajectory whose aphelion is constrained to no greater than 1 AU
- A re-designed Solar Probe mission is feasible under the NASA new ground rules

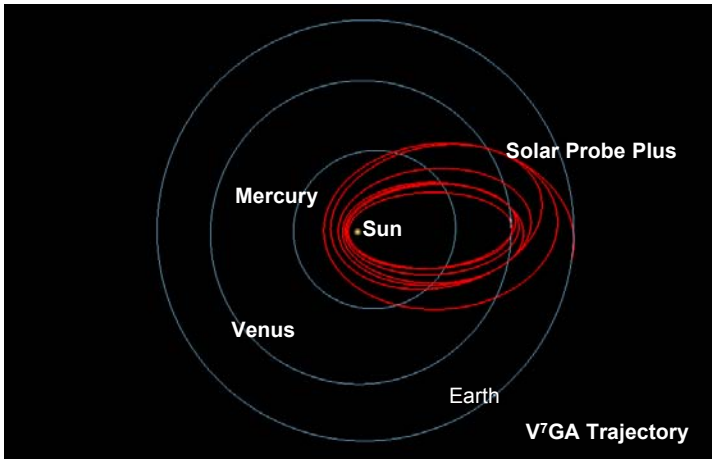
Key Mission Design Requirements

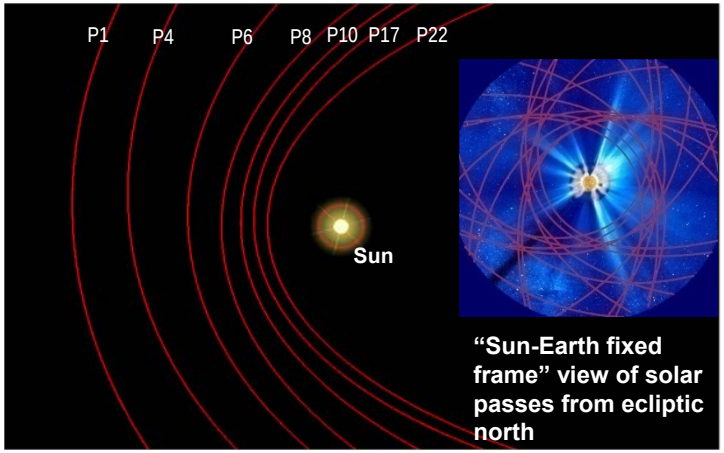
- Science Requirements
  - Minimum perihelion distance less than 10 R_S
  - Three solar flybys of perihelion less than 10 R_S
- Program Requirements and Constraints
  - Non-nuclear power source
  - Total mission duration less than 10 years
  - Launch in 2015

Redesigned Mission: Solar Probe Plus (SPP)

- **The V^7GA trajectory design is chosen as the Solar Probe Plus baseline trajectory**
- **This new trajectory design introduces a new mission concept:**

  The SPP spacecraft will explore the close Sun regions repeatedly at 3 to 4 near Sun passes per year over 7 years, reaching as close as 9.5 R_S from Sun's center via a trajectory controlled through 7 Venus flybys
- **SPP has significant advantages than previous Solar Probe:**
  - Numerous frequent solar passes (24 vs. 2)
  - Much more time spent within the close Sun regions (10 times more)
  - Wide variation perihelion directions as viewed from Earth
  - Perihelion gradually decreasing to 9.5 R_S
  - Solar orbits all within 1 AU

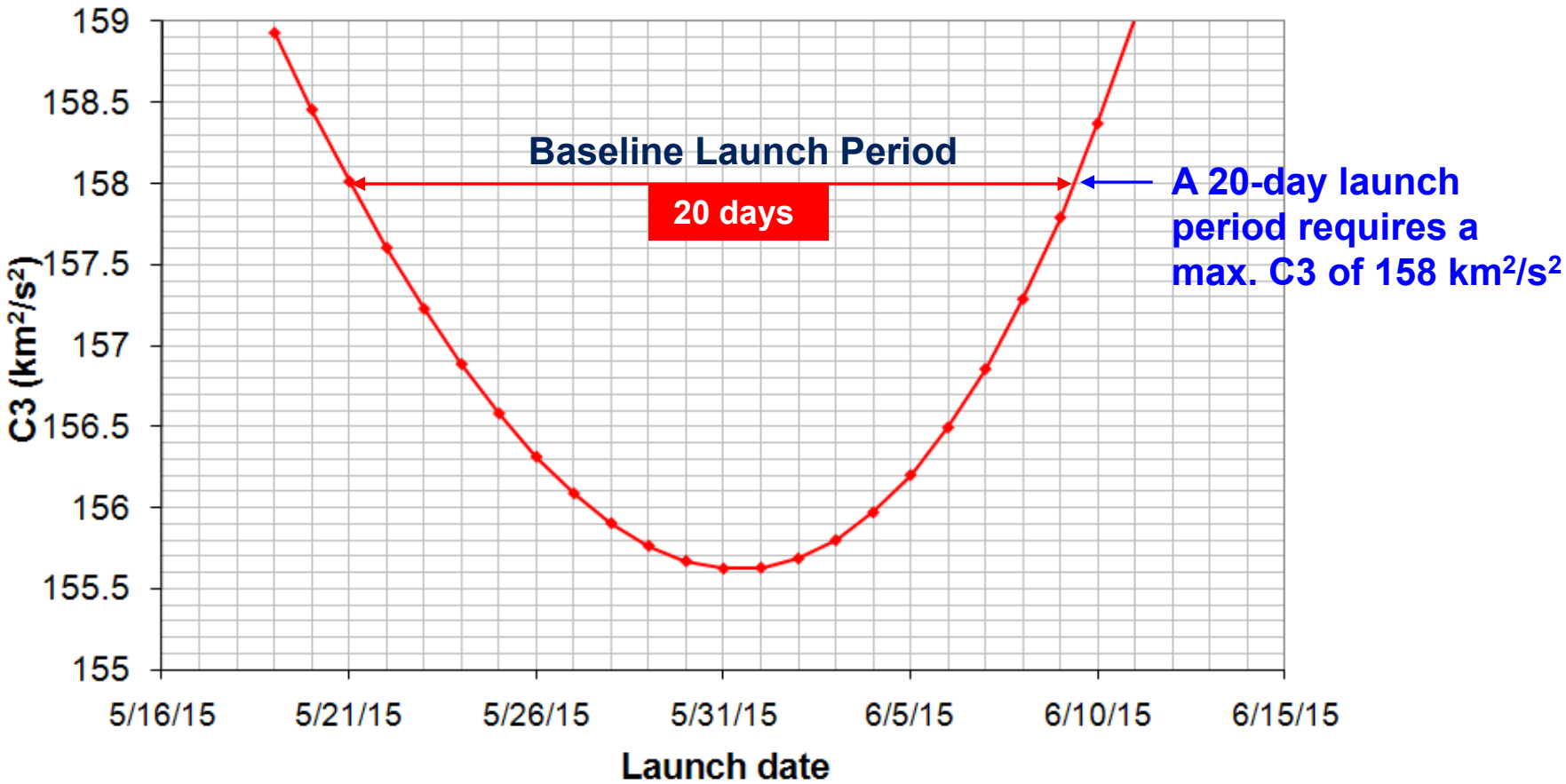




Overview of SPP Mission Design

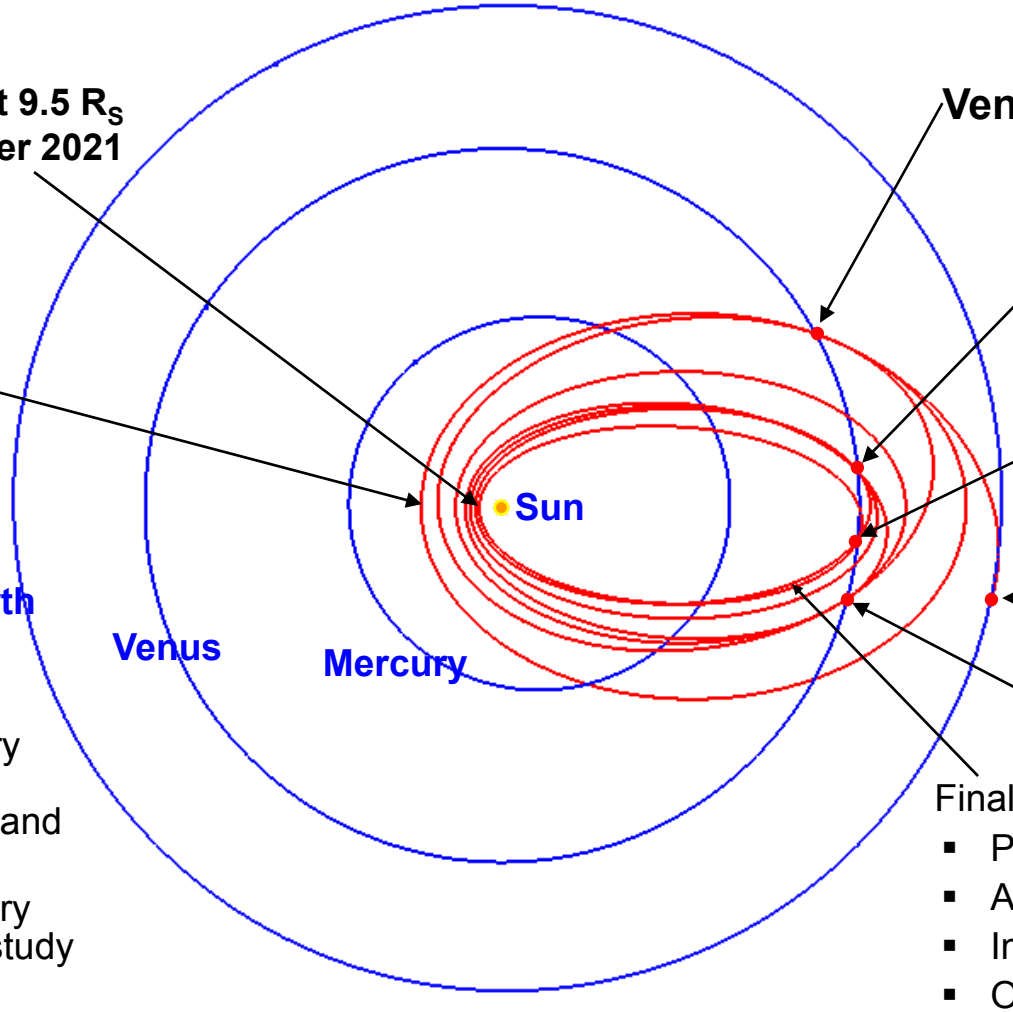
- Launch
  - Date: May 21 - June 9, 2015 (20 days)
  - Maximum launch C3: 158 km^2/s^2
- Interplanetary Trajectory
  - 7 unpowered Venus gravity assist flybys
  - No deep space maneuvers
  - 24 solar passes
  - Perihelion gradually decreasing to 9.5 R_S (0.0442 AU)
  - Maximum aphelion at 1.016 AU
- Final Solar Orbit
  - Perihelion: 9.5 R_S (8.5 R_s from surface of the Sun)
  - Aphelion: 0.73 AU
  - Inclination: 3.4 deg from ecliptic
  - Orbit period: 88 days
- Mission Timeline
  - Launch to 1st perihelion (35 R_S): 3 months
  - Launch to 1st min perihelion (9.5 R_S): 6.39 years
  - Baseline mission duration (including 3 passes @ 9.5 R_S): 6.88 years

Solar Probe Plus Launch Period & C3 Requirements



Launch DLA over the 20-day period is less than 10 degrees.

Solar Probe Plus Reference Trajectory

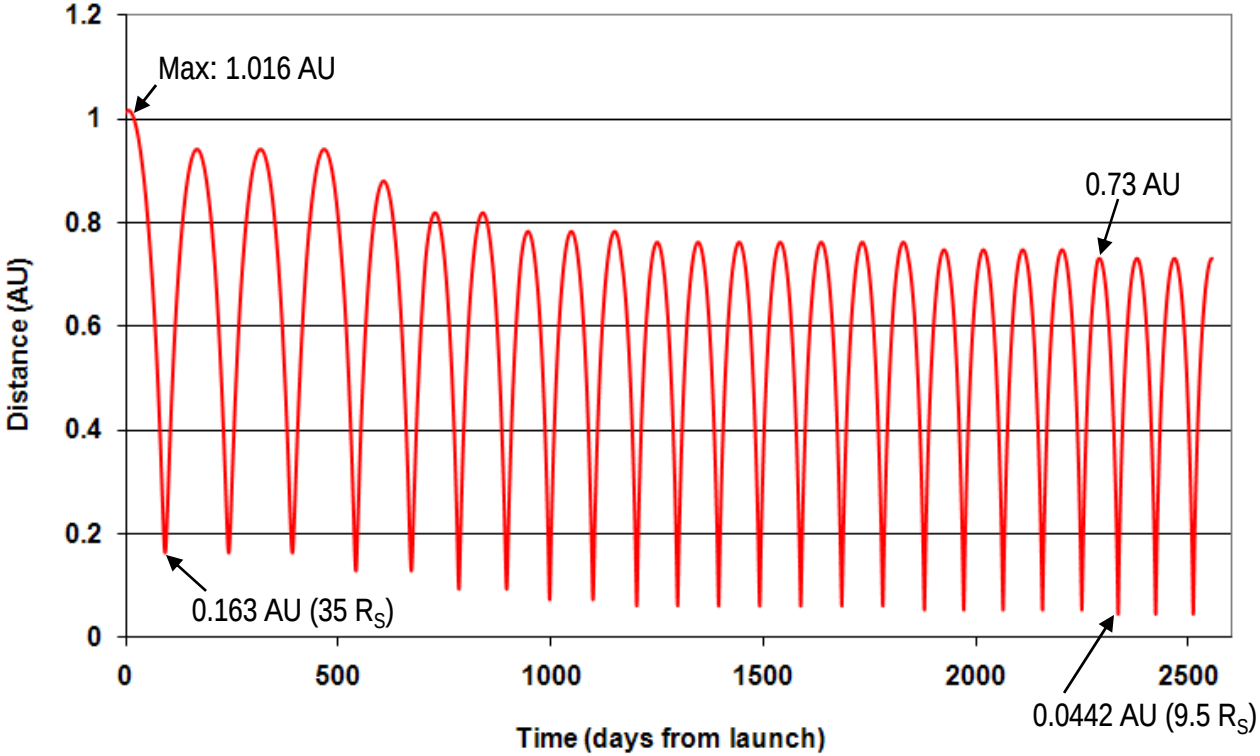


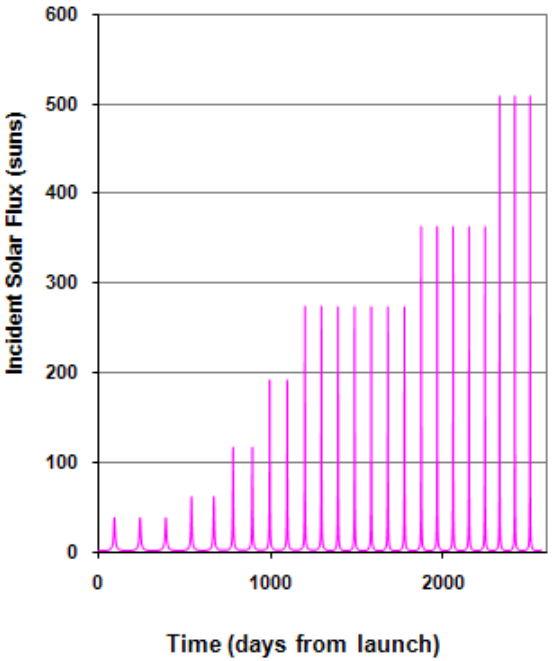
- An integrated trajectory with high fidelity force models including Sun and all planets and moon.
- The reference trajectory used for engineering study leading to MCR.

Final Solar Orbit

- Perihelion: 9.5 R_S
- Aphelion: 0.73 AU
- Inclination: 3.4° from ecliptic
- Orbit period: 88 days

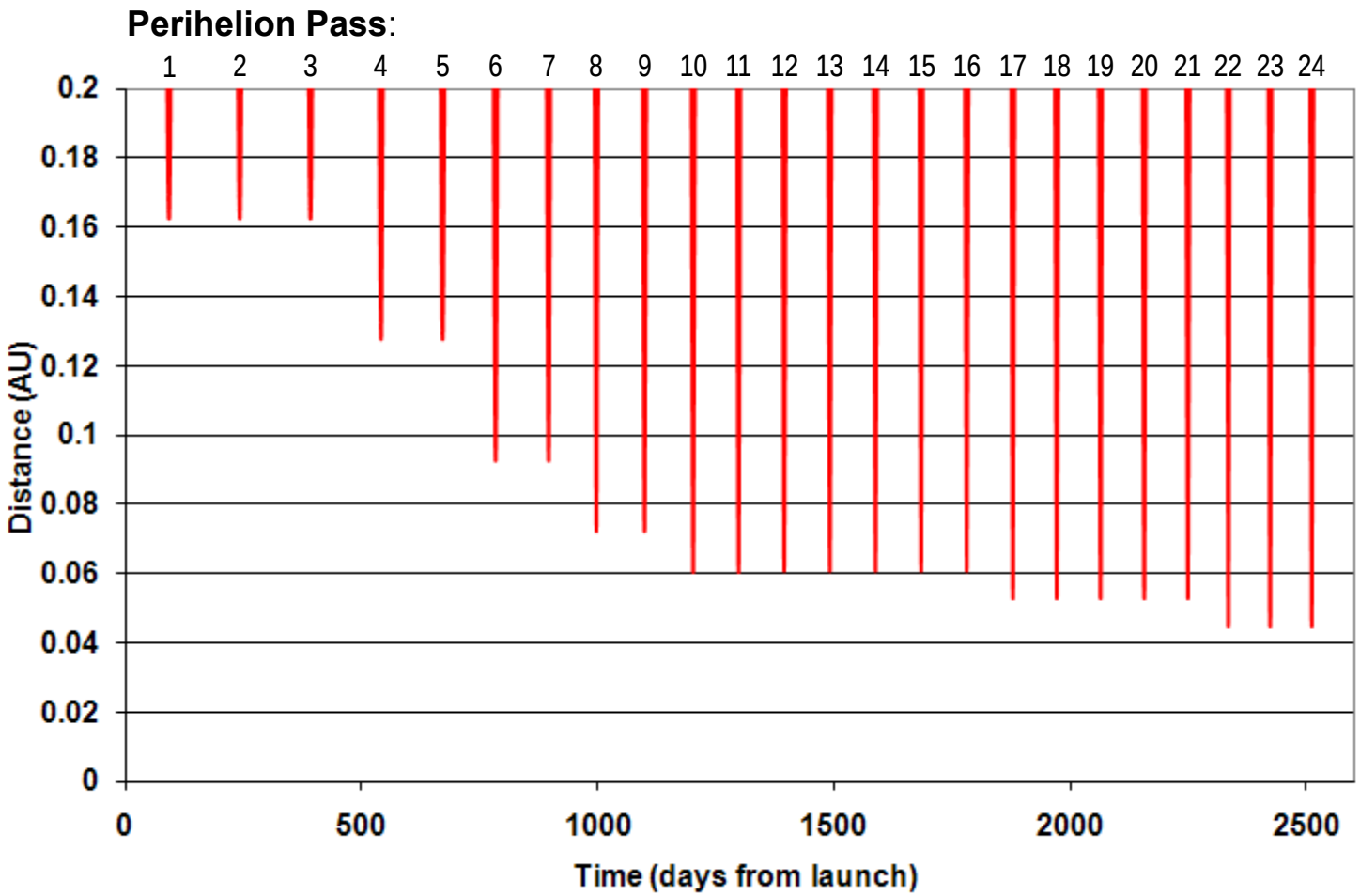
Solar Distances of the Reference Trajectory





- The range of solar distance defined by the trajectory is the main driver for spacecraft power and thermal system design.
- The solar distance profile (min, max, variation) allows for reasonably sized solar panels, making solar power feasible.
- The trajectory consists of 24 solar orbits, and the mission is conveniently divided into 24 orbits allowing for orbit-based operation process, saving mission operations costs.

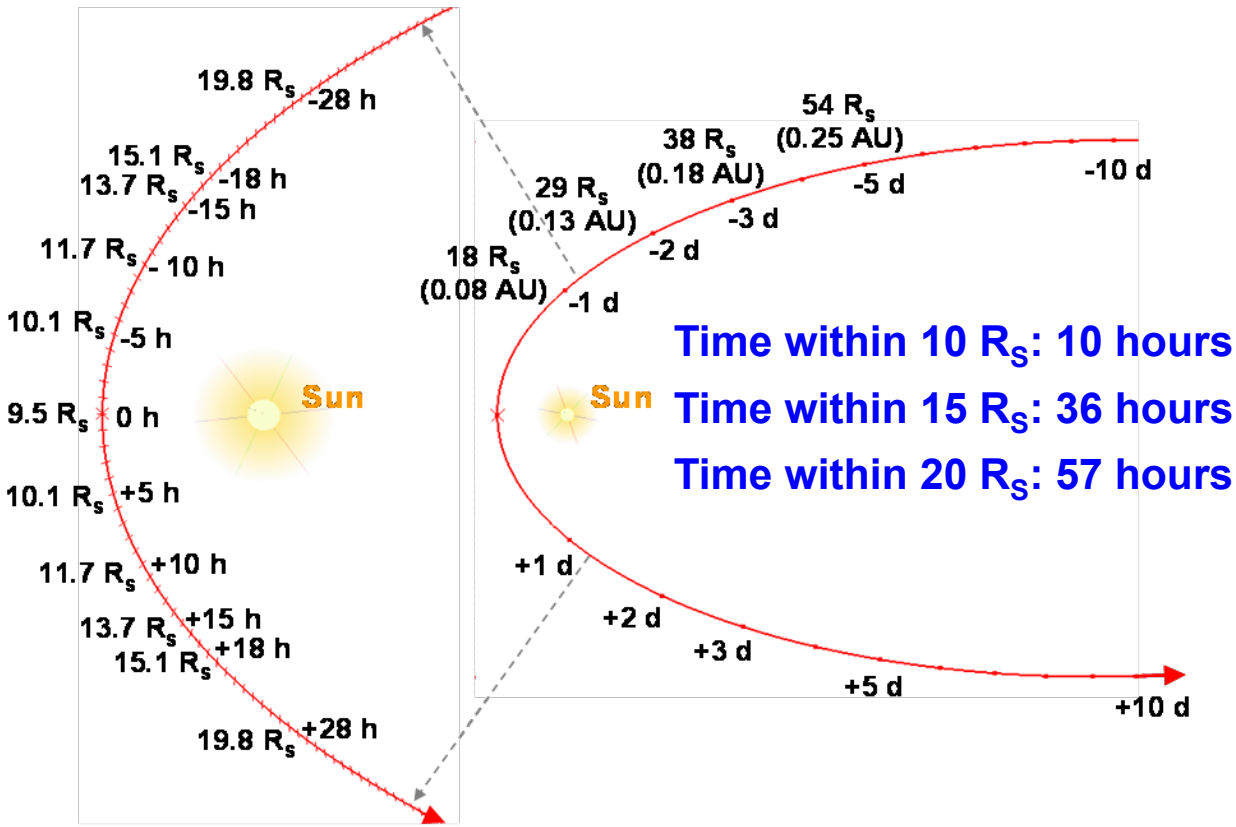
24 Solar Passes: Perihelion Gradually Decreasing



Solar Pass #	Perihelion (Rs)
1	35.0
2	35.0
3	35.0
4	27.5
5	27.5
6	19.9
7	19.9
8	15.5
9	15.5
10	13.0
11	13.0
12	13.0
13	13.0
14	13.0
15	13.0
16	13.0
17	11.3
18	11.3
19	11.3
20	11.3
21	11.3
22	9.5
23	9.5
24	9.5

Near Sun Trajectory

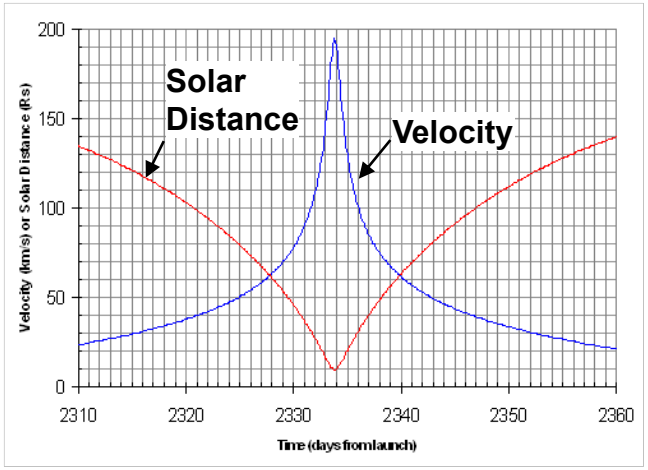
The 9.5-R_S Pass (P22)

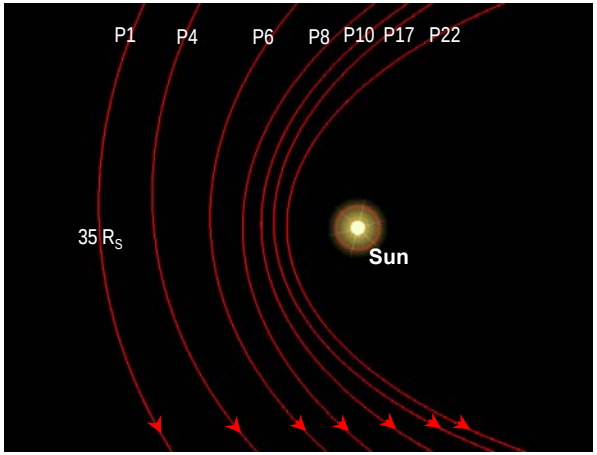


Time within 10 R_S: 10 hours

Time within 15 R_S: 36 hours

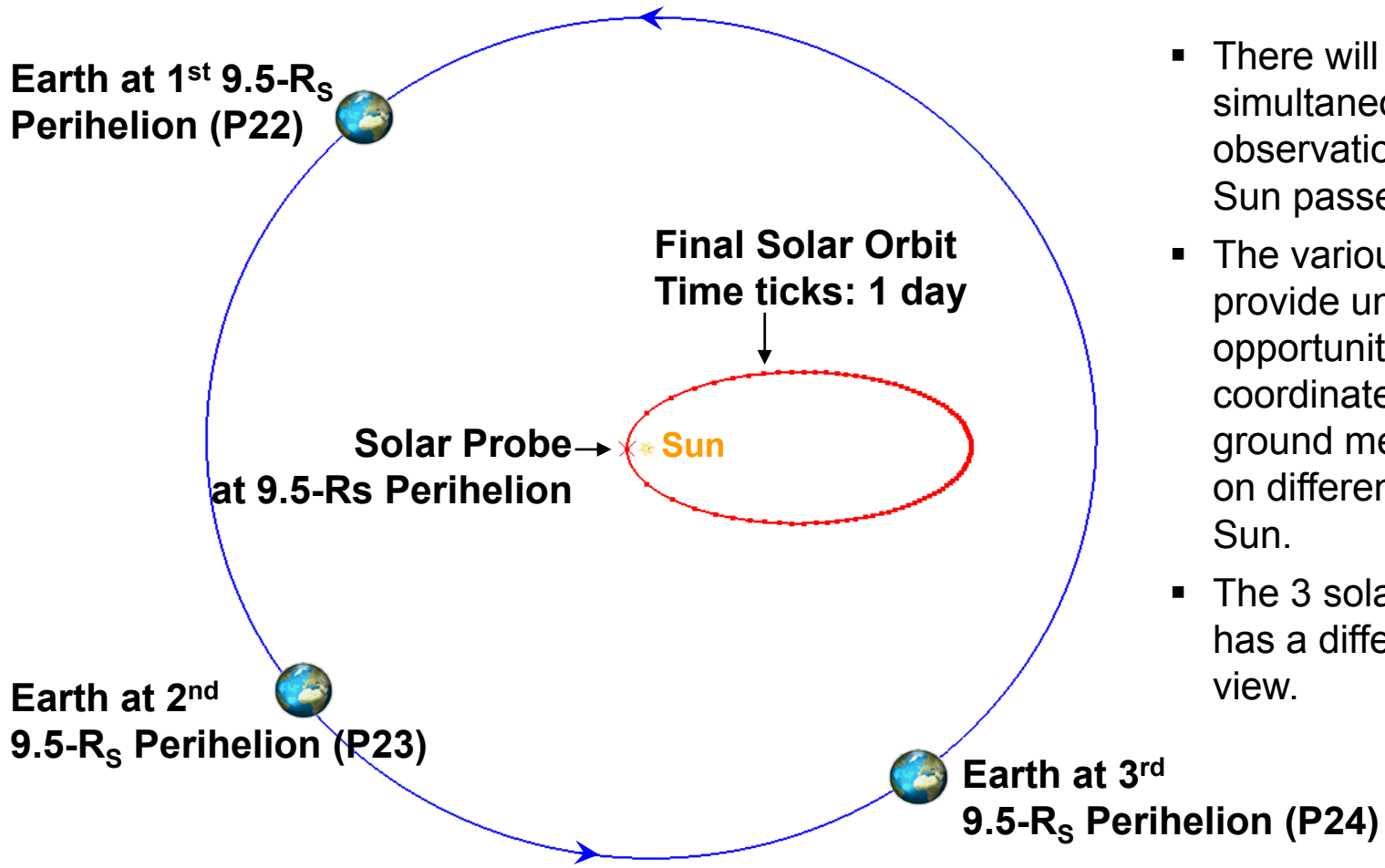
Time within 20 R_S: 57 hours





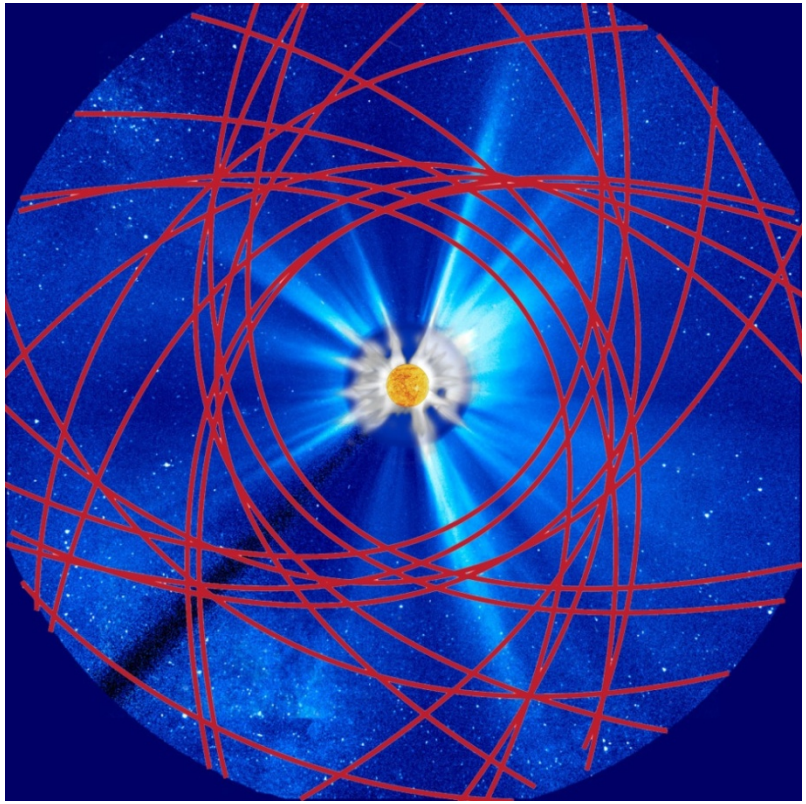
View from the North Pole

Earth Views of Various Solar Flybys (Earth Positions at Minimum Perihelia)



- There will be simultaneous ground observations of the near-Sun passes.
- The various solar passes provide unprecedented opportunities for coordinated in situ and ground measurements on different part of the Sun.
- The 3 solar passes each has a different Earth view.

Solar Passes within 30 R_S

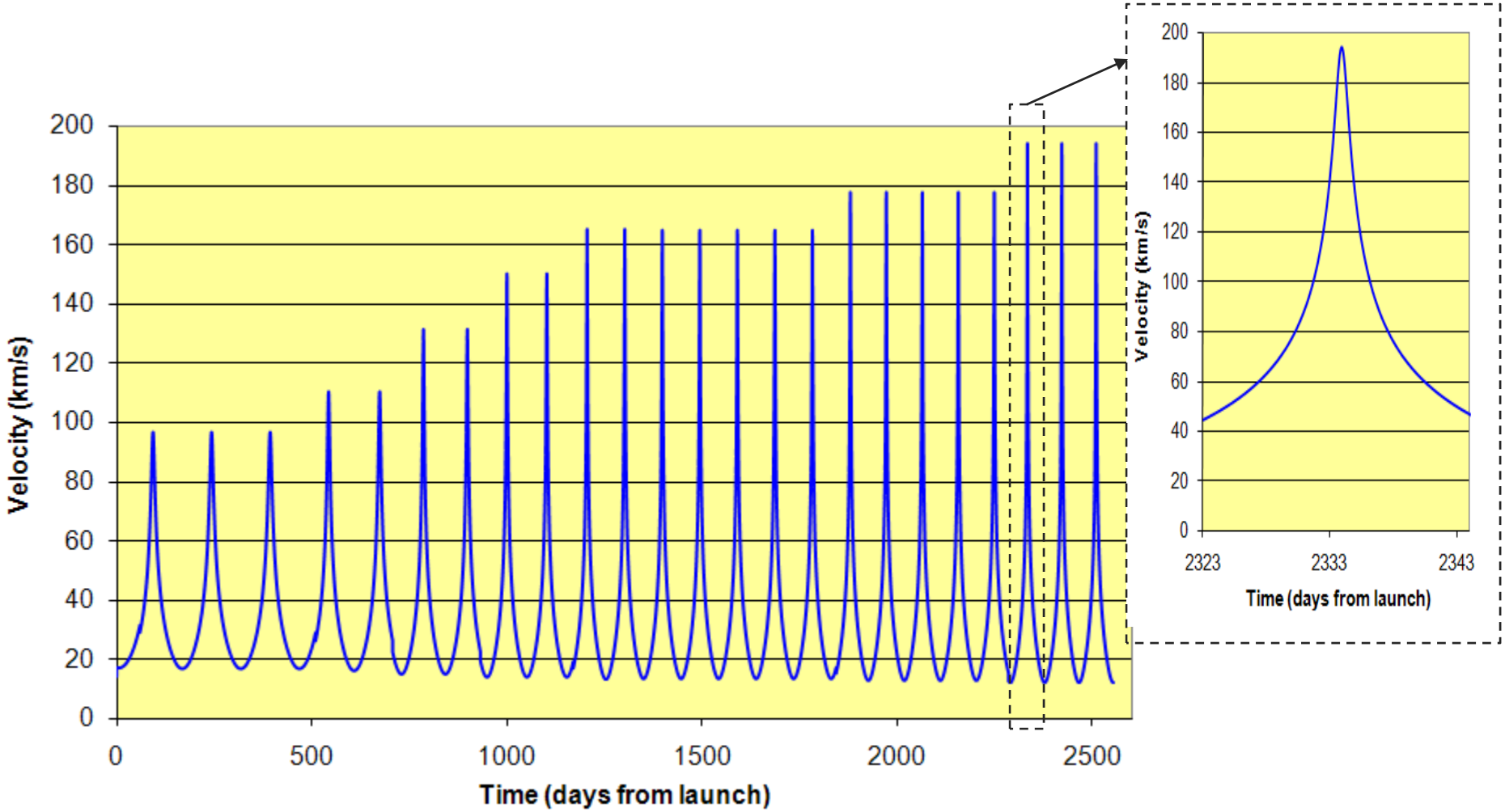


21 Solar passes (red lines) within 30 R_S, plotted with respect to the Earth. Earth is off the bottom of the figure.

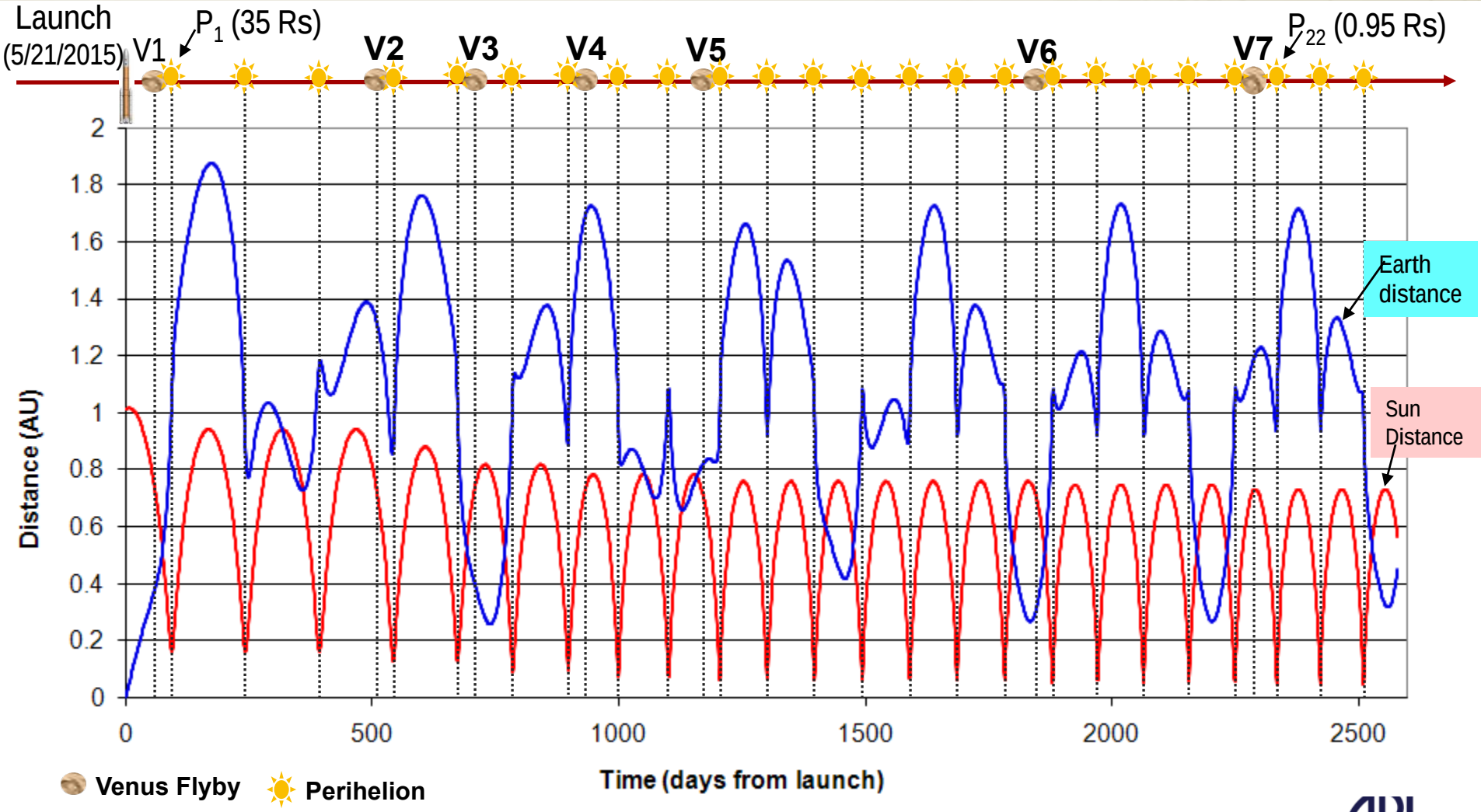
	Time within			
Solar Pass #	30 Rs (hr)	20 Rs (hr)	15 Rs (hr)	10 Rs (hr)
4	67			
5	67			
6	105	10		
7	105	10		
8	109	50		
9	109	50		
10	108	55	23	
11	108	55	23	
12	108	55	23	
13	108	55	23	
14	108	55	23	
15	108	55	23	
16	108	55	23	
17	105	57	33	
18	105	57	33	
19	105	57	33	
20	105	57	33	
21	105	57	33	
22	102	57	36	10
23	102	57	36	10
24	102	57	36	10
Total	2149	961	434	30

2149 hours within 30 R_S and **961 hours** within 20 R_S which is 10 times more than previous Solar Probe would have (96 hours in 2 passes)

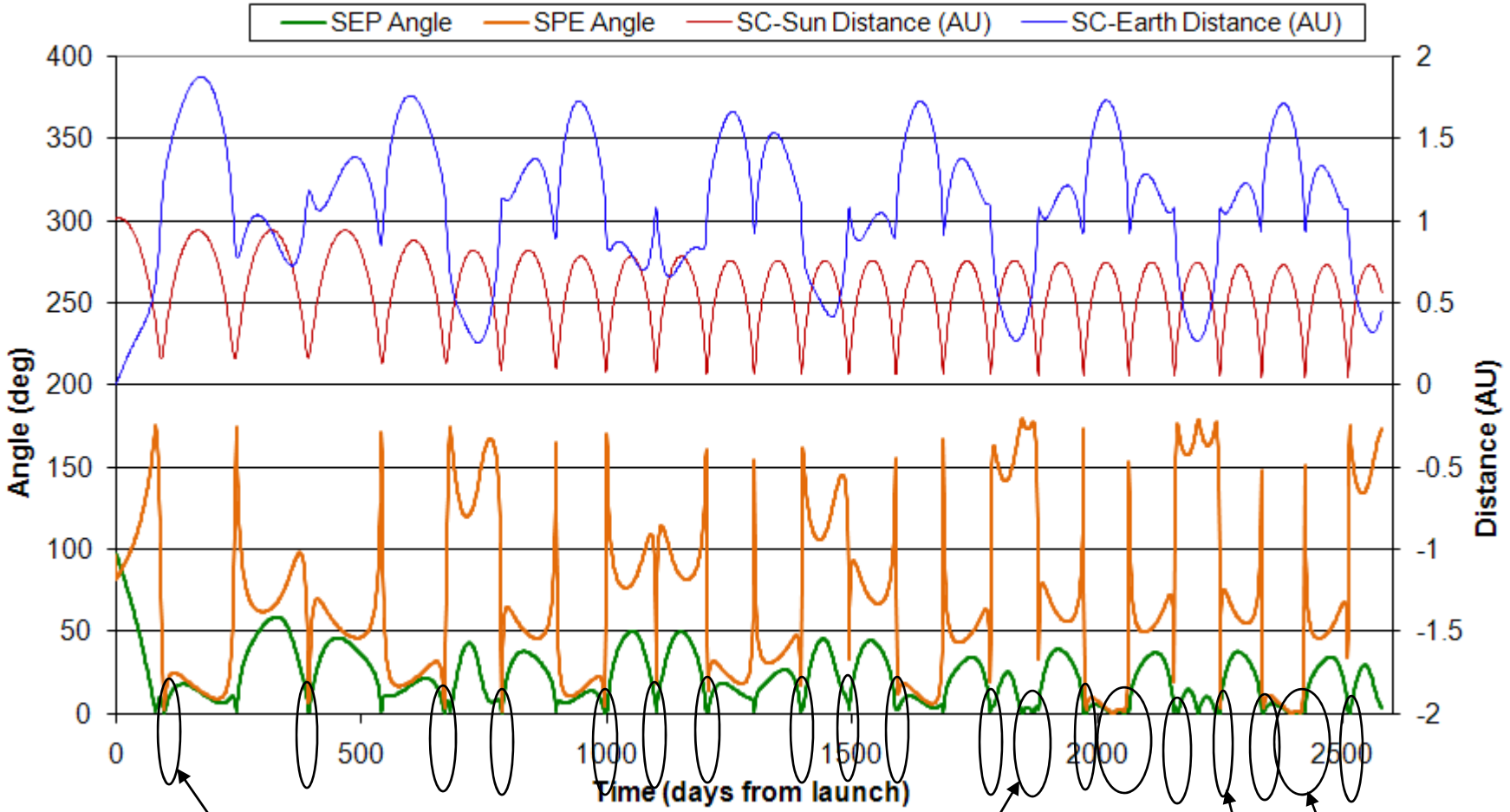
Solar Probe Plus Velocity Profile



Solar Probe Plus
Mission Profile and Timeline



Solar Conjunction

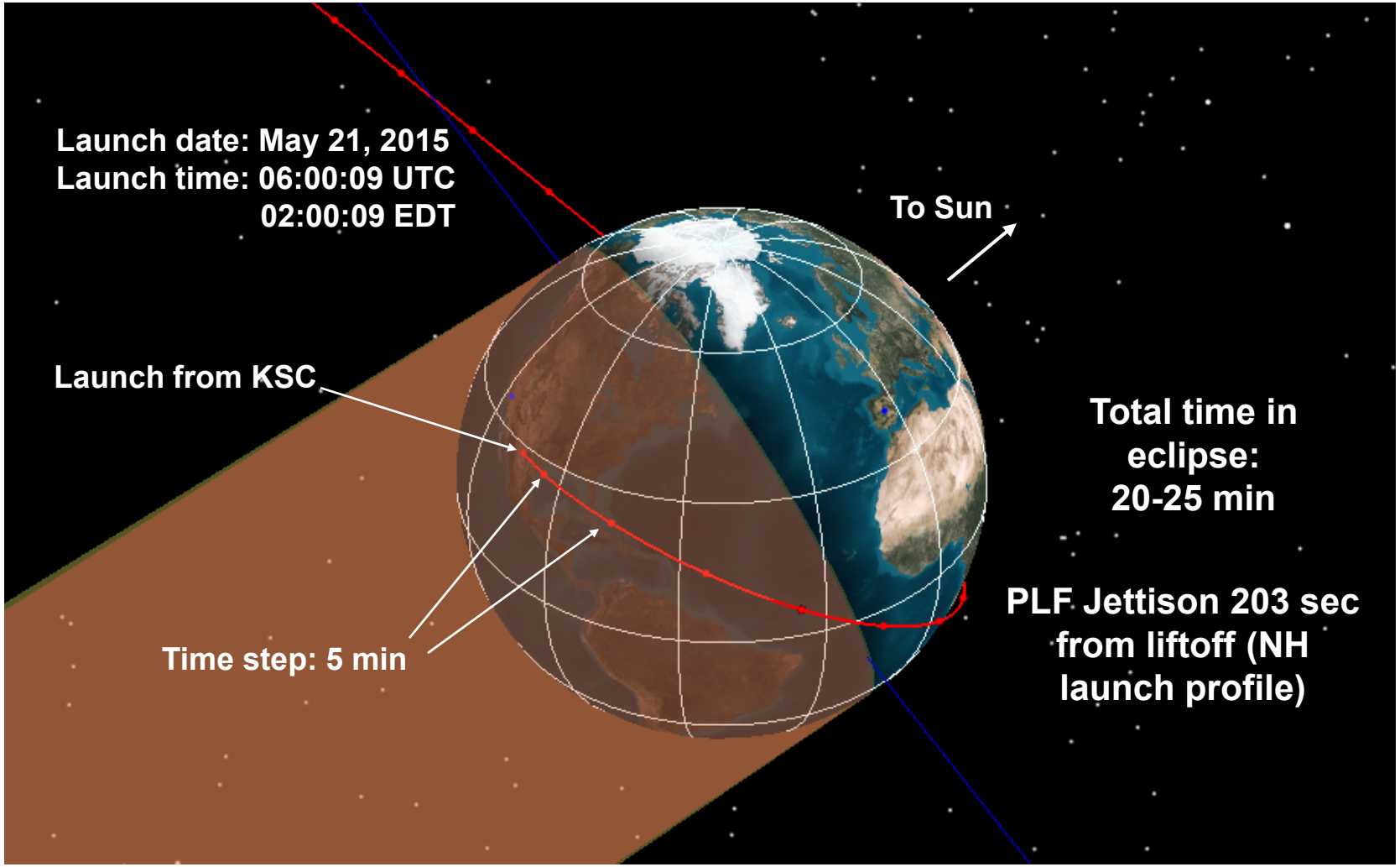


Solar Conjunction Periods (SEP angle < 3° (X), SEP angle < 1° (Ka))

Longer duration

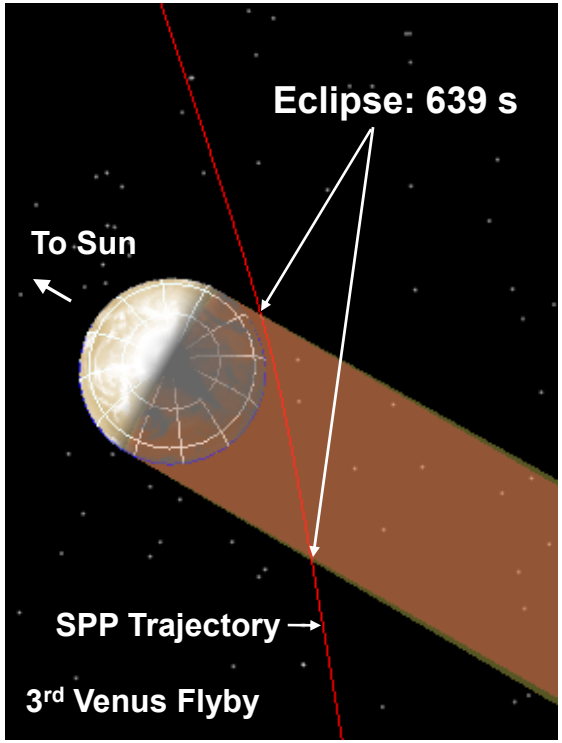
Short duration

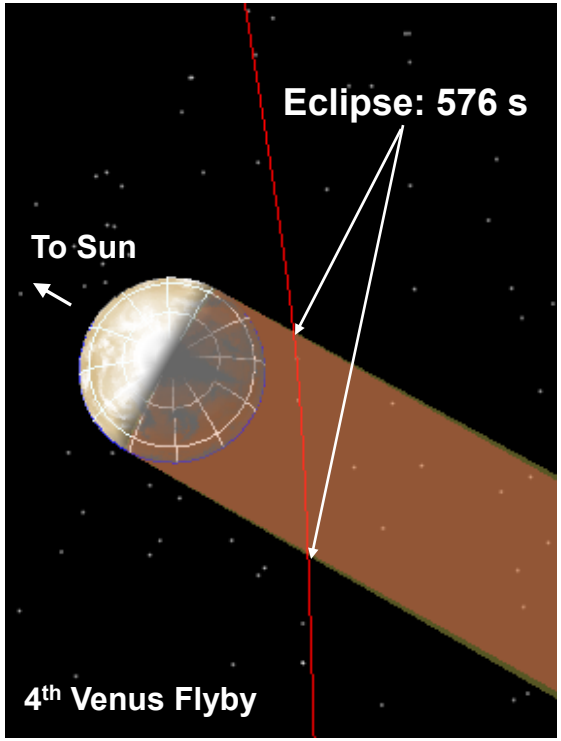
Solar Eclipse: During Launch

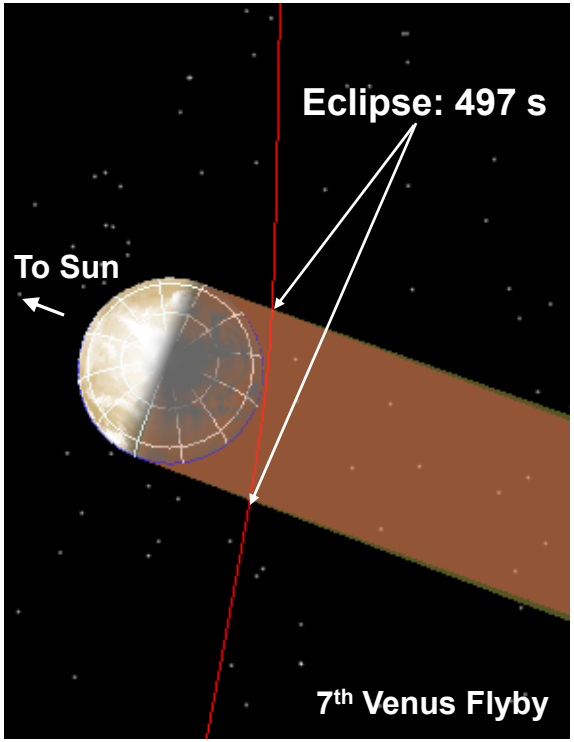


Solar Eclipse: Post Launch

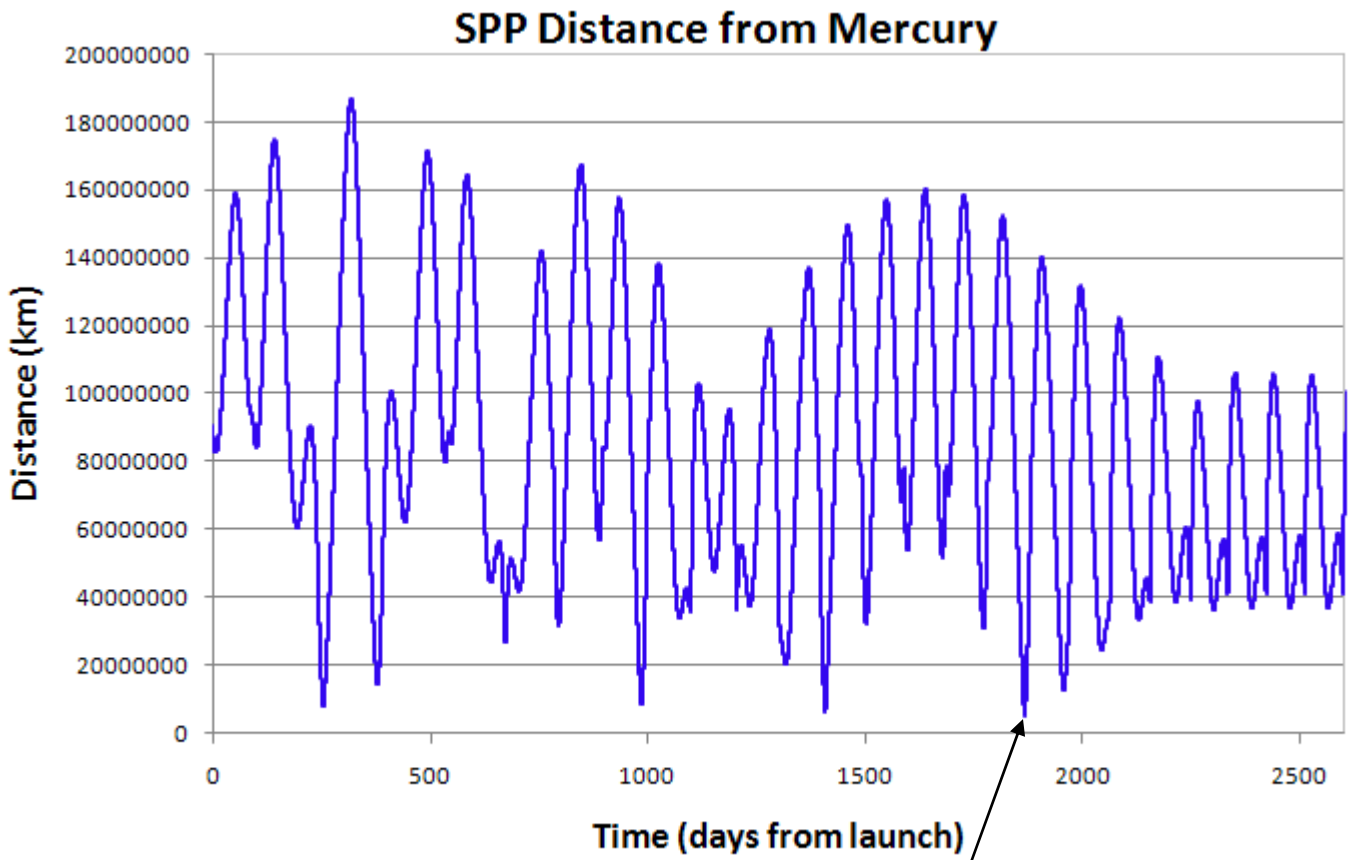
- Three brief solar eclipses will occur during 3 of the 7 Venus flybys
- Maximum duration of eclipse: 639 sec







Possibility for Close Encounter with Mercury?



Closest approach distance to Mercury is 5.1 million km

Flight Path Control and Trajectory Correction Concept

- Solar Probe Plus will follow the V^7GA trajectory after departure from Earth.
- There are no deep space maneuvers (deterministic ΔVs). Trajectory correction maneuvers (TCMs) may be needed for correcting launch errors and for targeting the Venus flybys.
- Sufficient ΔVs are allocated for this purpose.
- Venus flyby altitudes are much great than 300 km, except for the last one which will be at about 300 km.
- All Venus flybys are un-powered.
- There are no solar conjunctions during any of the 7 Venus flybys. TCMs for targeting the desired Venus B-plane aim point and for post-flyby correction can be placed if necessary.
- A mono-propulsion system with 18 thrusters will provide the required TCMs.

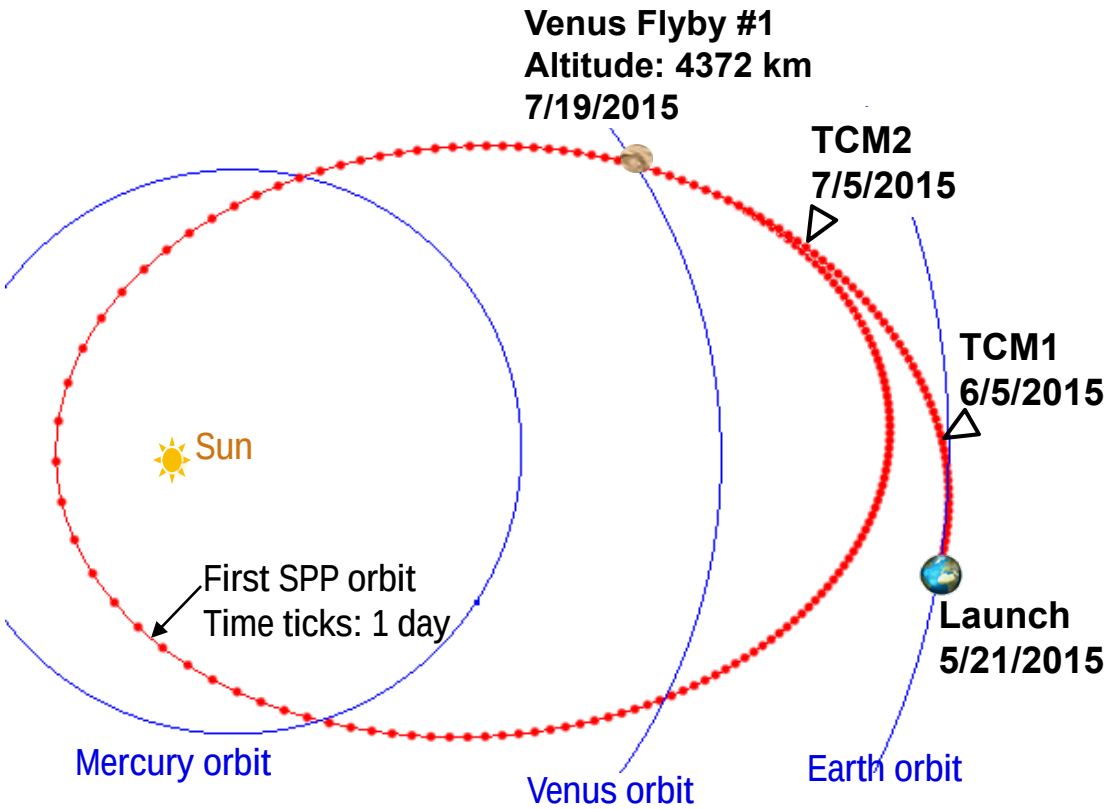
Preliminary Delta-V Allocation

- There are no deep space maneuvers, and no deterministic ΔV is required.
- All trajectory maneuver ΔVs are statistical and are allocated for navigation. They are estimated based on ΔVs budgeted and actually used in New Horizons and MESSENGER missions.
- Detailed Monte Carlo analysis is planned in Phase-A to further refine the ΔV requirements.

Usage	Event	ΔV (m/s)
Trajectory Correction Maneuver	Launch error correction	90
	Venus Flyby 1	8
	Venus Flyby 2	12
	Venus Flyby 3	12
	Venus Flyby 4	12
	Venus Flyby 5	12
	Venus Flyby 6	12
	Venus Flyby 7	12
Attitude Maneuver	Attitude control	43
Unallocated		17
Total Delta-V		230

Preliminary TCM Schedule

Event*	Date
Launch	5/21/2015
TCM1	6/5/2015
TCM2	7/5/2015
Venus Flyby #1	7/19/2015
TCM3	9/20/2016
Venus Flyby #2	10/11/2016
TCM4	1/25/2017
TCM5	4/16/2017
Venus Flyby #3	4/26/2017
TCM6	9/10/2017
TCM7	11/27/2017
Venus Flyby #4	12/7/2017
TCM8	7/11/2018
Venus Flyby #5	8/1/2018
TCM9	8/9/2019
TCM10	5/15/2020
Venus Flyby #6	6/5/2020
TCM11	5/23/2021
TCM12	8/12/2021
Venus Flyby #7	8/22/2021



TCMs planned for the first orbit

* Not a complete TCM schedule, main events used for checking solar conjunction and solar distances.
A refined TCM schedule will be developed in Phase -A in coordination with the Monte Carlo ΔV analysis.

Navigation Concept

- Navigation has no special requirements and will be straight forward:
  - Optical navigation is not required
  - Traditional radiometric Doppler range and range rate data will be used for spacecraft trajectory determination
  - Delta-DOR tracking data will be used before Venus flybys to enhance OD accuracy
- DSN tracking schedule for orbit determination:
  - Launch and Early Operations:
    - L to L+2 weeks: continuous
    - L+2 to L+4 weeks: five 10-hour passes per week
  - Venus Flyby
    - V-5 to V-1 week: five 10-hour passes per week
    - V-1 to V+1 week: one 10-hour passes per day
  - Other time
    - 3 passes per week

Summary of Mission Designs for Alternative Launch Opportunities

- **New types of trajectories have been developed, along with the baseline trajectory, enabling Solar Probe Plus to launch yearly from 2015 through 2018:**
  - **2015: baseline launch, via a seven-Venus-gravity-assist (V^7GA) trajectory**
  - **2016: backup launch, via an eight-Venus-gravity-assist (V^8GA) trajectory**
  - **2017: 2nd backup launch, via a one-Earth-seven-Venus-gravity-assist (EV^7GA) trajectory**
  - **2018: 3rd backup launch, via a V^7GA trajectory**

Launch Year	2015	2016	2017	2018
Mission Design	**V^7GA**	**V^8GA**	**EV^7GA**	**V^7GA**
Launch C3 (km^2/s^2)	158	154	158	159
Launch Period (days)	20	20	28	20
Number of Flybys	7	8	8	7
Minimum Perihelion	9.5 Rs	9.5 Rs	9.5 Rs	9.5 Rs
Total # of Passes	24	27	26	24
# of 9.5-Rs Passes	3	3	3	3
Mission Duration (years)	6.9	8.3	7.9	6.9
Date of 1st Min. Perihelion	Oct 2021	Apr 2024	Dec 2024	Dec 2024

Summary

- The Solar Probe Plus mission concept based on the V^7GA trajectory design accomplishes the science objectives and is technically feasible under the NASA ground rules.

- The Solar Probe Plus mission design offers significant advantages in both science return and technical implementation as compared to previous Solar Probe.

- Backup and alternative launch opportunities are identified, and viable mission designs are developed that enables Solar Probe Plus to launch in 2015, 2016, 2017, or 2018, with the same spacecraft design.

Solar Probe Plus: Environments

R. B. Decker

The Johns Hopkins University
APPLIED PHYSICS LABORATORY

Environmental effects investigated

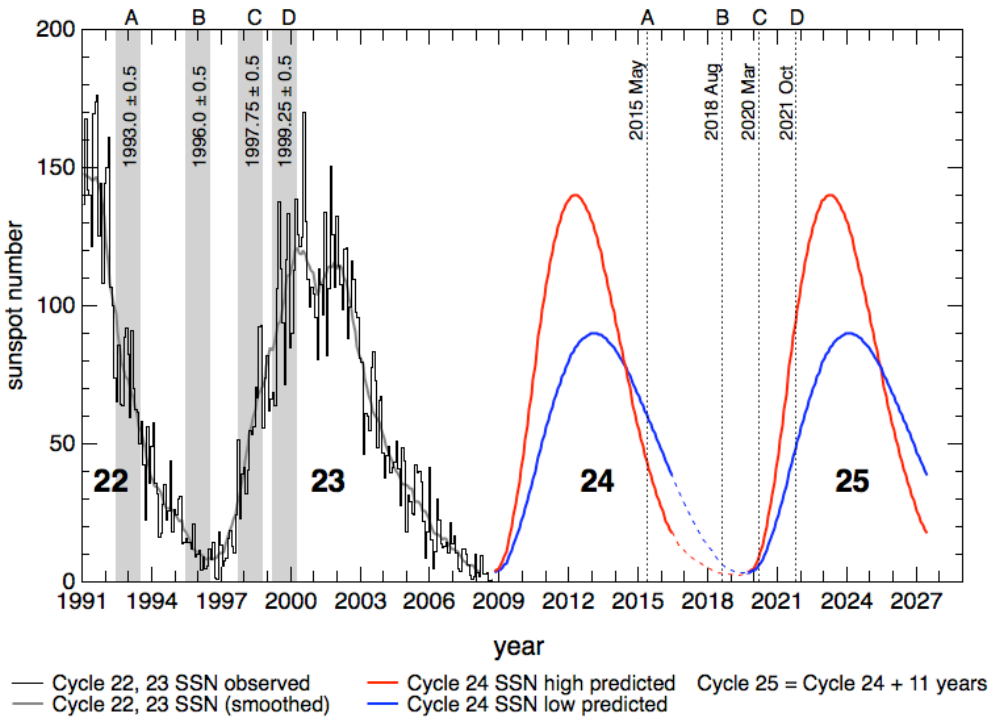
1. Increased radiation exposure due to solar activity
   a) Solar energetic particles (SEPs)
   b) Solar electromagnetic (em) radiation
2. TPS mass loss due to solar radiation (thermal vaporization)
3. TPS mass loss due to interplanetary dust impacts
4. Spacecraft charging
5. Nominal and disturbed solar wind
   a) Plasma pressures
   b) Magnetic torques

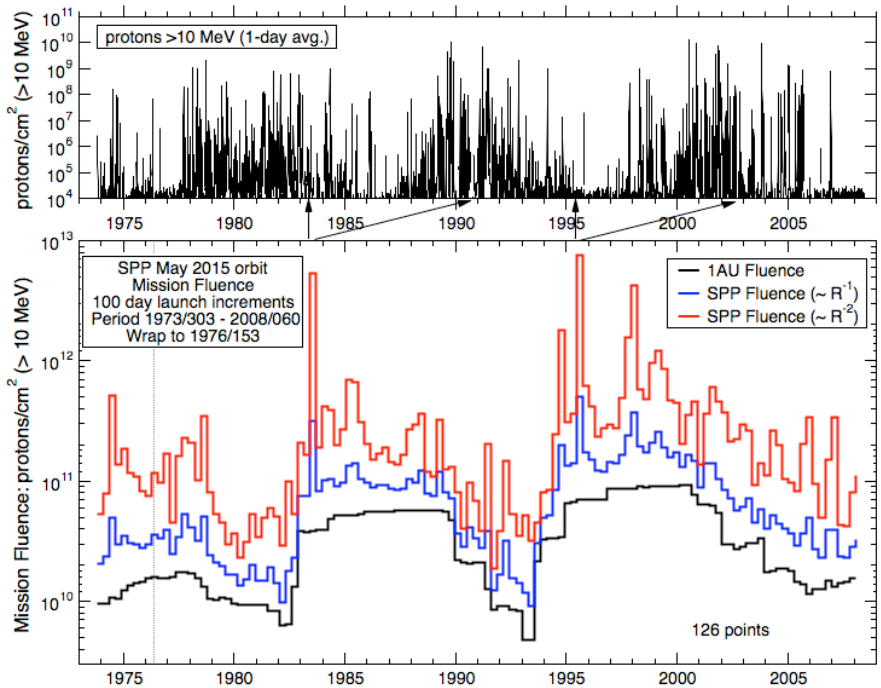
1a. Solar energetic particles (SEPs)

- Problem:

  Estimate mission-integrated fluence of SEP ions and electrons from coronal mass ejections (CMEs) and solar flares
- Method:
  - Use proton and electron fluence data in several energy channels as measured by multiple near-Earth spacecraft during solar cycles 21-23 (1974-2008)
  - Accumulate fluences along SPP's 7-year orbit for specified radial variation of fluence (~ r^{-a}) for 365 daily-spaced launches centered on chosen launch date
  - Use resulting log-fluence frequency histogram for chosen launch date to estimate mission integrated fluence at 95% confidence level
- Results:
  - Generic SPP hardness requirement set at 37 krads (Si) in 100 mils Al

SEP proton fluences: Sample model results and launch dates





- Left: Launch intervals (shaded) during solar cycles 23-24 used to model SPP launches during similar periods of solar activity in cycles 24-25 (2015-2021)
- Right: Mission-integrated fluence of >10 MeV protons for SPP launch every 100 days

1b. Solar electromagnetic (em) radiation

- Problem: Calculate background total solar irradiance (TSI) and its variability due to transient solar activity over SPP mission; estimate worst-case flare-associated flux of extreme em radiation during SPP perihelia
- Method:
  - Calculate TSI as a function of helioradius; include limb darkening effect and its variability for varying helioradius
  - Perform study on well-observed X17 flare (very close to worst-case flare scenario) and reproduce flare's em spectrum; assume S/C at closest perihelion when flare occurs; calculate:
    a) additional TSI due to event
    b) fluxes of high-energy X- and gamma-ray photons
- Results:
  - TSI variation over SPP mission: 1 - 500 suns
  - Contribution of X17 flare to TSI is insignificant even at 9.5 R_S

2. TPS mass loss due to solar em radiation (thermal vaporization)

- Problem:
  - Solar em radiation heats TPS, makes it susceptible to mass loss via thermal vaporization
  - Calculate mass loss rate; determine if it poses threat for TPS or for smooth operation of instrument payload
- Method:
  - Calculate TPS temperature from TSI; use vapor pressure from experiments, apply to Hertz-Knudsen-Langmuir formula to calculate mass loss rate
  - Revise and update Goldstein et al. (1980) analysis
    a) Get TPS upper mass loss rate requirement for smooth instrument operation
    b) Compare calculated mass loss rate with upper requirement
  - Integrate mass loss rate over mission duration to calculate total mass loss
- Results:
  - Upper mass loss rate requirement revised to 0.5 mg s^{-1} (was 2.5 mg s^{-1})
  - Peak mass loss rates are 160-600 times smaller than upper requirement
  - Mission integrated total mass loss ~ 0.68 g (no threat to TPS)

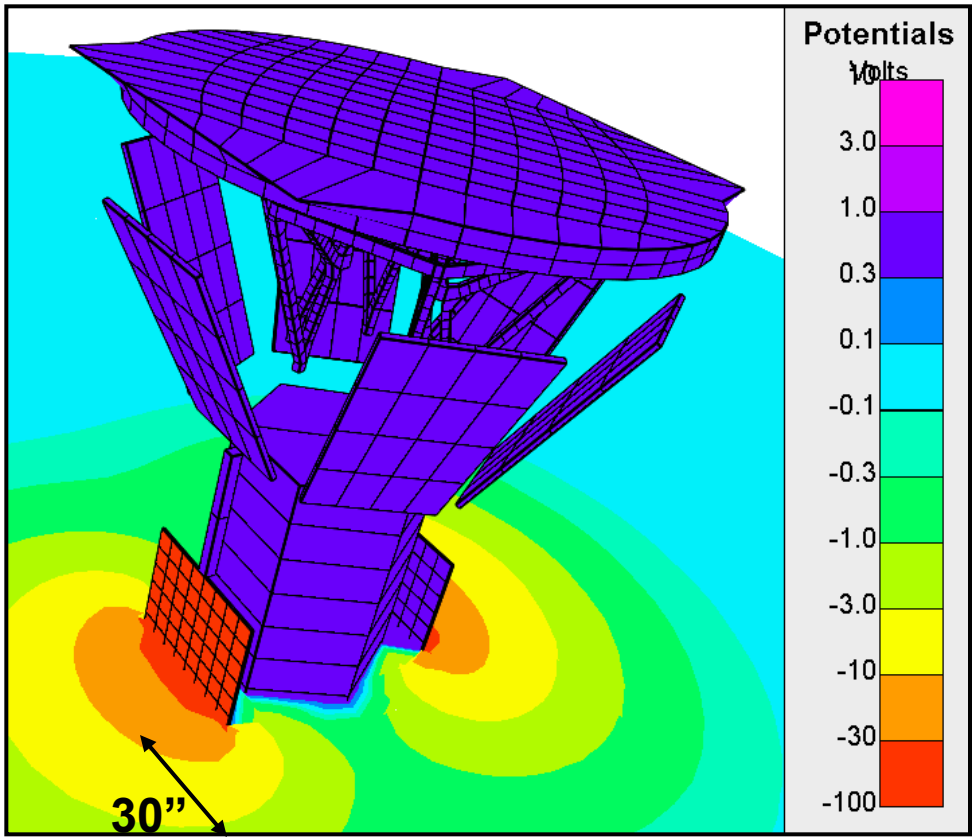
3. TPS mass loss due to interplanetary dust impacts

- Problem:
  - Inner heliosphere shows highest density of dust particles; hyper-velocity collisions lead to spallation of TPS material
  - Calculate mass loss rate due to dust impact on TPS; determine if it poses threat for TPS or for smooth operation of instrument payload
- Method:
  - Construct a hybrid dust particle distribution function (w/o collisions) to achieve highest possible flux of dust particles
  - Calculate worst-case single-particle collision scenarios, where all kinetic energy of particle results in mass loss
  - Calculate mass loss rate by multiplying single-particle effect with particle flux for various particle masses; calculate mission-integrated mass loss
- Results:
  - Peak mass loss rates are 500-1000 times less than upper limit of 0.5 mg s^{-1}
  - Mission integrated total mass loss ~ 0.7 g (no threat to TPS)
  - Equivalent dust-impact affected area on TPS ~ 7.8 cm^2 (insignificant)

4. Spacecraft charging: Initial results from simplified model

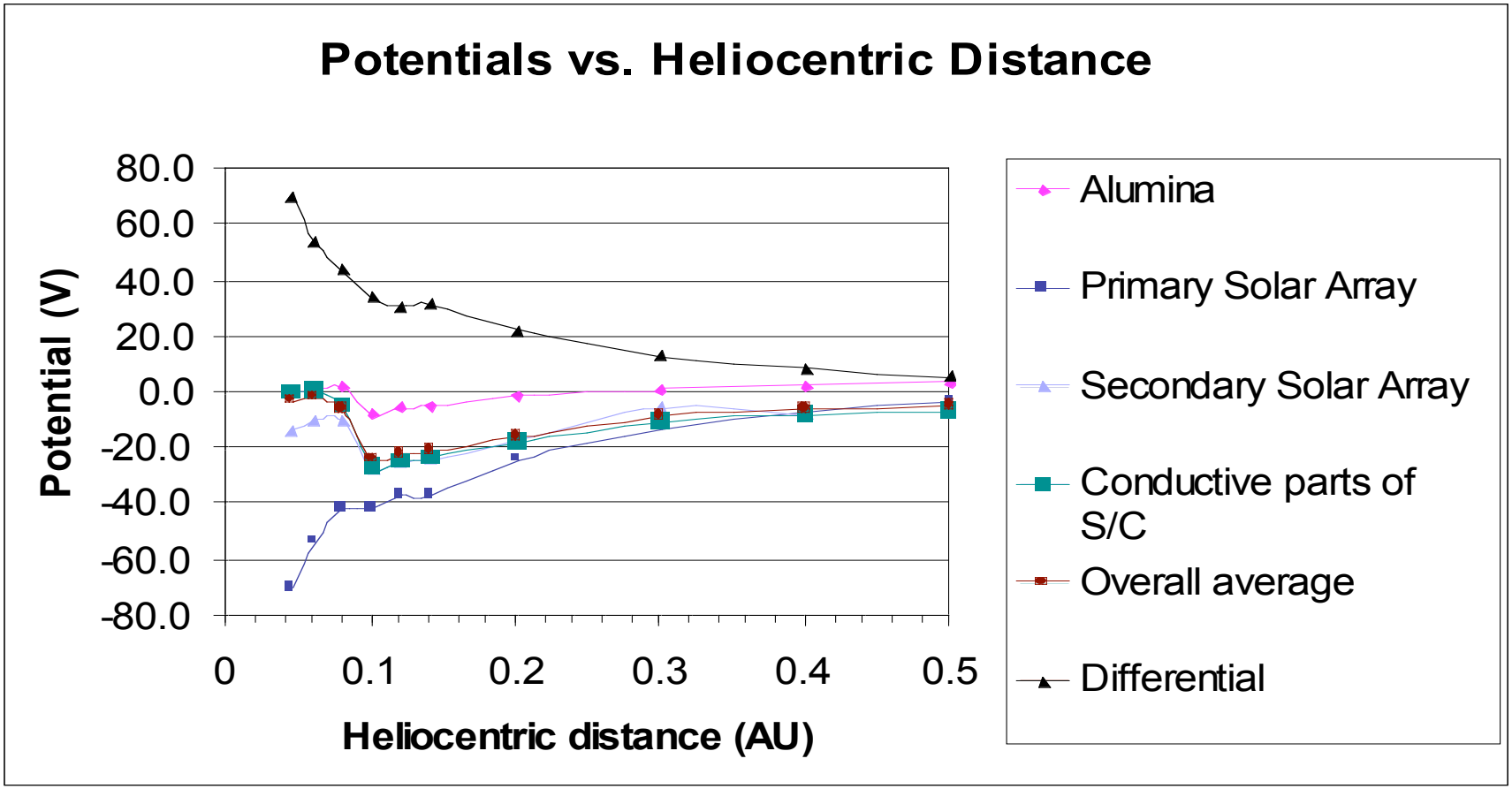
- Problem: Estimate degree of spacecraft surface charging expected and its effect on the potentials near the spacecraft
- Method:
  - Use conservative parameters to model solar wind plasma environment at near-solar trajectory points.
  - Build spacecraft model based on available information
  - Collect best available estimates of material properties at temperatures expected in near-solar environment
  - Perform spacecraft charging analysis using NASCAP software (industry standard)
- Initial results:
  - Overall surface charging relative to plasma ground remains within 100V for trajectory points in range 0.0442 to 0.5 AU
  - Effect on space potentials is largest near solar arrays and much smaller near other bus panels
  - Reducing resistivity of coverglass greatly reduces charging of solar arrays

Potentials near the spacecraft at perihelion: Initial results from simplified model

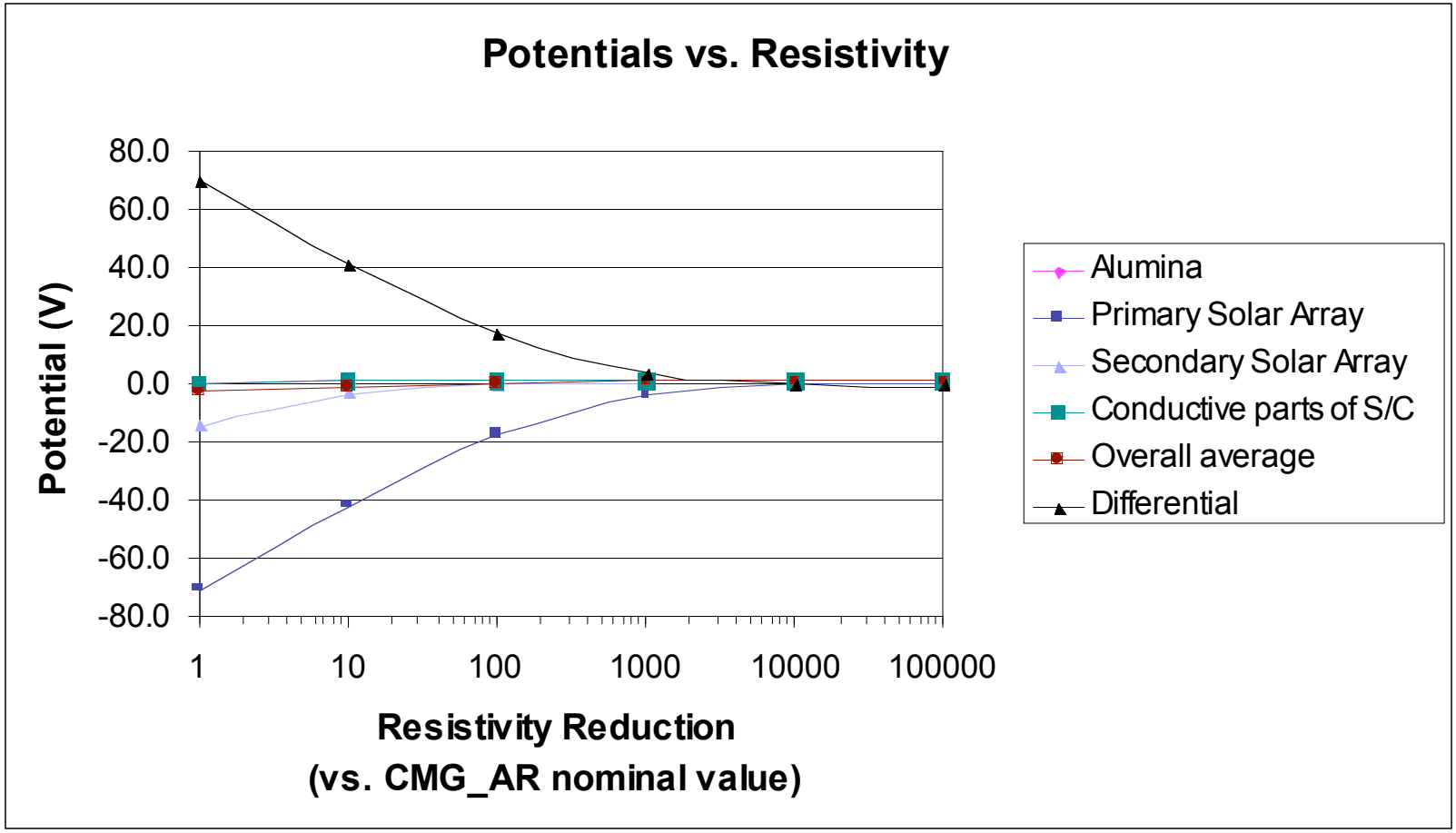


- Initial results of spacecraft charging for conservative environmental conditions and simplified properties of spacecraft components and materials
- At perihelion, sun-exposed TPS is essentially conductive and driven slightly positive via photoemission.
- Bus, struts, radiators, & back sides of solar arrays are conductive and equilibrate to same potential as TPS
- Front sides of solar arrays are shadowed & coverglass is highly electrically resistive, so they are driven to a negative potential of as much as 75V.
- The negative potentials on solar array coverglass surface produce negative space potentials near spacecraft that are large only near solar array surfaces
- Potentials near other bus panels are <10V

Spacecraft charging over range 0.0442 AU to 0.5 AU: Initial results from simplified model



Effects of reducing S/A coverglass resistivity: Initial results from simplified model



5a. Nominal and disturbed solar wind: Plasma pressures

- Problem: Estimate momentum transfer rate to SPP vs. radius *r* due to nominal solar wind pressure & to near-Sun disturbances such as CMEs
- Method:
  - Use published data and models that estimate *r*-dependence of key solar wind parameters, extrapolate parameters to $r < 0.3$ AU
  - Use data- and model-based predictions of CMEs (velocity, width, structure) to estimate momentum transfer to SPP from CMEs encountered near perihelia
- Results:
  - During its mission, SPP will experience pressures from nominal solar wind that are factors of 10^3 - 10^4 times smaller than radiation pressure (e.g., $P_{ph} = 3\times10^{-3}$ N m^{-2} at 9.5 R_S)
  - Impact with a CME at or near perihelion will change momentum of SPP by an insignificant fraction, $|\Delta P/P|_{SPP}$ ~1 part in 10^7-10^8

5b. Nominal and disturbed solar wind: Magnetic torques

- Problem: Estimate magnetic torquing of SPP due to nominal and disturbed magnetic fields
- Method:
  - Model steady SPP current as that of simple current loop with current < mA (consistent with info. from MESSENGER MAG Team)
  - Estimate characteristic rotation time due to nominal and turbulent solar wind magnetic field & to impulsive disturbances like CMEs
- Results:
  - SPP will move rapidly enough through ambient magnetic field and through disturbances like CMEs that steady magnetic torquing of S/C pointing though a significant angle is highly unlikely
  - No significant problems for SPP G&C are expected

Summary (1)

1. Increased radiation exposure due to solar activity
   - SEPs Generic SPP hardness requirement set at 37 krads (Si) in 100 mils Al
   - Solar em radiation
     - TSI variation over SPP mission 1 - 500 suns
     - Contribution of X17 flare to TSI is insignificant even at 9.5 R_S
2. TPS mass loss due to solar em radiation
   - Peak mass loss rates are 160-600 times smaller than upper requirement.
   - Mission-integrated total mass loss ~ 0.68 g (no threat to TPS)
3. TPS mass loss due to interplanetary dust impacts
   - Peak mass loss rates are 500-1000 times less than upper limit of 0.5 mg s^{-1}
   - Mission integrated total mass loss ~ 0.7 g (no threat of any kind to TPS)
   - Equivalent dust-impact affected area on TPS ~ 7.8 cm^2 (insignificant)

Summary (2)

4. Spacecraft charging: Initial results from simplified model
   - Overall surface charging relative to plasma ground remains within 100 V for trajectory points in radial range 0.0442 to 0.5 AU
   - Effect on space potentials is largest near solar arrays and much smaller near other bus panels
   - Reducing resistivity of coverglass greatly reduces solar array charging
5. Nominal and disturbed solar wind
   - Plasma pressures
     - Nominal SW pressure ~10^3-10^4 times smaller than radiation pressure
     - Fractional impulse on SPP from CME impact at 9.5 R_S insignificant
   - Magnetic torquing (from nominal SW ***B***, CME ***B*,** etc.)
     - No significant problems for SPP's G&C expected

Solar Probe Plus
Mission Concept Review

Spacecraft – Part 1

MK Lockwood
Spacecraft System Engineer

The Johns Hopkins University
APPLIED PHYSICS LABORATORY

Outline

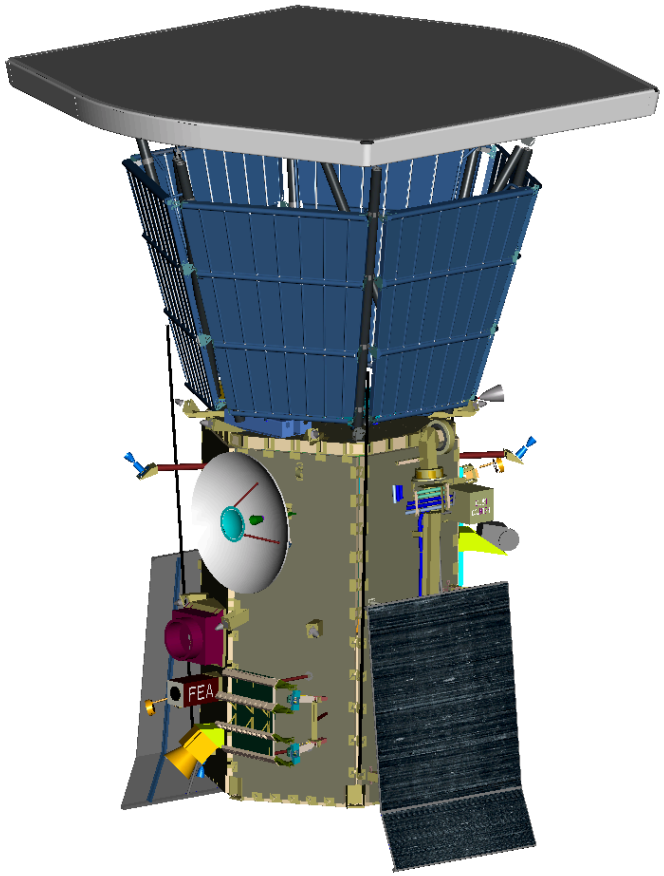
Part 1

- Reference vehicle intro
- Trades
- TPS function
- Solar array evolution
- Mechanism simplification
- Combined solar array
- Cooling system sizing and margin
- Reference vehicle
  - Block diagram
  - Mass, power

Part 2

- Communication
- Science data downlink
- S/C thermal results
- G&C, Cp-cg alignment
- Thermal safe mode
- Propulsion
- V&V solar encounter
- Margin summary
- TRL summary

Reference Vehicle Overview



- STDT 2008 instrument suite
- 610kg max launch wet mass
- Reference Dimensions:
  - S/C height: 3m
  - TPS max diameter:2.3m
  - s/c bus diameter: 1m
- C-C Thermal protection system
- Hexagonal prism s/c bus configuration
- Actively cooled solar power system
  - 343W electrical power at encounter
  - Solar array total area: 1.8m^2
  - Radiator area under TPS: 3.8m^2
- 0.6m HGA, 40W TWTA Ka-band science DL
- Science downlink rate: 138kb/s at 1AU
- Blowdown monoprop hydrazine propulsion
- Thrusters for attitude control
- Design Drivers: solar environ, mass, power

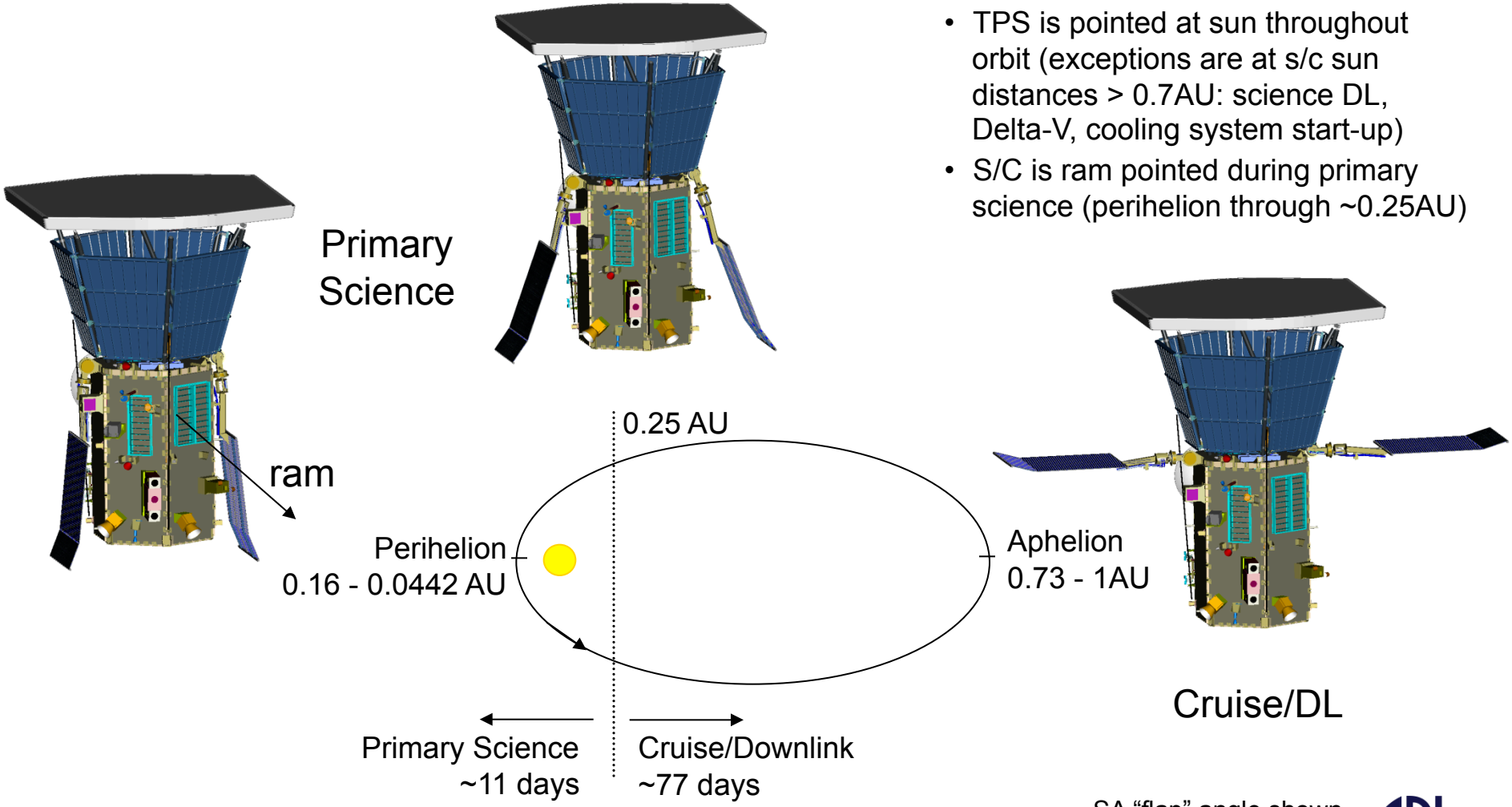
Instrument Accommodation Summary

- Instruments and accommodation requirements are based on the 2008 Solar Probe+ Study
- Reference Vehicle meets all instrument accommodation requirements
- Accommodation will be reviewed following instrument selection

Resource	Accommodation
Mass	47.2kg CBE
Power	57.2W CBE
Duty Cycle	100% duty cycle for orbit altitude < 0.25AU Off during science data downlink
Peak Data Rate	123.2 kbps
Data per Orbit	128Gbits/orbit including margin and housekeeping – final 3 orbits
FOV's	Met as 2008 SP+ Study
Pointing Direction	Ram direction is maintained
Jitter/Stability	Initial analyses indicate pointing budgets are met by Reference Vehicle

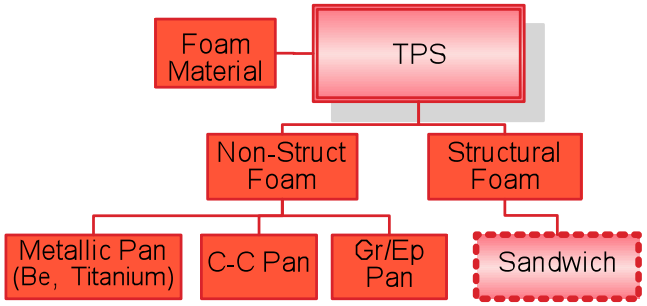
Science and Cruise Configurations

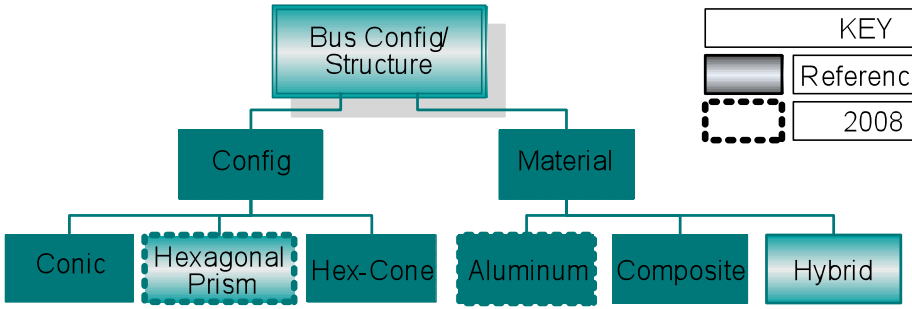
- TPS is pointed at sun throughout orbit (exceptions are at s/c sun distances > 0.7AU: science DL, Delta-V, cooling system start-up)
- S/C is ram pointed during primary science (perihelion through ~0.25AU)

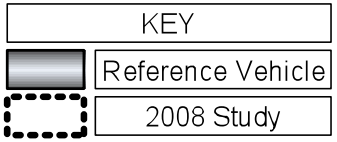


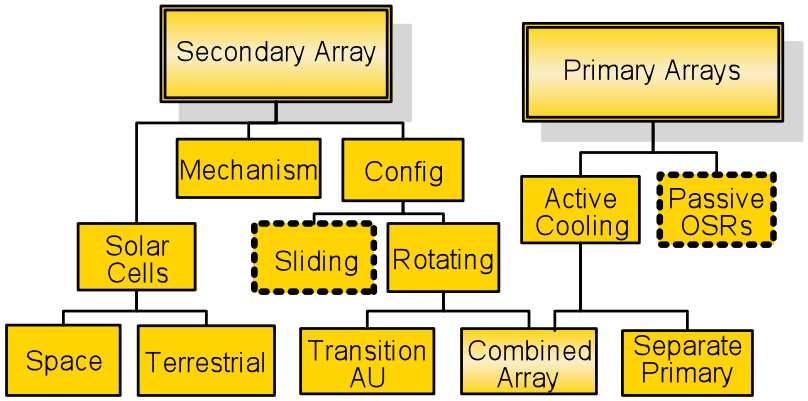
SA "flap" angle shown

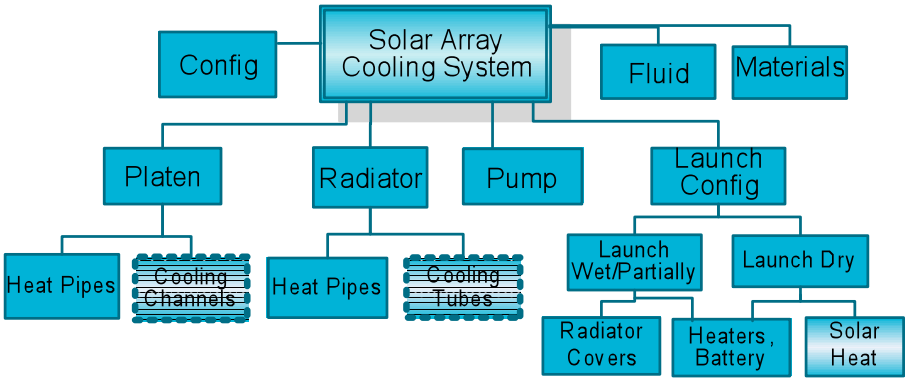
S/C Trades

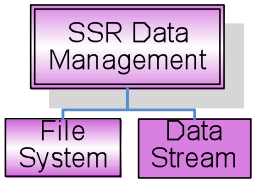

Foam Material
TPS
Non-Struct Foam
Structural Foam
Metallic Pan (Be, Titanium)
C-C Pan
Gr/Ep Pan
Sandwich

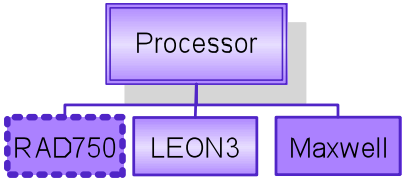

Bus Config/ Structure
Config
Material
Conic
Hexagonal Prism
Hex-Cone
Aluminum
Composite
Hybrid

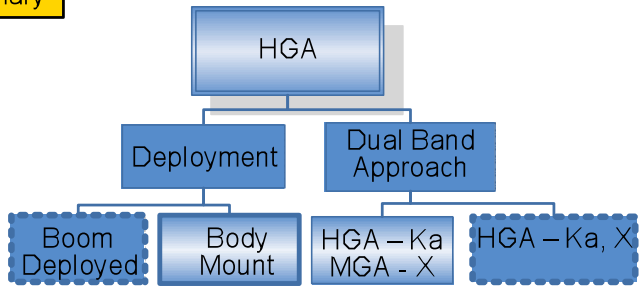

KEY
Reference Vehicle
2008 Study

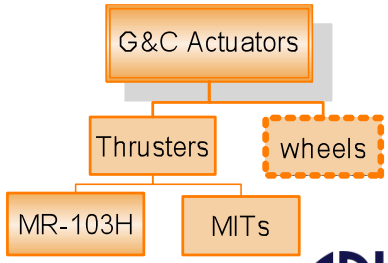

Secondary Array
Mechanism
Config
Solar Cells
Sliding
Rotating
Space
Terrestrial
Transition AU
Combined Array
Separate Primary
Primary Arrays
Active Cooling
Passive OSRs


Config
Solar Array Cooling System
Fluid
Materials
Platen
Radiator
Pump
Launch Config
Heat Pipes
Cooling Channels
Heat Pipes
Cooling Tubes
Launch Wet/Partially
Launch Dry
Radiator Covers
Heaters, Battery
Solar Heat

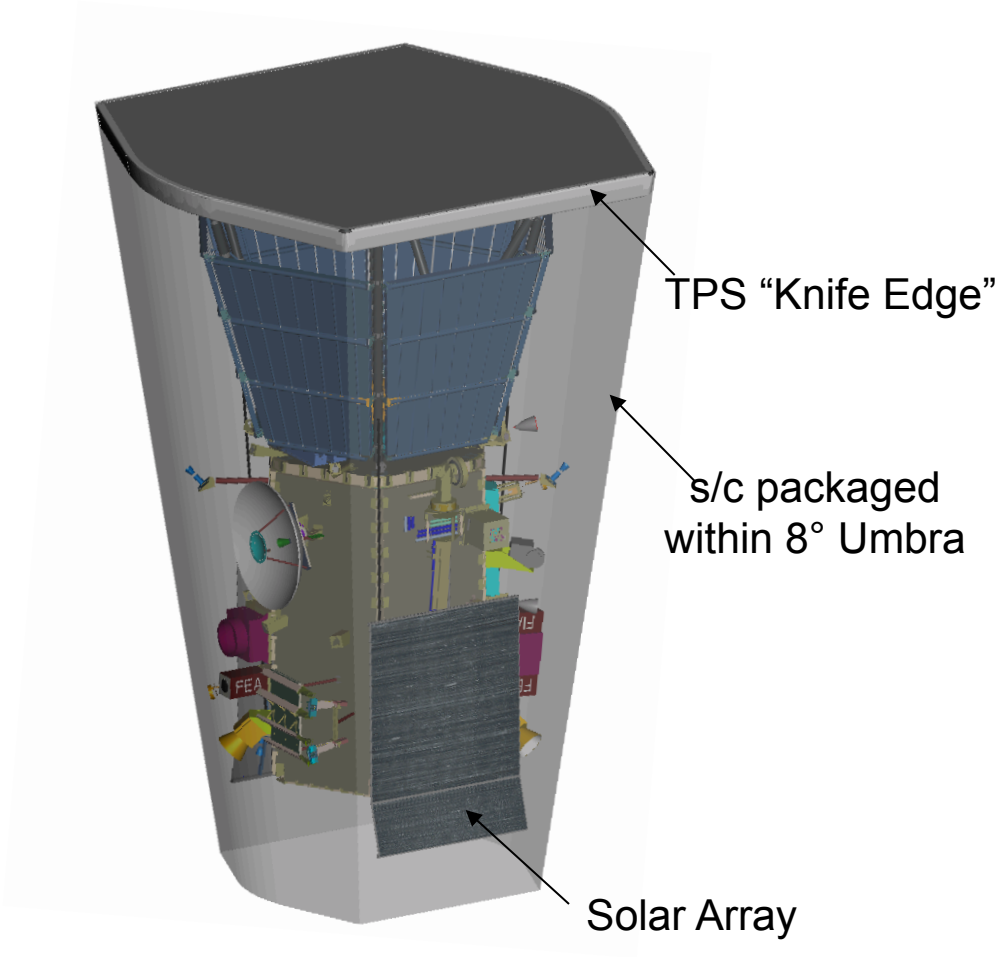

SSR Data Management
File System
Data Stream


Processor
RAD750
LEON3
Maxwell


HGA
Deployment
Dual Band Approach
Boom Deployed
Body Mount
HGA – Ka MGA - X
HGA – Ka, X


G&C Actuators
Thrusters
wheels
MR-103H
MITs

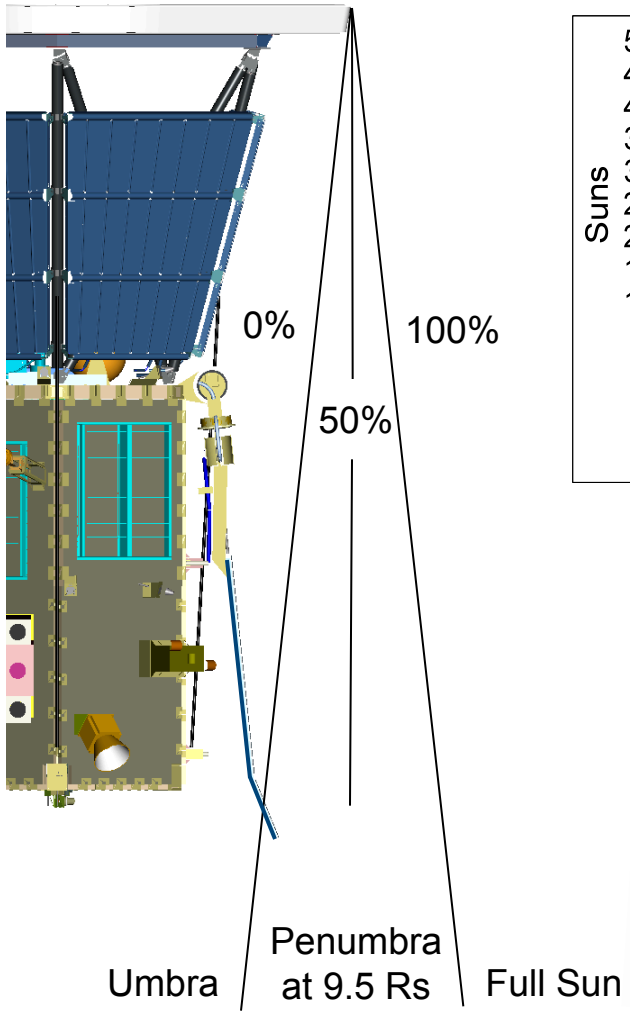
TPS

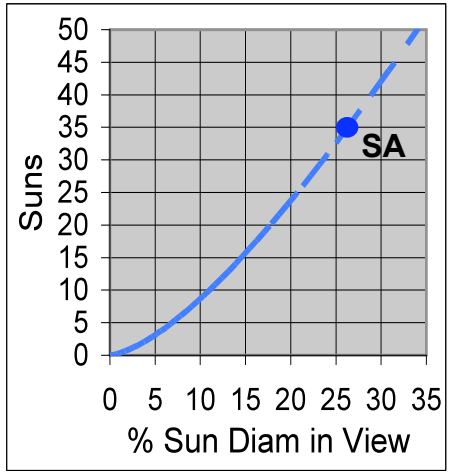


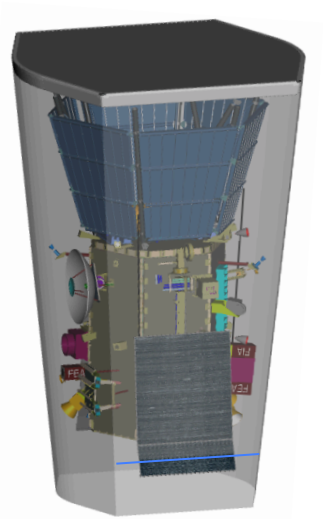
- TPS protects s/c from solar environment (512 suns at closest approach, 9.5Rs), enabling s/c components to operate within standard space-flight thermal environment
- TPS planform sized to ensure all s/c components remain within TPS umbra*
  - Umbra angle, 6°, is greatest at 9.5Rs
  - s/c packaged within 8° envelope, providing 2° margin
- TPS “Knife Edge” provides uniform shadow and illumination across solar array strings

*Exceptions are PWI antennas (not shown), and deployed solar arrays

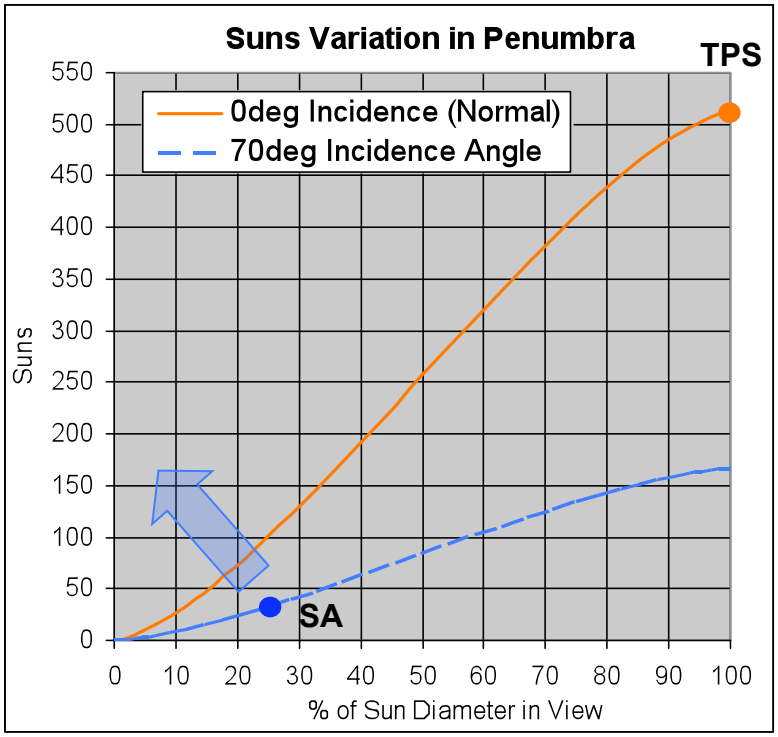
Solar Array Illumination – 9.5Rs



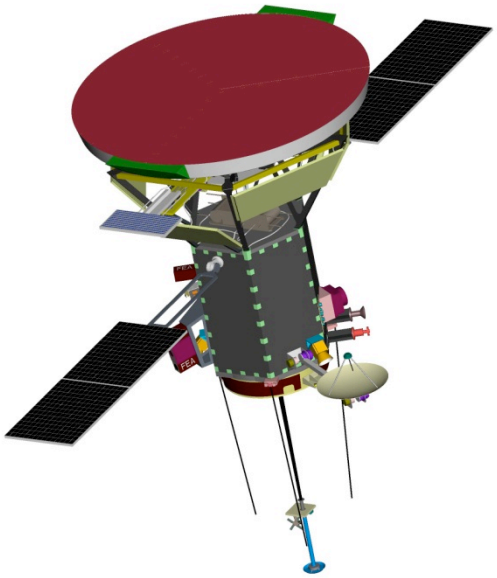




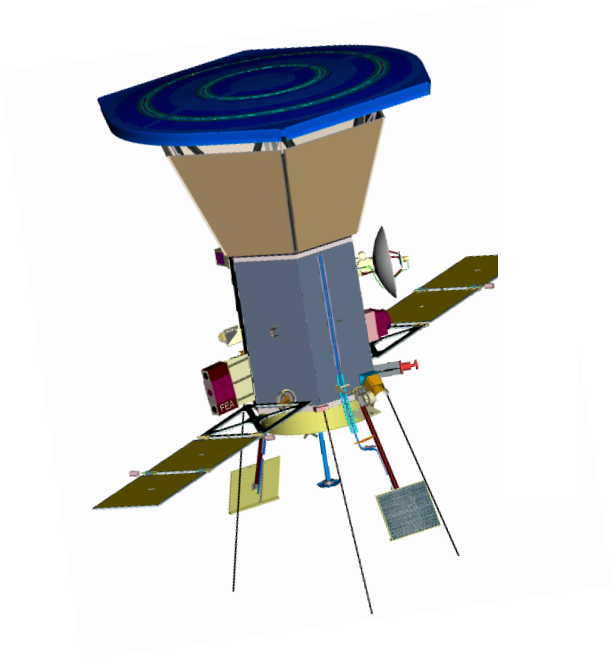
Penumbra and cosine effect reduce peak suns on solar arrays at closest approach from 512suns to ~35suns



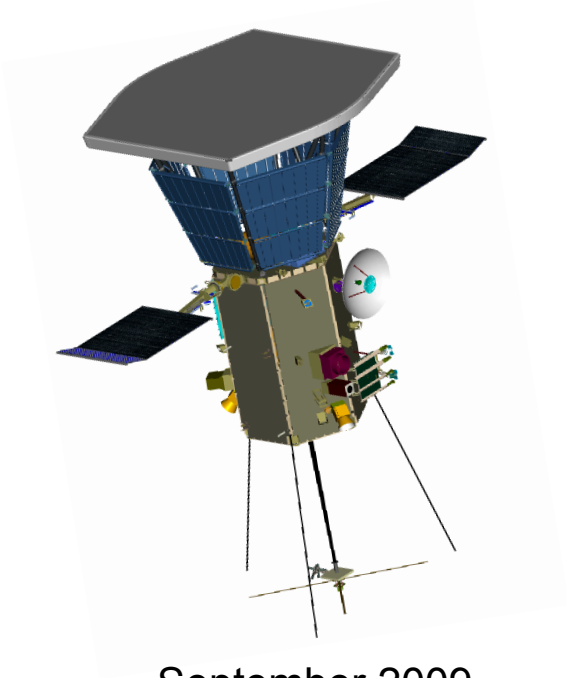
Solar Array Evolution



March 2008
Study Report concept
Sliding actively cooled secondary arrays
Passively cooled primary arrays

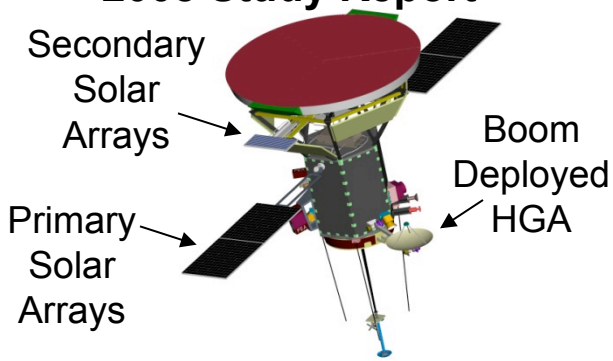
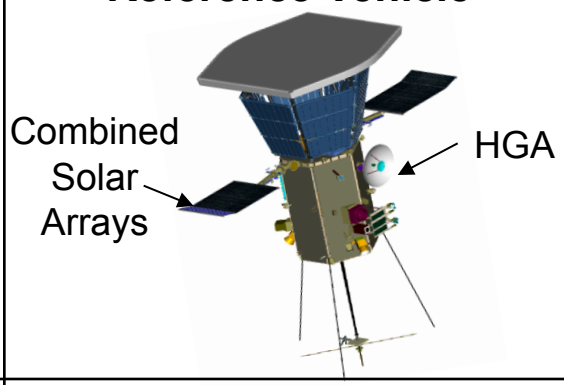


January 2009
Rotating actively cooled secondary arrays
Passively cooled primary arrays



September 2009
"Combined Array"
Combines primary and secondary array
– 2 arrays instead of 4
Entire array is actively cooled

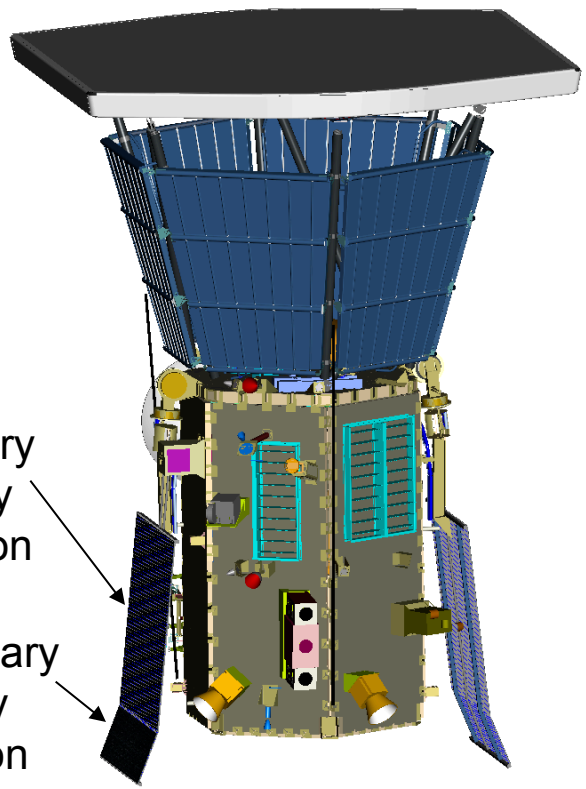
Reduced Mechanisms since 2008 Study Report Concept

	2008 Study Report	Reference Vehicle
		
# of Solar Arrays (SA)	**4 SA: 2 Secondary; 2 Primary**	**2 Combined SA**
Transition Between SA Wings Each Orbit	**Yes**	**No**
Secondary SA Mechanism	**Slide**	**Flap and Feather**
Primary SA Mechanism	**Deployment and Feather**	
HGA Mechanisms	**Single axis gimbaled HGA boom and single axis gimbaled HGA**	**Single axis gimbaled HGA**
HGA Operation	**Boom deployment & stow each orbit; HGA gimbal position – comm & stow**	**HGA gimbal position – comm & closest approach**
Number of Mechanisms	**8**	**5**

All reference vehicle mechanisms operate in standard spacecraft environment

Combined Array

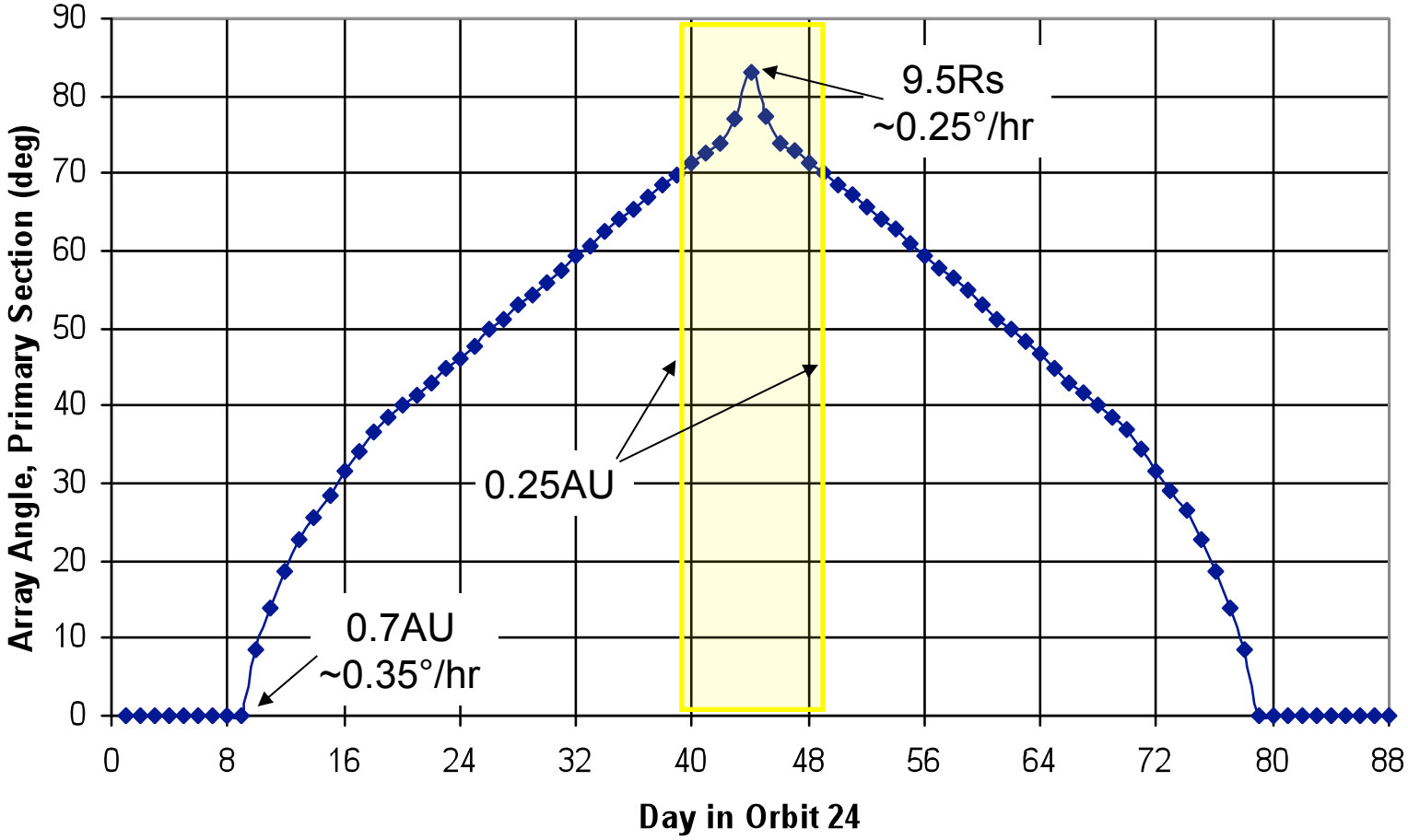
- Motivation
  - Provides increased solar load to cooling system to eliminate freezing at aphelion
  - Allows cooling fluid selection for effectiveness at high temperature



- Secondary array section
  - Tipped up 16° (fixed) relative to primary section
    - Meets 70° max incidence constraint
  - Solar cell and cooling channels designed for high solar flux environment
- Primary array section
  - Lower solar flux
  - Standard space cells with higher packing factors
  - Acts as additional radiator area; reduces radiator area req'd under TPS
- Eliminates transition between arrays during ops
- Eliminates primary arrays and mechanisms
- Maintains bus panel area for instrument packaging
- Arrays located on panels parallel to ram direction to minimize dust environment

Array Rotation – Autonomous

Maximum array rotation rate vs. solar distance is ~0.35°/hr (0.0001°/sec)



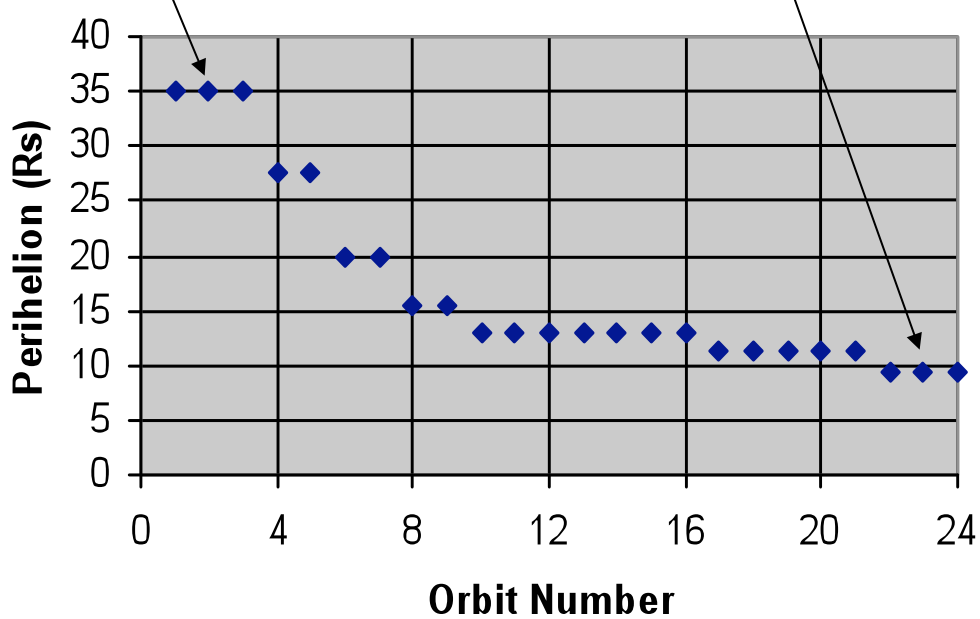
Subtract 16° to determine secondary section angle

Array Control and Walk-In Calibration

- Array position is controlled autonomously by G&C based on array angle vs. solar distance table
- Solar array positioning is calibrated in flight prior to first encounter
  - Array angle vs. array temperature vs. solar distance
  - Power vs. array angle
- Array includes temp sensors on outer strings for use in autonomous monitoring of max array temp relative to array temp limits.
- Data from 1st encounter and subsequent encounters with progressively higher heating used to further update array positioning parameters as needed

- 38 Suns
- 1.6° umbra angle
- String 1: ~20 Suns
- Thermal sensitivity to array angle: ~1/2 of that at 9Rs

- 512 Suns
- 6° umbra angle
- String 1: ~35 Suns
- Thermal sensitivity to array angle: ~2x that at 35Rs

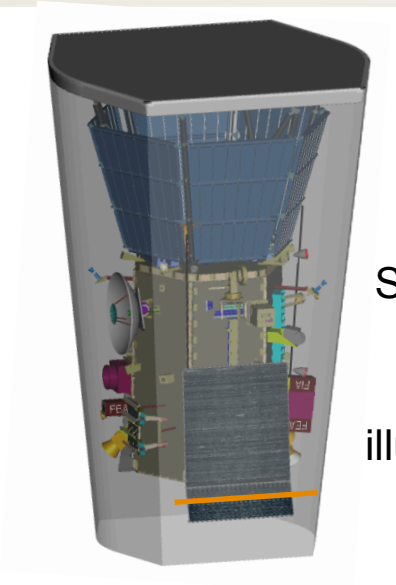


Cooling System Sizing, Margin

- Cooling system provides cooling for combined solar arrays
- Cooling system sizing provides capacity for
  - Electrical power loads with margin
  - Electrical power capability uncertainty and string out
  - Array alignment relative to TPS knife edge with margin
  - G&C control of array position with margin
  - Temperature sensor uncertainty for array temperature limit
  - Additional thermal margin and conservative thermal loads
  - Heat from TPS with margin

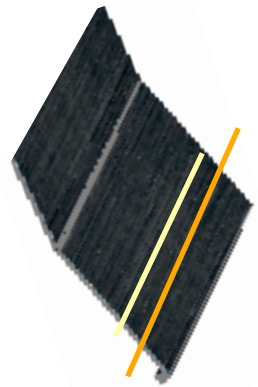
- Cooling system CBE mass and power are based on cooling system capacity defined by above.

TPS Knife Edge to Solar Array Alignment



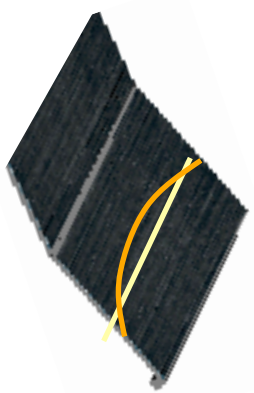
Solar array uses one string per row to enable uniform illumination along each string.

- Alignment errors of strings relative to TPS knife edge (k.e.) result in variation of sun intensity and shadow across string.
- Classes of alignment errors:



Displacement: shadow of TPS k.e. on array does not fall on expected string for given solar distance and array angle

- Mitigated by small change in array angle of incidence for given solar distance thru in-flight calibration



ex. TPS k.e. deformation

Skew/"waviness": shadow of TPS k.e. on array is not parallel to strings

- Power generated is reduced for a given thermal load
- RSS of all ground and in-flight skew estimated to be < 2.5mm
- For a given elec power, this results in <40W increase in thermal load.
- 400W thermal margin is included in cooling system thermal budget for alignment errors.

Actual TPS k.e. shadow
Expected string

Solar Array Cooling System Sizing - 9Rs

2° attitude excursion < 60 seconds

	Platen Temp (C)	Elec Load Power (W)	Thermal Load (W)	
	150	439	**5790**	**Thermal System CBE Sizing**
Additional Thermal Margin				
	137	410	5200	Actual Platen Temp +/-4°C
G&C/Autonomy Array Positioning with Margin : ~0.2° (TBR) change in incidence in both arrays in same direc relative to sun				
G&C Array Positioning: ~0.1° (TBR) both arrays same direc	117	**343***	4450	**Electrical Load Power Req'd**
Alignment Error with Margin	106	343*	4050	

Additional Power Margin : 11% (string out) At 117°C power available is 381W
Power Loads Margin : 32% (contin + margin) on 260W CBE load power = 343W elec load power req'd

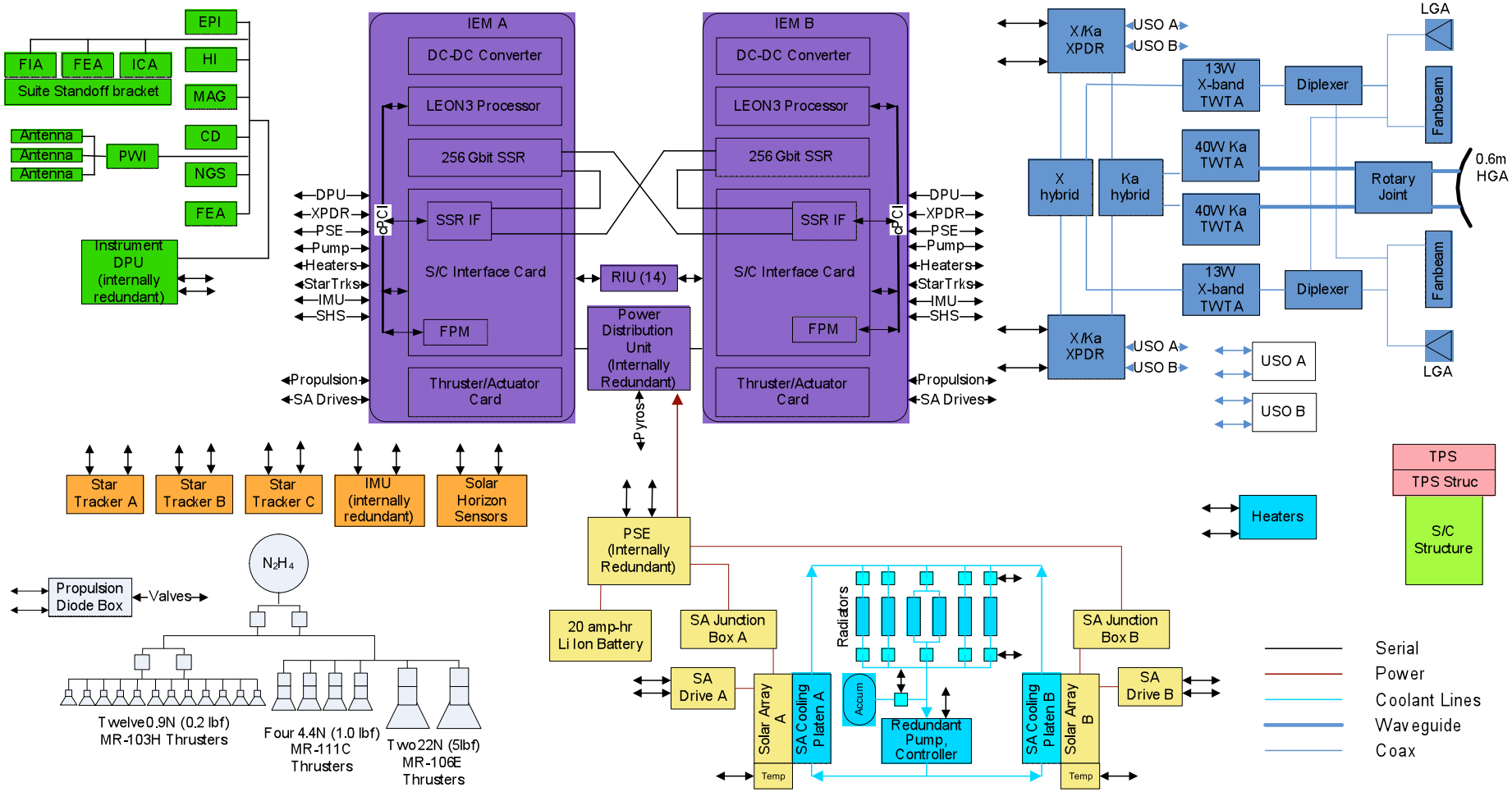
NOTE: All thermal loads include 300W NTE from TPS;** Pump power CBE 60W included (currently 40W) *at 137°C;
Thermal load uses alpha cell= 0.85; alpha OSR = 0.2; alpha gaps = 1; All values are TBR.

**TPS load currently ~360W including margin

Cooling System Low Temperature Performance

- Cooling system designed to maximize effectiveness at closest solar approach
- Cooling system is designed and operated to avoid freezing during low temperature operating phases
  - Launch transient – cooling system is launched with all fluid in accumulator; after launch arrays and radiators are heated sequentially using solar heat to a 20°C temperature prior to sequential release of fluid into arrays and radiators.
  - Launch correction maneuver transient – four radiators (TBR) remain dry until after launch correction maneuver. Allows ΔV thruster orientation in any direction while maintaining adequate solar heat into cooling system to avoid freezing
  - Aphelion for orbits 1-4 steady state – adequate margin
  - Venus eclipse transient – 11 minutes is max duration; cooling system is warm prior to transient – adequate margin

Block Diagram



Mass

Name	CBE Total (kg)	Total (kg)
Instruments Total:	47.2	51.9
Accommodation Hardware Total:	5.7	6.4
Telecommunications Total:	31.6	34.7
Guidance and Control Total:	16.1	17.4
Power Total:	57.2	64.2
Cooling System Total:	66.6	78.8
Thermal Protection System Total:	71.3	84.8
Thermal Total:	13.0	14.9
Avionics Total:	27.9	31.4
Propulsion Total:	24.5	26.4
Mechanical Total:	39.6	45.6
Harness Total:	21.5	25.7
Spacecraft Dry Mass	422.3	482.0
Propellant Total:		68.2
Spacecraft Wet Mass		550.2
Launch Mass		610.0
Allocated Margin	59.8	14.2%
Unallocated Margin	59.8	12.4%
Total Mass Margin + Contingency		28.3%

Power Rollup (1 of 2)

	Post-Separation Total (W)	Heat Rad 1 Total (W)	Heat Rad 6 Total (W)	Recharge Post Sep, Rad 1,6 Total (W)	Launch Error Correction Total (W)	Recharge Post LC Total (W)
Instruments CBE Power:	0.0	0.0	0.0	0.0	0.0	0.0
Accommodation CBE Power:	0.0	0.0	0.0	0.0	0.0	0.0
Telecommunications Total:	45.2	45.2	45.2	12.2	45.2	45.2
Guidance and Control Total:	37.4	37.3	37.3	37.3	37.3	37.3
Power Total:	16.5	16.0	16.0	16.0	16.0	16.0
Thermal Control Total:	0.0	60.0	60.0	69.0	63.0	69.0
Avionics Total:	34.0	34.0	34.0	34.0	34.0	34.0
Propulsion Total:	26.4	26.4	26.4	26.4	92.5	26.4
Spacecraft Totals						
Power CBE w/o Harness Loss	159.5	218.9	218.9	194.9	288.0	227.9
Heater Power Req'd	0.0	39.8	39.8	41.6	0.0	21.0
Harness Loss	2.4	3.9	3.9	3.5	4.3	3.7
Power CBE	161.9	262.6	262.6	240.0	292.3	252.6
Battery Charge				22.8		5.0
Max s/c - sun distance (AU)	1.02	1.02	1.02	1.02	1.01	1.01
S/A Temp Capability Calc (C)		105	105	70	70	70
Load Capability	0.0	245.1	245.1	335.0	343.1	343.1
Load with 30% Total Margin	210.5	341.3	341.3	334.9	380.0	333.4
Power Reserves				72.2		85.5
Margin				30.08%		33.83%
Duration	1.00	0.76	0.69		0.73	

Power Rollup (2 of 2)

	Inst CO, HGA SSR Playbacks	Science: 35Rs - 0.25AU	Science: 9.5Rs to <35Rs	Cruise/DL
	Total (W)	Total (W)	Total (W)	Total (W)
Instruments CBE Power:	57.2	57.2	57.2	0.0
Accommodation CBE Power:	0.3	0.3	0.3	0.0
Telecommunications Total:	95.2	45.2	10.4	95.2
Guidance and Control Total:	44.7	45.0	45.0	37.3
Power Total:	16.0	16.0	16.0	16.0
Thermal Control Total:	66.0	66.0	66.0	69.0
Avionics Total:	34.0	34.0	34.0	34.0
Propulsion Total:	26.4	26.8	26.8	26.4
Spacecraft Totals				
Power CBE w/o Harness Loss	339.7	290.4	255.6	277.9
Heater Power Req'd	0.0	0.0	0.0	0.0
Harness Loss	5.1	4.4	3.8	4.2
Power CBE	344.8	294.8	259.5	282.1
Battery Charge				
Max s/c - sun distance (AU)	0.7			0.76
S/A Temp Capability Calc (C)	106	130	137	90
Load Capability	550.7	390.0	343.0	494.8
Load with 30% Total Margin	448.2	383.2	337.3	366.7
Power Reserves	205.9	95.2	83.5	212.7
Margin	59.70%	32.29%	32.19%	75.41%

Solar Probe Plus Spacecraft – Part 2

MK Lockwood
Spacecraft System Engineer

The Johns Hopkins University
APPLIED PHYSICS LABORATORY

Outline

Part 1

- Reference vehicle intro
- Trades
- TPS function
- Solar array evolution
- Mechanism simplification
- Combined solar array
- Cooling system sizing and margin
- Reference vehicle
  - Block diagram
  - Mass, power

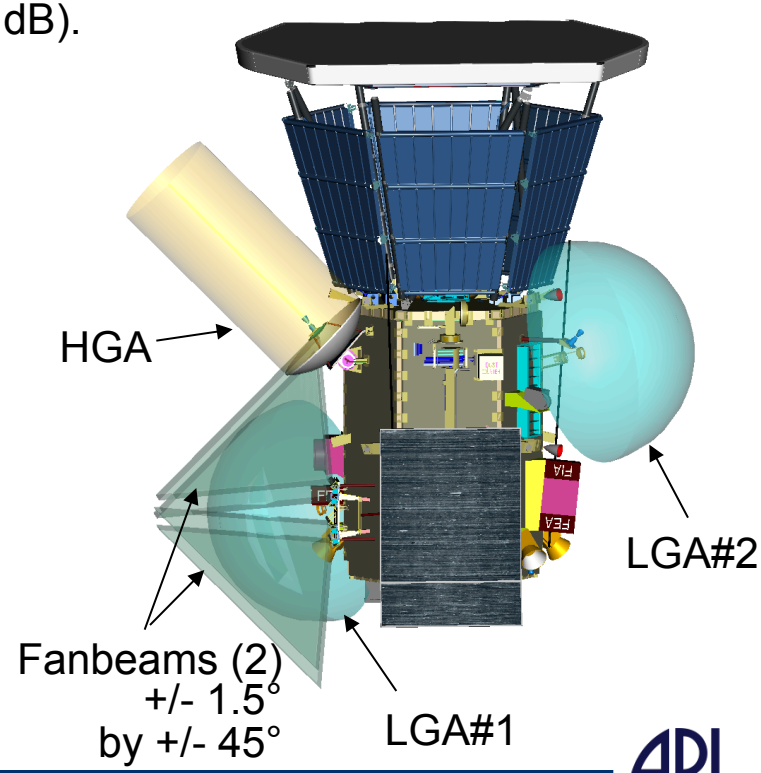
Part 2

- HGA, Communication
- Science data downlink
- S/C thermal results
- G&C, Cp-cg alignment
- Thermal safe mode
- Propulsion
- V&V solar encounter
- Margin summary
- TRL summary

Communication

- 40W Ka-band TWTA, 13W RF X-band TWTA
- 0.6m aperture HGA dedicated to Ka-band – primary science downlink
- Two X-band Fanbeams aligned with HGA for TT&C
- Separate Ka and X-band:
  - Reduces umbra/TPS size requirements by eliminating extended HGA feed.
  - Eliminates dichroic reflector in Ka-band path (0.5 dB).
- X/Ka-band transponder – TRL 4 (NASA HQ funding to TRL 5-6 over next 12 months); USO – TRL 6; all other communication components are flight proven

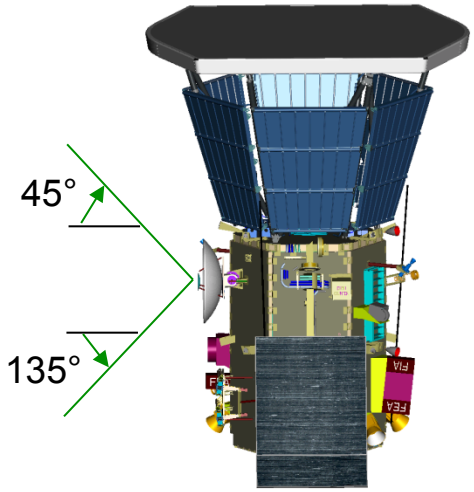
Earth Range (AU)	0.6m HGA Ka-band 40W DSN 34m (kbps)	Fanbeam X-band 13W DSN 34m d/l, u/l (kbps)	LGA X-band DSN 70m d/l, u/l (bps)
0.5	555	2.5, 4	167, 389
1.0	138	630, 1	42, 97
1.8	42	0.195, 0.320	6, 30

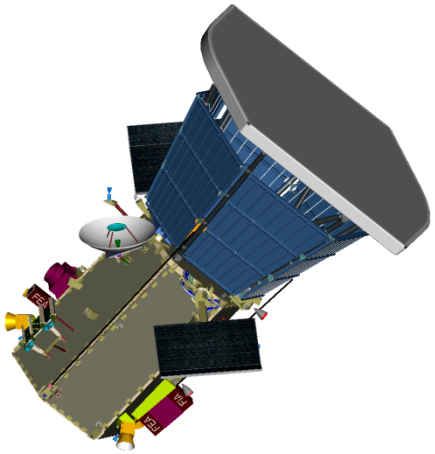


Body Mounted Single Axis Gimbal HGA

Mode 1:

- **SPE: 45° to 135°**
- **s/c sun distance: 0.28 to 0.76AU**
  - **TPS to sun**
- **Roll about s/c z axis for HGA to Earth**





SA 45° "feather" angle shown

Mode 2:

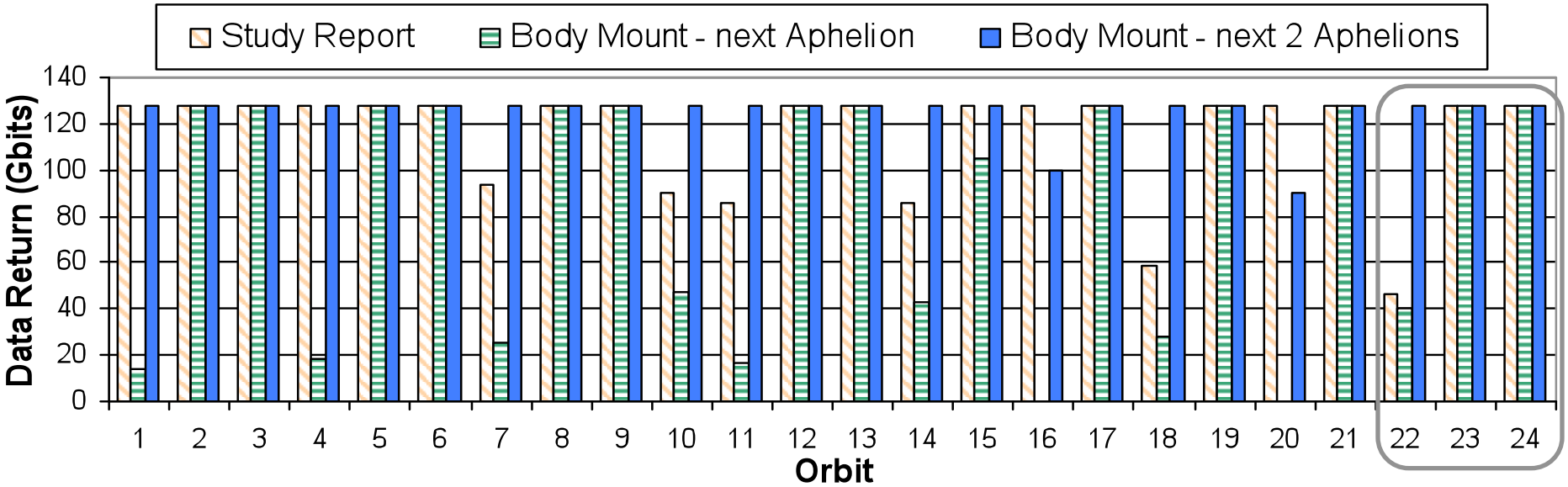
- **SPE: 0° to 45°**
- **s/c sun distance: 0.7 to 0.76AU**
- **Slew up to 45° off sun**

- Body mounted HGA
  - Returns 128Gbit/orbit data for all but 2 orbits within the same or following orbit
- Motivation
  - Eliminate deployment of HGA in and out of umbra compared to boom mounted HGA
  - Reduce temperatures on HGA
- Slew results in packaging constraints due to thermal; however thermal design for slew:
  - Provides ability to increase efficiency of ΔV maneuvers at s/c sun distances >0.7AU.
  - Supports solar heating of radiators after launch

Science Downlink

- 128Gbit data collected per orbit
  - Includes science data, margin and housekeeping
- 256Gbit SSR
  - Enables storage of 2 orbits data
- SSR cross-strapped to each IEM
- Science downlink rate: 138kb/s at 1AU
  - Data rate adjusted, by day, for s/c Earth distance
- Downlink for 8 hours per day
  - 10 hour DSN contact
- No downlink for SEP<1°

Science Data Downlink



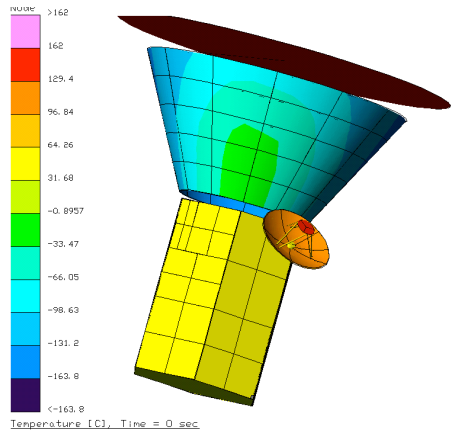
- **For 3 orbits with altitude < 10Rs and for 19 of 21 early orbits:**
  - **Downlink of 128Gbits/orbit possible within same or following orbit**
  - Provides partial DL from each orbit prior to data collection from following orbit
    - Data latency for retained data is allowed to increase to ~6 months
- For two of 24 orbits (#16 and #20):
  - Orbit 16: 100Gbits DL after #17 encounter; Orbit 20: 90Gbits DL after #21 encounter
- DSN contacts fit within number included in 2008 Study Report

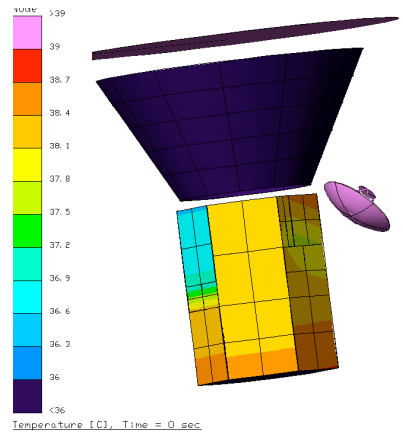
S/C Thermal Design

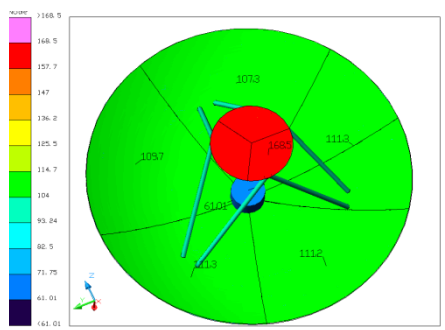
- Detailed integrated thermal model of SPP developed
- Heater power, s/c louvers and radiators sized to maintain s/c between 10°C and 40°C.
- S/C thermal design cases
  - Hot cases - Radiator area ~0.9m2
    - TPS hot case – 9.5Rs, TPS to sun, 50W from TPS, s/c power load 343W
    - Slew hot case – 0.7AU, TPS 45° slew from sun, no heat from TPS
  - Cold case – 1.02AU, TPS to sun, no heat from TPS
    - Heater power req'd to maintain 270W min s/c power load
    - Louvers: one 20 blade louver, one 10 blade louver
- Battery temperature maintained between 10°C and 30°C
  - During battery discharge, phase change material used to maintain battery temperature
    - Largest DoD occurs during launch, sizes paraffin to 1.5kg
    - Constrains time between battery discharge events of this DoD to 6-7 hours (less than time required to recharge battery)
  - When battery is not discharging, 6W heater power maintains battery temperature.

S/C Thermal Design - Slew

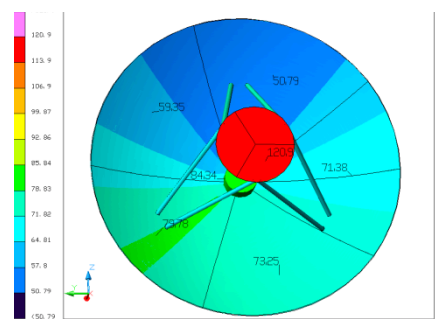
- S/C design accommodates 45° slew maneuver for s/c-sun distance ≥0.7AU
  - Components with low/no power consumption during slew are packaged on s/c panels illuminated during slew
  - Heat load on cooling system radiators due to solar illumination during slew is ~1100W, within system capacity at 0.7 – 0.76AU.
  - Max HGA temperature of 111°C is well below the M55J/RS-3 temperature limit of 180°C, adhesive temp limit is 190°C with a post-cure.
  - Star tracker apertures located to avoid illumination
  - Instrument apertures to be considered







45° from S/C: The sun facing portion of the dish has a maximum temperature of 111°C



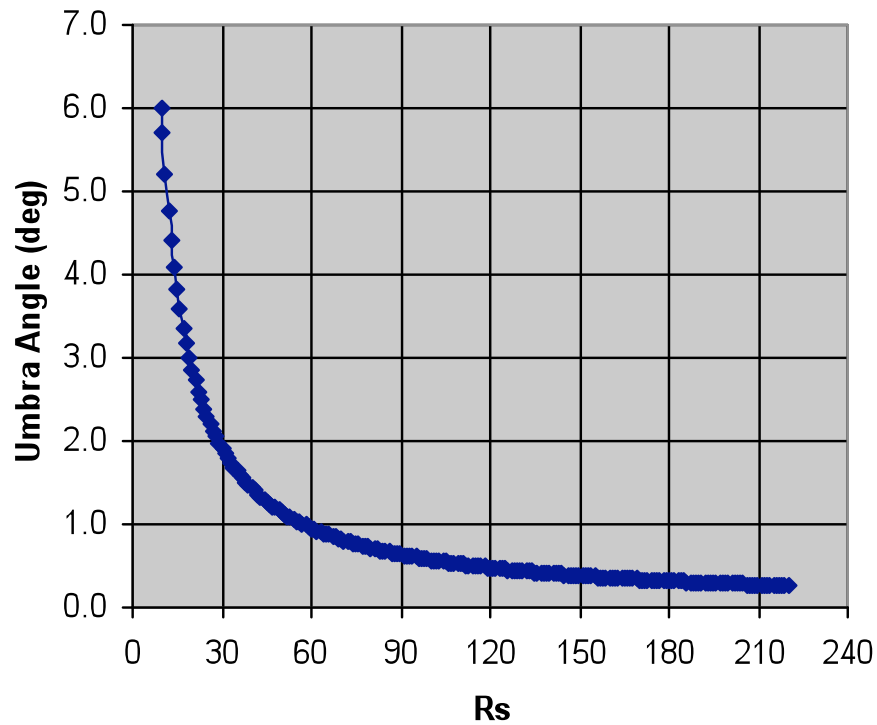
0° from S/C: The sun facing portion of the dish has a max ΔT of 29°C due to the shadowing affects by the HGA

G&C

- Working requirements: attitude control
  - s/c control to within 0.05° (TBR) for s/c sun distance < 0.25 AU (TBR) and during science data downlink
    - Array angle maintained within 0.1° (TBR) relative to sun at encounter (req'd for s/c attitude changes parallel to array rotation)
    - HGA requires total pointing error to be within 0.2° (TBR)
    - s/c packaged inside 6° umbra plus 2° umbra margin at 9.5Rs
  - s/c control to within (TBD) degrees for s/c sun distance > 0.25AU and when science data downlink is not occurring
  - Array angle controlled autonomously
    - Defined as function of s/c sun distance
    - Calibrated in-flight
- G&C components
  - IMU, star tracker, MR-103H – flight proven
    - Star tracker performance assessment in dust environment in progress – results indicate background light due to dust is within qualified environments
    - MR-103H delta-qualification cycle testing planned
  - Thermal safe mode sensors, algorithm – further definition by April 2010

G&C Environments, cg-cp Offset

- Solar pressure is primary influence on s/c
- Torques due to dust, solar wind, magnetic fields are significantly smaller
- Constraint on center of pressure (cp) to radial location of center of gravity (cg) is required to minimize propellant use and thruster firings
  - 2cm (TBR) cp-cg offset working requirement for Reference Vehicle
- To reduce cp-cg offset in flight, if needed
  - Umbra angle allows off-sun pointing up to a few degrees through most of orbit
    - Sun pointing maintained through critical science phase
  - 1-1.5° (TBR) off sun pointing provides correction for ~2cm cp-cg offset

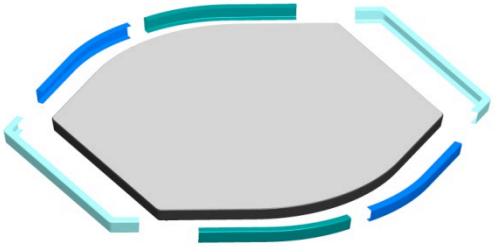


Radial Cp – Cg Alignment

- Working Requirement: TPS center of pressure (cp) within 2cm (TBR) of vehicle radial location of center of gravity (cg)
- Adjustment of cp-cg
  - Up to ~6mos after I&T start, late in TPS manufacturing process:
    - TPS "closeout" approach used to adjust center of pressure relative to TPS-truss mounting locations
  - After TPS is delivered and vehicle mass properties are measured, cp adjustment options
    - TPS translated radially relative to truss
      - as-delivered TPS larger than req'd to maintain umbra
      - provides ~2.5cm* (TBR) shift in cp
    - TPS shimmed at TPS-truss mounting locations to provide TPS cant angle; cant angle, TPS optical properties define cp shift
- Approaches to be traded in Phase A

*2.5cm cp shift capability includes significant margin; ex. an instrument/component would have to grow by ~30kg to require a cp shift of 2.5cm.

Exploded view of TPS closeouts



Jitter/Stability

Table 3.12-1. Overall system pointing budgets*.

Payload	Pointing Requirements (per axis)		
	Control degrees, 3σ	Knowledge degrees, 3σ	Jitter degrees, 3σ
Communication (HGA)	²0.2 *x*, *z*	N/A	N/A
In Situ Instruments <0.3 AU	²1 *x*, *y*, *z*	²0.3 *x*, *y*, *z*	²0.3 *x*, *y*, *z* over 10 s
Magnetometer <0.3 AU	²1 *x*, *y*, *z*	²1 *x*, *y*, *z*	²1 *x*, *y*, *z* over 0.05 s
Plasma Wave Sensor <0.3 AU	²1 *x*, *y*, *z*	²1 *x*, *y*, *z*	N/A
Hemispheric Imager <20 R_S	²1.0 *y*, *z*	²1.0 *y*, *z*	²0.03 *y*, *z* over 1 s

- ACS Thrusters
  - Initial G&C analysis completed using STEREO s/c model in SPP environments to assess feasibility of ACS thrusters; results show thrusters are expected to meet instrument pointing requirements as defined in 2008 Study
  - Thruster firing time tags to be downlinked for use in science data analysis
- Cooling pump frequency can be tuned to avoid frequencies near s/c structure natural frequency
- Array rotation rates vs. solar distance during science phase are ~0.25°/hr
- Phase A, B analyses/design for higher fidelity results with instrument requirements

*Solar Probe+ Mission Engineering Study Report, March 10, 2008

Thermal Safe Mode

- Working requirements
  - Maintain s/c within TPS umbra (exceptions: comm slews, Delta-V maneuvers, radiator heating sequence)
  - If critical fault* or unexpected processor reset, resume control within (TBD) sec
  - No failure in fault protection can put s/c in unsafe configuration or state
- Thermal safe mode:
  - Temporary safe mode enacted during a critical fault or unexpected processor reset
    - 1-2 minute (TBR) max duration
    - Maintain TPS pointed at sun
    - Initiate change in array position to 70° (TBR) – thermally safe angle
    - Non-critical loads are turned off
    - s/c operates on battery power as needed
  - Autonomous return to nominal mode after processor reset and attitude data is available
- Components:
  - s/c packaged to provide 2° s/c attitude margin at closest approach (9.5Rs)
  - Sun sensors positioned such that sensor sees sun before any other s/c component
    - Sensors placed to detect 1°(TBR) s/c offpoint of TPS from sun
  - Thrusters
  - FPGA-based or small processor-based controller – s/c attitude, array position
  - Control algorithm to thrust in opposite direction of sun detection

* See Fault Management briefing (presented tomorrow)

Propulsion

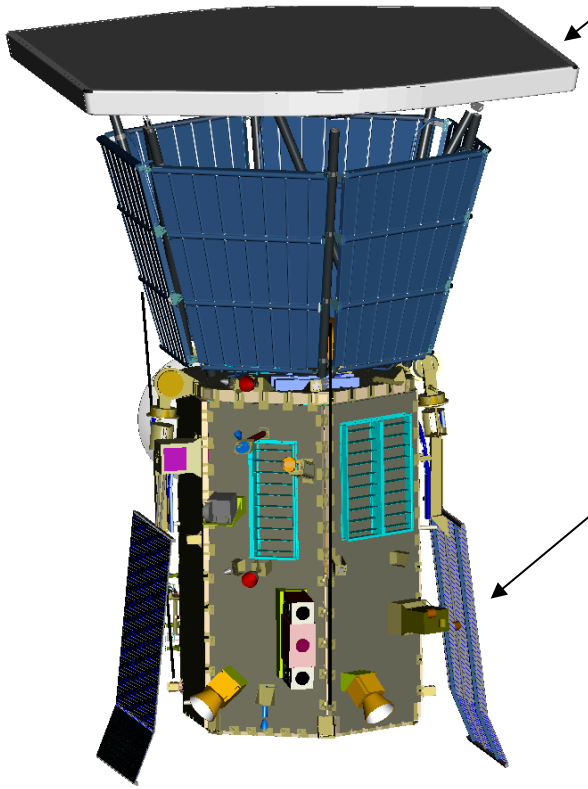
- Attitude control
  - MR-103H thrusters are utilized for attitude control in Reference Vehicle
    - Flight heritage from New Horizons and Cassini
    - delta qualification for thruster cycles is planned
  - MIT thruster is carried as back up until instruments defined
    - MR-103M uses same thruster as New Horizons and Cassini MR-103H thrusters, but different valve to provide minimum impulse
    - MIT was flight qualified, but if selected for SPP, a valve re-qualification and REA proto-qualification is planned
    - Assessment of MIT test results and status will occur by April 2010
  - MR-103H vs. MIT vs. wheels trade will occur by April 2010.
  - Thruster cycles vs. thruster layout and pointing req's will occur by April 2010
- Propulsion Tank
  - Custom spherical tank based on ATK 80259-1 elastomeric diaphragm tank is included in Reference Vehicle
    - 79kg propellant capacity; 375-100 psi operating pressure range
- All other propulsion system components are TRL 8-9

Solar Encounter V&V

- Solar encounter V&V utilizes a series of tests, high fidelity analyses, and simulations to provide end-to-end V&V
  - V&V of flight s/c and cooling system is accomplished with full scale thermal vacuum test of flight vehicle using TPS simulator and solar array simulators
  - However, a s/c-sized thermal vacuum chamber that can simulate the 512 sun environment at SPP closest solar approach does not exist. Flight h/w tests that cannot be completed at 512 suns environment:
    - Full scale TPS (thermal, optical, thermo-mechanical performance)
    - Full scale solar array thermal performance from cell surface to cell-platen interface at array angle of incidence in TPS penumbra
    - Full scale solar array electrical power performance at array angle of incidence in TPS penumbra
    - Cp-cg alignment and knife edge shadow shape due to thermomechanical deformation
    - Flight control and effect on solar array heating and electrical power
  - V&V for closest solar approach
    - S/C, prototype, subsystem, and component tests
    - High fidelity subsystem and integrated vehicle-level analyses
    - Monte Carlo simulations
    - Software and hardware-in-the-loop simulations

S/C Thermal Vacuum Test

Thermal Vacuum Configuration



TPS Simulator
Flight-like TPS bottom pan temperature simulated

S/A Simulator
Flight-like cooling platen with heaters simulating solar cell heat input

- To simulate the 512 suns environment in Thermal Vac, a TPS Simulator and Solar Array Simulators replace the flight TPS and Solar Arrays.
- TPS Simulator simulates the TPS bottom pan temperature, providing the flight-like heat input to the s/c and cooling system radiators
- Solar Array Simulators provide the flight-like heat input to cooling system platens
- Enables a flight-like s/c and cooling system test

Solar Encounter V&V Thermal Performance

TPS	Spacecraft	Cooling System	Power System/Arrays	G&C
ORNL-thermal performance, foam, coating coupons		Heat transport, TV - breadboard	Thermal cycle, panel, coupon	
JSC radiant heat thermal conductivity, scaled shield		Thermal perf, heat load cycling, temp cycl, TV - Prototype	Thermal exposure-long term-CICs, high temp	
Coating optical properties		Dust impact testing	Thermal expos-long term-bypass diodes, high temp & current	
Coating adhesion				
Life cycle testing			Thermal-partial strings, max intens & temp, norm-max incidence	
Radiation, EUV testing				
Dust impact testing			Dust impact testing	
TPS high fid thermal model		Cooling System-High Fid Sim		
TPS thermomechanical model		Cooling Sys Hydraulic Sim		
Integrated TPS - S/C thermal model		Integrated S/A Thermal Environ, Cooling System, Elec Power		
Environments, Dust Modeling				
G&C Sim-Monte Carlo, S/C Control, S/A Control				
S/A to TPS knife edge alignment				
	S/C TV-simulated TPS, simulated S/A heat flux		TV-panels,cycles op limits, surv temp plateaus	
Testbeds - Software and Hardware-in-the-Loop Simulations				
Comprehensive Performance Tests, Autonomy Tests, Operational Tests, Mission Simulations				

Prototype, Dev Test	Flight H/W Tests	Analyses	Flight Testbed

Solar Encounter V&V
Electrical Power, Control, Mechanical

	TPS	Spacecraft	Cooling System	Power System/Arrays	G&C
Electrical Power Performance			Pump - performance	Flash-cells,max intens,min-max temp,pre-post irrad	
			TV, delta p - Breadboard	CICs-incidence angle, 1-sun	
			TV, delta p - Prototype	UV-CICs, high temp, vac, pre-post I-V solar sim	
				Flash-string, incidence, max intens, min-max temp	
				Dust impact testing	
			Cooling Sys Hydraulic Sim	Dust environ, impact model	
				Elec Power-high fidelity sim	
	TPS thermomechanical model (k.e. distortion)		Integrated S/A Thermal Environ, Cooling System, Elec Power		
	G&C Sim-Monte Carlo, S/C Control, S/A Control				
	S/A to TPS knife edge alignment				
				Flash, max intens, normal	
				Dark I-V survival current	
G&C Performance	Coating optical properties	ACS thruster cycle test			SHS Test
					SHS Model
		Plume analysis			Star Track Perf
		S/C Model	Cooling system therm model		Environ Models
	TPS thermomechanical model		Fluid momentum model	Power Performance Model	G&C Algorithms
	G&C Sim-Monte Carlo, S/C Control, S/A Control				
	TPS flatness	Mass Properties			
		Cp-cg alignment			
Mech	TPS/TSA Bearing Life Test	Mass Properties	Jitter charac-pump, flex lines		
	TPS CTE and fatigue tests	Cp-cg, ke Model	Flex lines-mechanical cycle		
	TPS thermomechanical model				
		S/C Jitter Analysis			
Testbeds - Software and Hardware-in-the-Loop Simulations					
Comprehensive Performance Tests, Autonomy Tests, Operational Tests, Mission Simulations					

Prototype, Dev Test	Flight H/W Tests	Analyses	Flight Testbed

Margins Summary

	Margin
Mass	28%
Power	30%
Uplink	>6dB
Downlink	2dB science, 3dB emergency
Data Rate	30%
Data Downlink Time	25%
Critical CPU Time	60%
Delta-V	8%
Tank Capacity (above margined Delta-V)	16%
Packaging within Umbra	2°

Technology Development Summary

	TRL	Comments
TPS[i]	4	Builds on NASA sponsored tech dev at JHU/APL
Solar Array Power[ii]	4	Combines space heritage & terrestrial concentrator cell technology
Cooling System	4	SPP components leverage space heritage
Thermal Safe System	4	Initial concept in Ref Veh, further definition in Phase A
X/Ka-band Transponder[iii]	4	NASA HQ funding to TRL 5-6 over next 12 months
LEON3[iv]	5	Fully qualified Aeroflex Standard Cell RadHard ASIC implementation derived from flight proven IP core heritage design
USO	6	Flight qualified, but not flown
Mechanical[v] **–Flex Lines**	6	Flexible SS hoses with flight heritage delivered by several vendors; additional qualification planned for articulation req's

i TPS truss structure assembly is TRL 6
ii PSE, SAJB, battery are TRL 7
iii Communication, except transponder and USO, are TRL 8-9
iv Avionics, except LEON3, are TRL 7-8
v Mechanisms, except flex lines, are TRL 8; s/c structure is TRL 8;
Propulsion: MIT backup option is TRL 6; tank is TRL 7; all other propulsion components are TRL 8-9

Backup

Design Drivers, Features (1 of 4)

Design Driver	Feature	Benefit
Solar environment - closest approach	TPS carbon foam and alumina coating	Provides umbra; allows s/c to operate in typical thermal environment; meets science mass-loss reqs
	TPS knife edge	Provides penumbra with uniform illumination along a solar array string
	Flapping rotation of solar array	Reduces local heat flux on arrays - controls array incidence angle to sun and extent of array in penumbra.
	Solar cell system leverages space, terrestrial systems	Enables higher TRL for solar cells
	Active cooling system	Enables rejection of heat absorbed by solar arrays.
	Body mounted single axis gimbal HGA	Eliminates deployment/stow of HGA into and out of sun.
	Thermal safe mode for critical fault	Protects s/c if critical fault or unplanned processor reset occurs during closest approach to sun
Solar environment variability: 1-512 suns	Combined array	Provides radiator area at perihelion and heat to cooling system at aphelion. Sized to meet power requirements.
	Cooling system coolant stored in accumulator during launch; released after solar heating of arrays, radiators	Provides low temperature protection for cooling system during launch, launch correction. Provides cooling capacity required at closest approach.

Design Drivers, Features (2 of 4)

Design Driver	Feature	Benefit
Mass	Sandwich construction TPS.	Provides thermal protection, meets launch load req's, minimizes mass compared to alternative TPS designs
	Minimize power loads	Reduces size of actively cooled solar arrays; Enables single panel per wing; Reduces cooling system sizing
	Replace wheels with ACS thrusters	Reduces s/c volume, reduces TPS and s/c structure size and mass. Eliminates wheel mass, min propellant penalty.
	S/C packaged in hex prism	Smaller TPS compared to conic-hex configuration
	Mini-channel SA platen, tube radiators, water working fluid	Minimizes cooling system size and mass
	Solar heats radiators after launch prior to fluid release.	Minimizes mass for launch-transient cooling system freeze protection.
	Aluminum core, composite face sheet s/c structure	Reduces s/c structure mass

Design Drivers, Features (3 of 4)

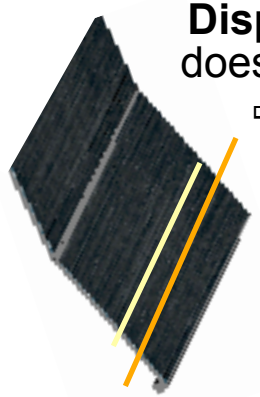
Design Driver	Feature	Benefit
Power	Separate science and DL phases; science DL < 0.76AU	Enables 100% science instrument duty cycle during science phase, uncoupled ops; reduces array sizing
	Water cooling system fluid	Maximum heat removal with minimum pump power
	Replace wheels with thrusters	Reduces power, array and cooling system size and mass.
	LEON3 processor	Provides processor throughput with less power, cost
	Write to both SSRs w/ backup IEM in standby	Provides redundant recorded data with minimal power

Design Drivers, Features (4 of 4)

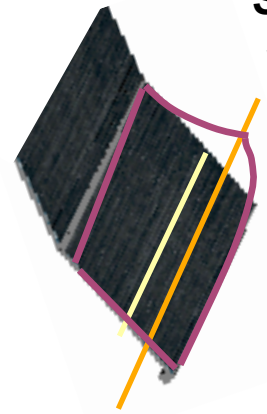
Design Driver	Feature	Benefit
Science Instruments	STDT 2008 Science Instruments Accommodation Req's met	Provides placeholder for use in Reference Vehicle; to be updated following instrument selection.
128Gbit data* for 3 orbits<10Rs	Redundant 256Gbit SSR cross-strapped to redundant IEMs	Enables return of 128Gbit data during the following 1-2 aphelions, including for orbits with solar conjunction. Increases data return for early orbits.
	File system	Simplifies mission operations; enables use of CFDP; simplifies flight software development and ground processing
Dust	Solar arrays placed parallel to ram	Reduces solar array dust environment
	Dust impact modeling	Solar array platen, radiator tubes assessed and updated as req'd to minimize probability of penetration; TPS and solar cell damage currently assessed to be well within design margins
	s/c blanket standoff	Protects s/c
	Dust illumination modeling	SPP star tracker environment currently assessed to be within qualified environments
S/C Charging	s/c charging model and effects refinement	Assess effects of charging and possible mitigation

*includes margins, s/c housekeeping

TPS Knife Edge (K.E.) to Solar Array Alignment

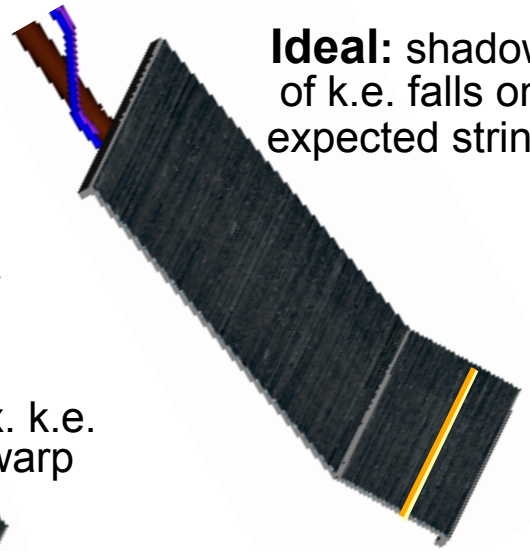


Displacement: shadow of k.e. does not fall on string expected
⇨Mitigated with in-flight calibration



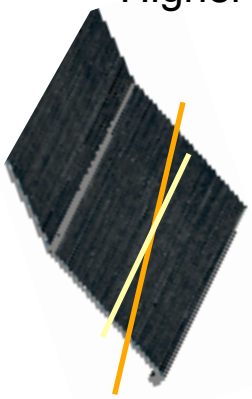
Skew-Angle: part of a string is at a different incidence angle
⇨Higher sensitivity

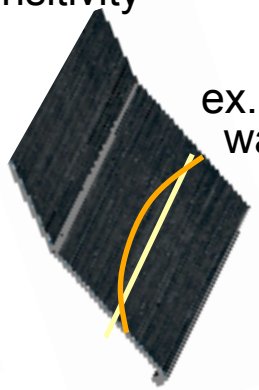
ex. platen warp



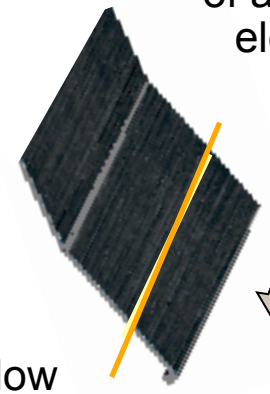
Ideal: shadow of k.e. falls on expected string

Skew-Radial: part of a string extends farther into penumbra
⇨Higher sensitivity

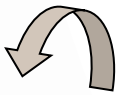




ex. k.e. warp



Skew-Elevation: part of a string is at a higher elevation, relative to TPS knife edge
⇨Low sensitivity



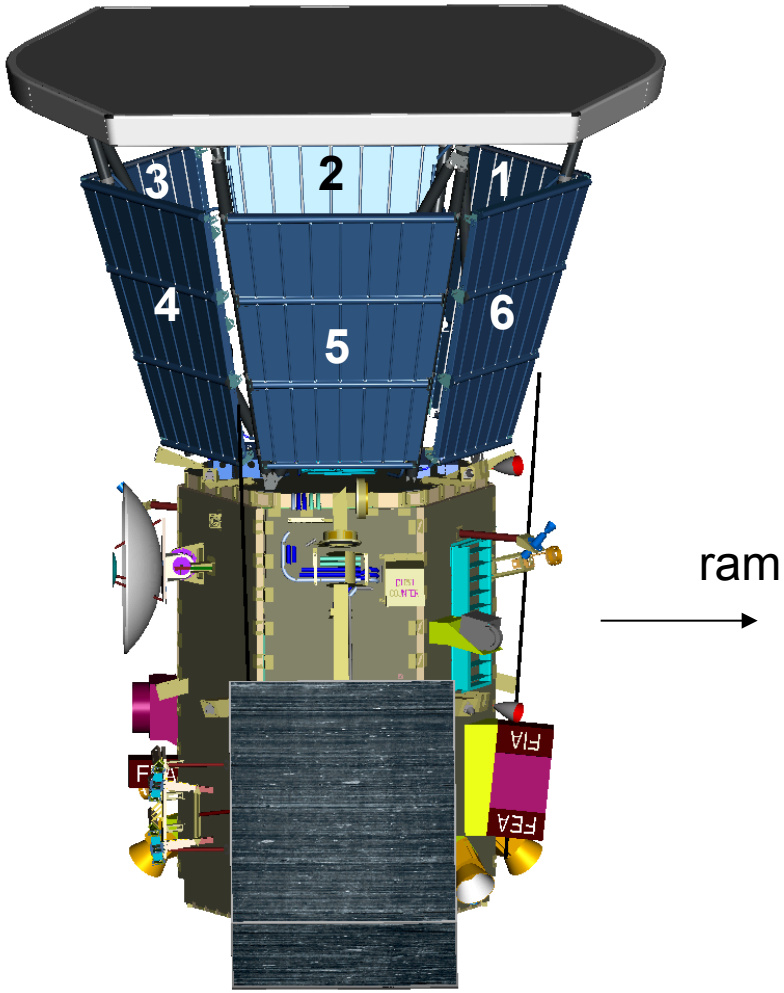
ex. array feather rotation

Actual TPS k.e. shadow
Expected string

Power Rollup
Heat Rad 2, 5; Recharge; Maneuvers

	Heat Rad 2	Recharge Rad 2	Maneuver - 1,6,2 rads	Heat Rad 5	Recharge Post Rad 5	Heat Rad 3, 4
	Total (W)	Total (W)	Total (W)	Total (W)	Total (W)	Total (W)
Instruments CBE Power:	0.0	0.0	0.0	0.0	0.0	0.0
Accommodation CBE Power:	0.0	0.0	0.0	0.0	0.0	0.0
Telecommunications Total:	45.2	45.2	45.2	45.2	45.2	45.2
Guidance and Control Total:	37.3	37.3	37.3	37.3	37.3	37.3
Power Total:	16.0	16.0	16.0	16.0	16.0	16.0
Thermal Control Total:	60.0	66.0	69.0	60.0	69.0	66.0
Avionics Total:	34.0	34.0	34.0	34.0	34.0	34.0
Propulsion Total:	26.4	26.4	61.2	26.4	26.4	26.4
Spacecraft Totals						
Power CBE w/o Harness Loss	218.9	224.9	262.8	218.9	227.9	224.9
Thermal From TPS	0.0	0.0	0.0	0.0	0.0	0.0
PSE Heat Dissipated	19.9	41.8	34.7	23.5	34.3	34.2
Thermal Power Dissipated	225.7	253.5	284.2	229.2	249.1	245.9
Hot Case Thermal Dissip w/30%	293.4	329.6	369.5	298.0	323.8	319.7
Heater Power Req'd	44.3	16.5	0.0	40.8	20.9	24.1
Harness Loss	3.9	3.6	3.9	3.9	3.7	3.7
Power CBE	267.2	245.0	266.7	263.6	252.6	252.7
Battery Charge		20.0			15.0	
Max s/c - sun distance (AU)	0.9	0.9	0.9	0.83	0.83	0.83
S/A Temp Capability Calc (C)	105	89	89	106	94	94
Load Capability	199.5	418.2	418.2	234.7	483.7	342.1
Load with 30% Total Margin	347.4	338.5	346.7	342.7	343.4	328.5
Power Reserves		153.2	151.5		216.2	
Margin		62.53%	56.80%		85.57%	
Duration	0.84		0.80	0.84		0.84

Radiator Numbering



Solar Probe Plus
Mission Concept Review

Thermal Protection System

Douglas Mehoke

The Johns Hopkins University
APPLIED PHYSICS LABORATORY

Outline

- **Requirements**
  - TPS Study Goals
  - SPP TPS Working Requirements
- **Options Considered**
  - Historical Design Evolution
  - Mechanical Trades
  - Thermal Trades
  - Material Studies
- **Reference Design and Feasibility**
  - Peer Review Summary
- **Go-forward Plan**
  - Material Testing
  - Ongoing effort – Intermediate Prototype
  - TRL 6 Demonstration Schedule
  - Technology Assessment

TPS Study Goals

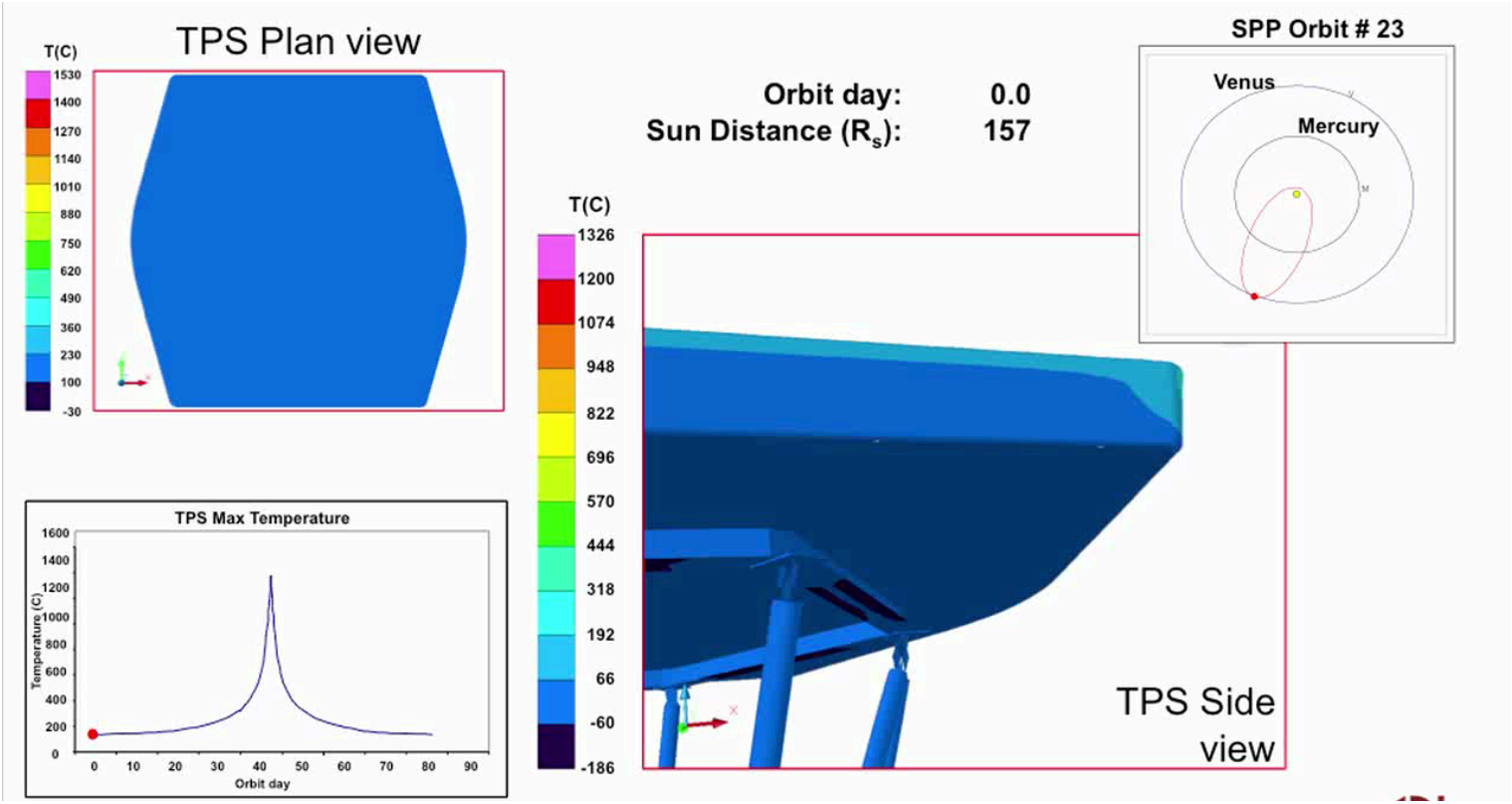
- Develop the technologies needed to show a feasible TPS concept and the design options available for the mechanical, structural, and thermal designs
- Define fabrication processes needed for making a thin shell C-C structure of the size required
- Perform material coupons testing needed to define basic material properties
- Develop fabrication processes for C-C/foam structural assemblies

SPP TPS Study Team

Douglas Mehoke	TPS Lead	The Johns Hopkins University Applied Physics Laboratory
Ed Eubanks	TPS Mechanical Design	The Johns Hopkins University Applied Physics Laboratory
Edward Schaefer	Structural Concepts and Evaluations	The Johns Hopkins University Applied Physics Laboratory
Shelly Conkey	Structural Analysis	Alliant Techsystems Inc., Space Systems
Gayle Knutzen	Thermal Analysis	Alliant Techsystems Inc., Space Systems
Elizabeth Congdon	Materials and Coatings, TSA Mechanical Design	The Johns Hopkins University Applied Physics Laboratory
Erin LaBarre	C-C Materials Testing	The Johns Hopkins University Applied Physics Laboratory
David Drewry, Ryan Deacon, Jeff Paulson, and Bob Matteson	Materials and Coatings	The Johns Hopkins University Applied Physics Laboratory
Aaron Brown and James Thompson	Carbon-Carbon Materials	Carbon-Carbon Advanced Technologies, Inc.
Ray Dirling	C-C Materials	Science Applications International Corporation
Ernest Ward and Bryan Leyda	Foam Structures	ERG Materials and Aerospace Corporation
Brian Williams and Ed Stankiewicz	Foam Structures	Ultramet Advanced Materials Solutions
Mike Droege	Foam Structures	Ocellus Inc.
James Klett, Jim Kiggans, David Harper, Chuck Schaich, and Ralph Dinwiddie	Foam Structure Testing	Oak Ridge National Laboratory
J. Gembarovic	Foam Structure Testing	Thermophysical Properties Research Laboratory
Ben Nielsen	Foam Structure Testing	Precision Measurements and Instruments Corporation
Dennis Nagel, James Spicer, Mark Buchta, Stephen Ryan, Kevin Hemker, Justin Jones, and Greg Schlichter	Coatings Design and Application	The Johns Hopkins University
Tom Wolf	Coating Properties	The Johns Hopkins University Applied Physics Laboratory
Steve Vernon, Gordon, Maahs, and David Weir	TSA Mechanical Design	The Johns Hopkins University Applied Physics Laboratory
Kyle Manning	Structural Analysis	The Johns Hopkins University Applied Physics Laboratory
Hadi Navid, Andy Webb, Ken Turner, Dennis Miller, Carl Clayton, Tony Scarpati, and Patrick Kilduff	Space Simulation Lab	The Johns Hopkins University Applied Physics Laboratory
Mike Steinfort	Foam Structures Testing	The Johns Hopkins University Applied Physics Laboratory

SPP Design Working Requirements - External Environment

- **The Solar Probe Plus TPS requires a fundamentally new design approach, based on the solar environment and the system mass**
- **Of the materials available, only C-C combines the strength, density, manufacturability, and CTE to make an effective heat shield**



SPP TPS Design Working Requirements

Environmental req	Value
Structural	Launch loads - Room temperature Thermal loads - Fligth temperatrues
Ionizing radiation	37 krad (Si)
Solar flux	70 W/cm^2
Dust impacts	survive dust env
Configuration req	
TPS mass (goal)	58.5 kg CBE
Support structure mass (goal)	10 kg CBE
Heat shield	234 cm (X axis) 226 cm (Y axis -flats)
Knife edge length	154 cm
	+/- 0.038 cm deformation at temperature
Umbra margin	2 deg
Support cooling radiators	4.5 m2
TPS bottom to rad top	8 in
Heat flow into radiators	<300 W (TBR)
Heat flow into S/C	<50 W
TPS Mechanical adjustment	**Value**
Total flatness	+/-0.10 cm
	measured at TBD # of points
During TPS fab	5 cm (TBR)
Translation during final assy - in plane	2.5 cm
-out of plane	2.5 cm
Science measurement req	
Mass loss	0.5 mg/s
Surface charging	< TBD V

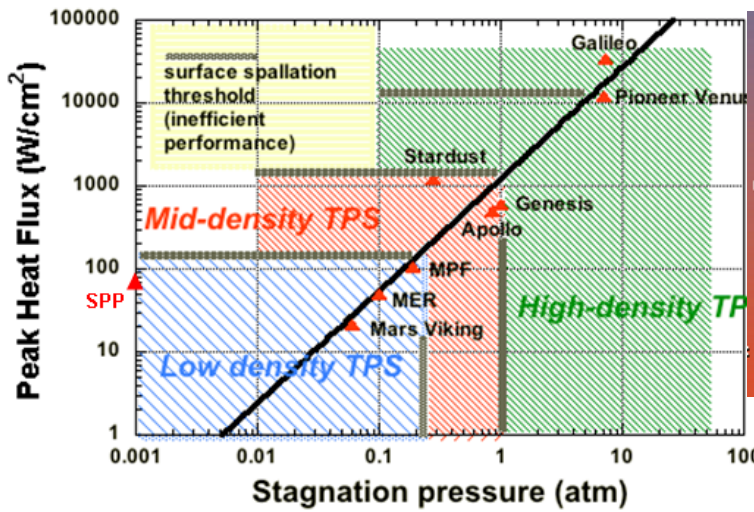
Derived requirement	Value
Coating a/e	0.6 (TBR)
Cone half angle	90 deg
External shield surface	no exposed C-C
Max temperature (C)	1400 C (TBR)
Temperature gradient	~ 1000 C
Insulation thermal conductivity	< 0.14 W/m C
Insulation density	< 0.05 g/cc
Bottom pan surface emissivity	< 0.8
Shield thickness	11.4 cm
Enclosed shell	Yes
Venting	~ 1 psi /sec
Rad MLI temperatures	< 220 C at CA
	> -120 C at aphelion
TPS bottom pan temperature	< 300 C (TBR)
Material usage temperatures	
C-C	< 2000 C
Ti, Be	500 C
Gr/Ep composite	350 C

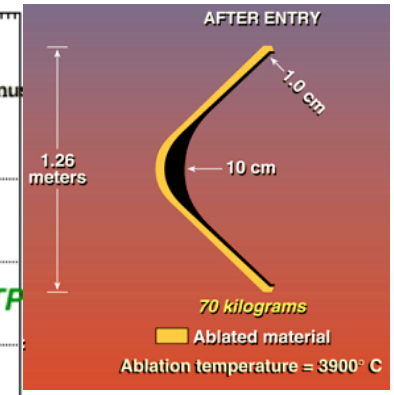
Options Considered – TPS Design Heritage

- There is considerable design heritage from re-entry designs that are applicable to the SPP TPS design
- However, there are key differences that make the SPP TPS a new type of design

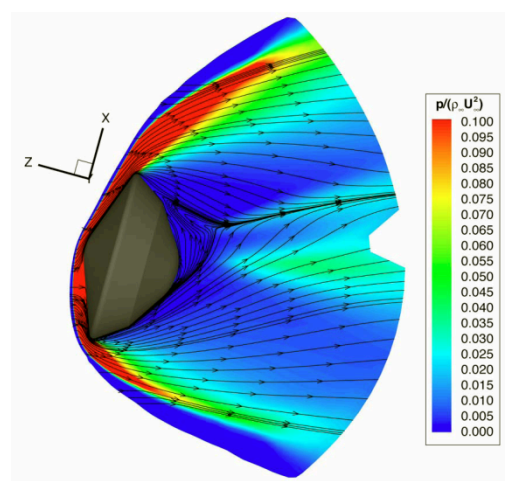
More severe requirements
- Mass loss
- Mass limitation
- Shell thickness
- Heating transient

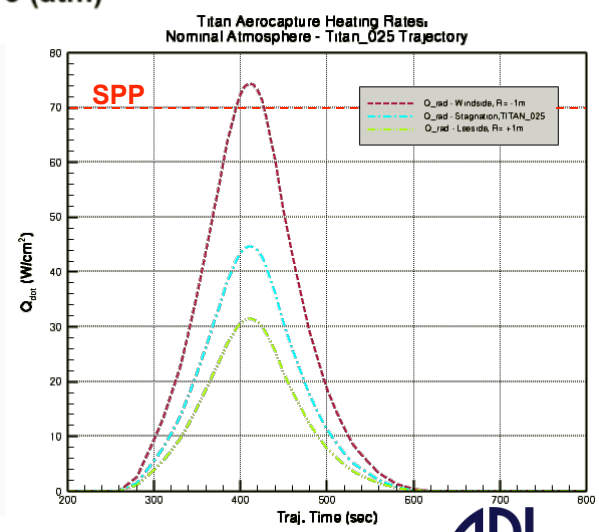
Less severe
- Aero loading
- Vacuum env





Galileo probe heat shield ablation



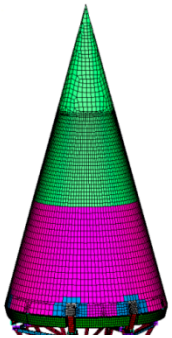


Options Considered - Previous Study Design Options

Genesis heat shield

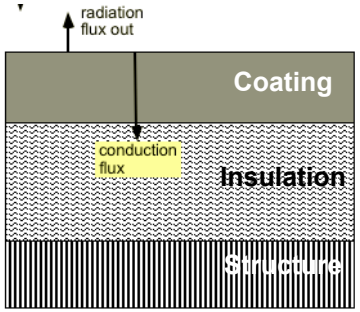
- Highly loaded joints in C-C area complicate design and analysis
- Large heat leak into bus through support system

SP RMS Primary heat shield



- The cone shape can be used to reduce the input flux. This reduction is limited below a 45 half cone

Non-ablative TPS



- Standard TPS design approach
- Lower risk
- Higher mass

Options Considered – TPS Study Design Approach

Component Level - complete

- The design approach used a series of trades in the mechanical, thermal, and materials disciplines, using a common design, to develop the different design alternatives and compare them equally

Subsystem Level - complete

- Once the basic design approach had been defined, a series of trades were conducted, including testing, with the related subsystems leading to a Reference Design

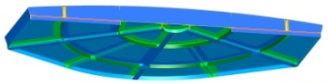
Reference Design - complete

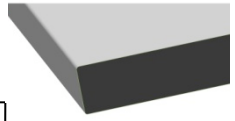
- The Reference Design provided a consensus approach for the feasibility study, it does not represent a baseline

TRL Demonstration Level – in process

- Develop the supported material property data, produce the required integrated analyses, design full-sized developmental test units, define needed fabrication processes, fabricate and test the TRL 6 Prototypes

Options Considered – Mechanical Design Trade Tree





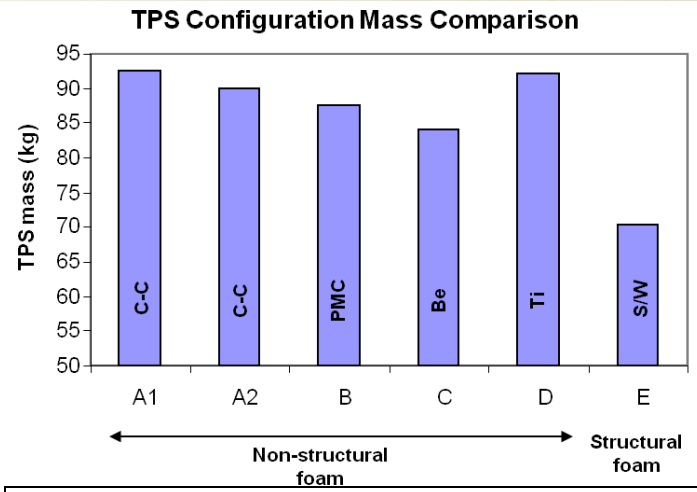
- **TPS**
  2.7m ∅, 152 cm KE
  - **Non-Structural Foam**
    - **Structural Pan**
      - **Foam**
        15 cm at center,
        7.6 cm at edges
      - **Top Shield**
        C-C Ring
        Stiffened Shell
      - **Bottom Pan**
        - **Config A**
          C-C
          - **A1**
            Overwraps
            92.6 kg
          - **A2**
            B-Studs
            89.9 kg
        - **Config B**
          PMC
          87.5 kg
        - **Config C**
          Beryllium
          84.2 kg
        - **Config D**
          Titanium
          - **D1**
            Isogrid
          - **D2**
            Sandwich Panel
            92.9 kg
  - **Structural Foam**
    - **Sandwich Shell**
    - **Sandwich Panel**
      - **Config E**
        Flat Panel
        11.4 cm foam
        70.3 kg
        - **Reference Design**
          Flat Panel
          11.4 cm foam
          2.34 m (Y), 154 cm KE
          55.3 kg
          - **TRL 6 Demonstration**

Options Considered – Mechanical Trades Conclusions (1 of 2)

- Configurations (B) & (D) provide the lowest technical risk.
- Configuration (E) has the potential for providing the lightest weight solution, but carries higher local risk

  A key design factor is characterizing the core failure and the bond to the carbon-carbon face sheets

- A large C-C shell, using the same materials planned for SPP, was constructed using an existing 2 m tool (courtesy Lockheed Martin Astronautics) as a pathfinder to test the feasibility of fabricating the SPP TPS

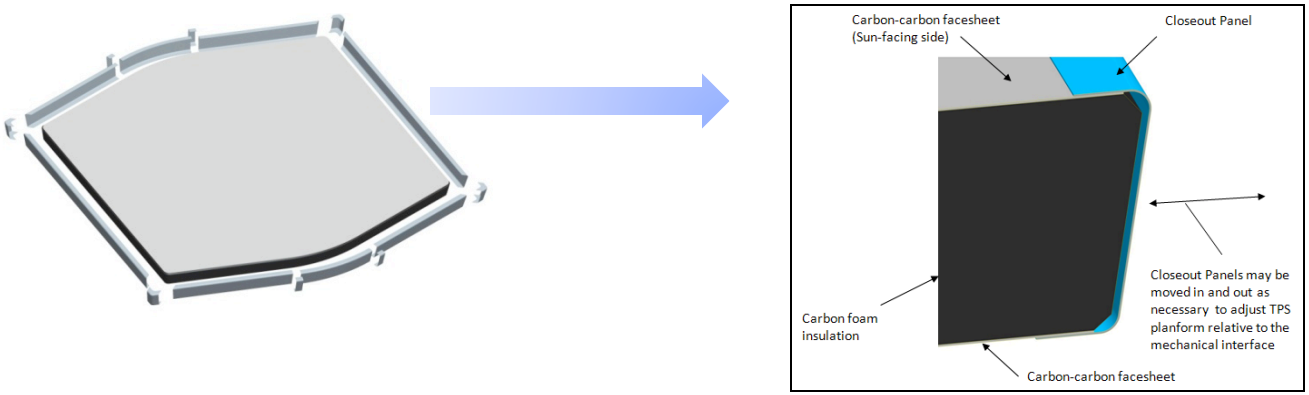


Note: All designs were taken through initial sizing analyses using the same size and shape, estimated masses are +/- 10% of expected final result

Options Considered – Mechanical Trades Conclusions (2 of 2)

Two types of TPS shimming are planned to minimize the radial distance between the C_p and C_g

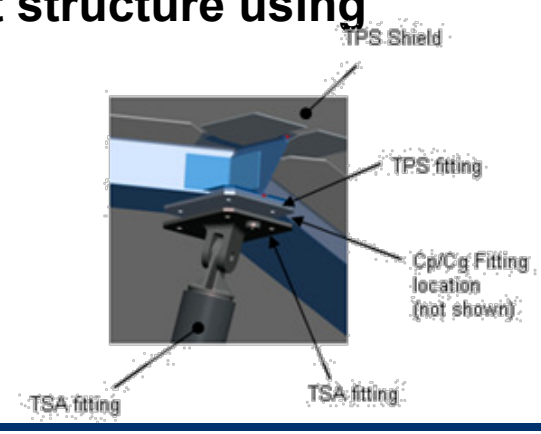
- **During Fab of the TPS (~ 5 cm translation using edge members)**



- **During integration of the TPS to the flight structure using**

  X-Y plane (mounting plane): ~ 2.5 cm

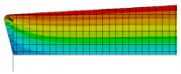
  Z axis (out of plane): ~ 2.5 cm



Options Considered – Thermal Design Options

Component Level

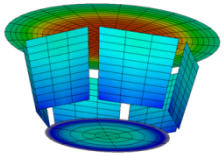
- Trades run on small component only level models



- Optical Properties
- Foam Conductivity
- Foam Thickness
- Foam Profile

Subsystem Level

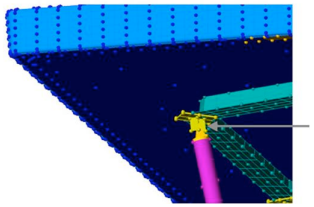
- Trades run on simplified TPS models with similar configurations



- Radiator Heat Leak Requirements
- Radiator Location
- TPS Radiator Spacing
- Foam Joints
- TPS Edge Closeout

Reference Design

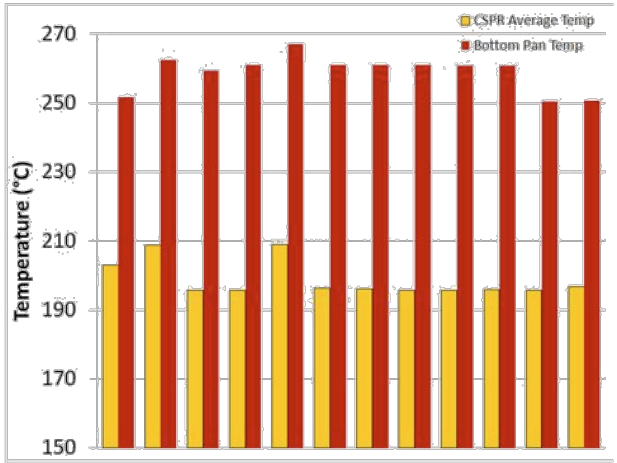
- Trades run on TPS integrated model

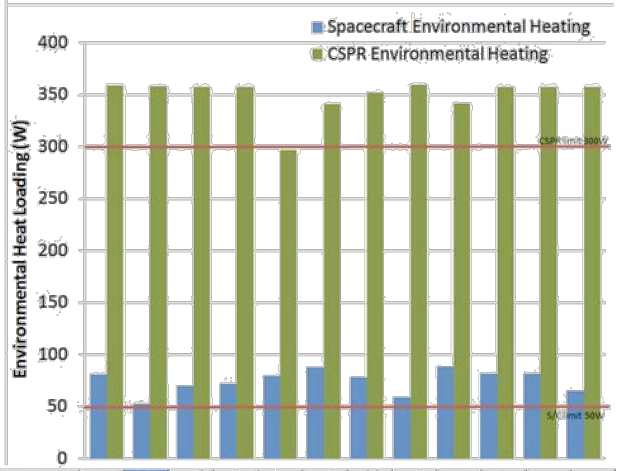


- Coating $\alpha_S/\varepsilon_{IR}$
- MLI e*
- Radiator Area

Options Considered – Thermal Design Conclusions

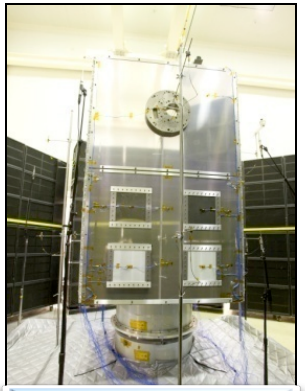
- Heat loading is the limiting requirement (with 30% margin), not the pan material temperature
- The parasitic heating into the radiator is a combined requirement involving radiator back MLI temperature and TPS pan temperature
  - Spacing between the radiator and TPS bottom pan is the critical parameter
  - The parasitic heating is not the limiting factor for the radiator temperature
  - Radiator MLI e* does not affect pan temperature
- For the configurations analyzed, the amount of foam in the TPS corners does not adversely affect the temperature profile along the edge
- Items under continued study include the coating a/e, and MLI e*



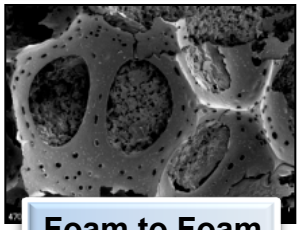


Barrier Blanket MLI e*	0.20	0.03	0.20	0.20	0.20	0.20	0.20	0.20	0.20	0.20	0.20	0.20
Spacecraft MLI e*	0.03	0.03	0.01	0.03	0.03	0.03	0.03	0.03	0.03	0.03	0.03	0.03
Top deck MLI e*	0.03	0.03	0.03	0.01	0.03	0.03	0.03	0.03	0.03	0.03	0.03	0.03
CSPR MLI e*	0.03	0.03	0.03	0.03	0.01	0.03	0.03	0.03	0.03	0.03	0.03	0.03
Strut-CSPR cond (W/°C)	39	39	39	39	39	9	113	39	39	39	39	39
Strut-S/C cond (W/°C)	31	31	31	31	31	31	31	29	32	31	31	31
Strut-TPS cond (W/°C)	35	35	35	35	35	35	35	35	35	30	40	35
Awnings	None	None	None	None	None	None	None	None	None	None	None	Yes

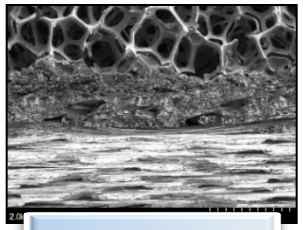
Options Considered – Material Studies



Acoustic Tests



Foam to Foam Bonding



SEM

Carbon-Carbon

- Mechanical Properties
- ACC-4 v. ACC-6
- SEM Imaging

Carbon Foam

- Mechanical Properties
- Thermal Conductivity
- Foam Packing Testing

Bonding

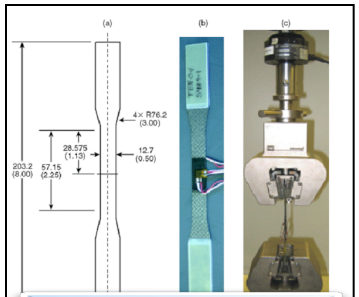
- Foam Bonding
  (High Temp Adhesives, Co-Bond)
- C-C/Foam Bonding
  (High Temp Adhesives, Co-Bond, Braze)

Analogs

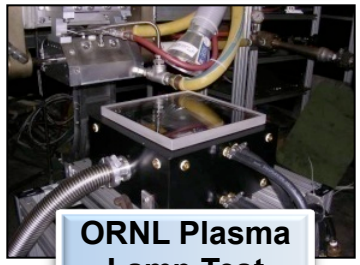
- T-Sections, Foam T-Sections, and I-Sections
- Co-bonding C-C to C-C

Sandwich Panel Design

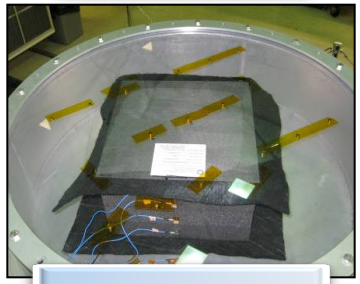
- Interface Bond Procedure/Surface Preparation
- Acoustic Testing (APL and GSFC)
- Mechanical Properties



Mech. Properties



ORNL Plasma Lamp Test

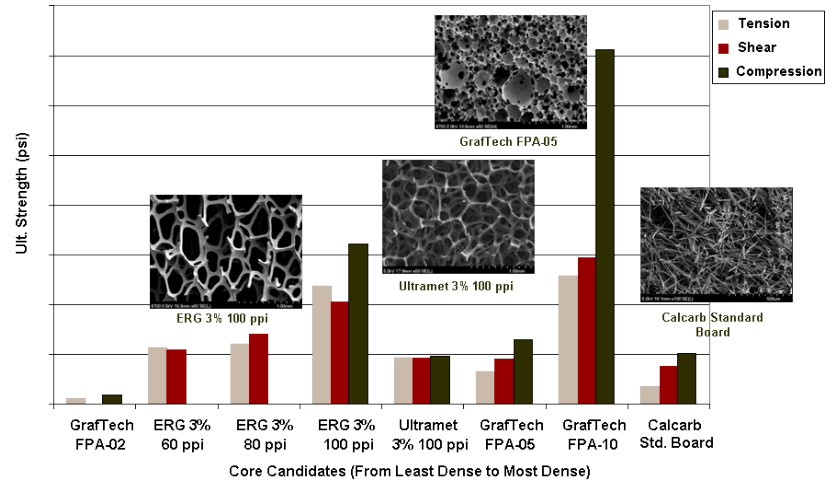


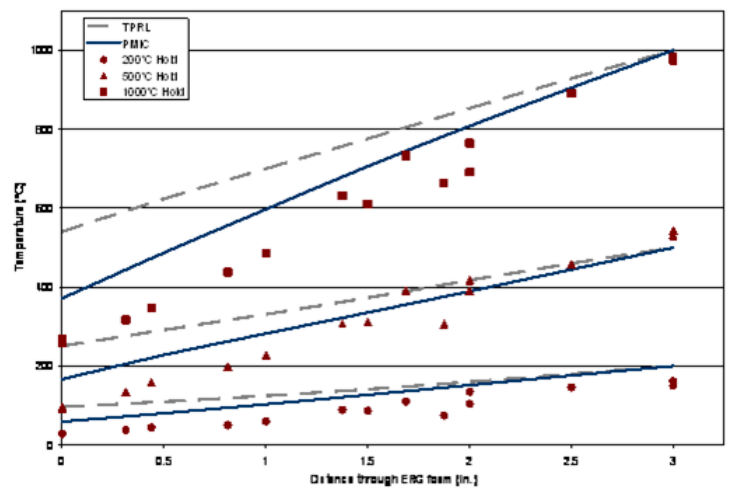
Packing Test

Options Considered – Material Study Conclusions (1 of 2)

<u>Carbon foam and bonding</u>

- There are two candidate foams with similar properties
- Carbon foam mech props do not scale with density
- A foam testing methodology has been developed
- Foam thermal conductivity measurements were not consistent, especially at high temperatures
- Scarf joints and discrete bonding will be used to pack the foam in the Reference Design
- Co-bonding method was successful in making a C-C/foam sandwich bond
- Analog testing used to develop design allowables





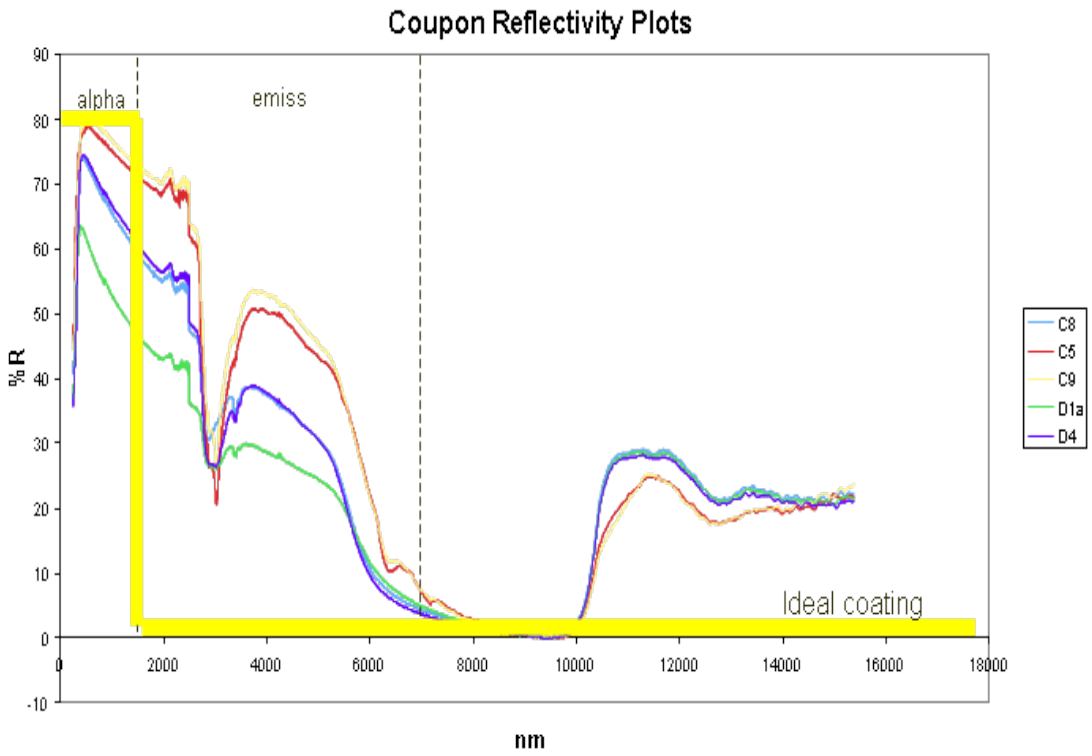
Options Considered – Material Study Conclusions (2 of 2)

<u>C-C</u>

- SPP ACC-4 was 2x stronger in tension than compression, not generally true for C-C
- SPP ACC-4 set tested was slightly stiffer in compression than tension, not typical of C-C
- Comparison of SPP ACC-4 with ACC-6
  - Similar or higher tensile strength
  - Similar elastic modulus in tension
  - Lower compressive strength
  - Higher elastic modulus in compression
  - Similar flexural strength
  - Similar or lower ILT strength (square coupons)
  - ILS and IPS in family (required thicker samples)
- The test plan for developing carbon foam, C/C and sandwiches material properties is underway

Coating Study

- The temperature of the TPS is driven by the sun distance and its optical properties
- The high operating temperature of the front surface means the solar absorptivity and the IR emissivity are coupled much more than is usually found
- The study has been looking into the parameters that control the shape of the reflectance curve over the wavelengths of interest



Coatings – Optical Properties

What makes a coating white?

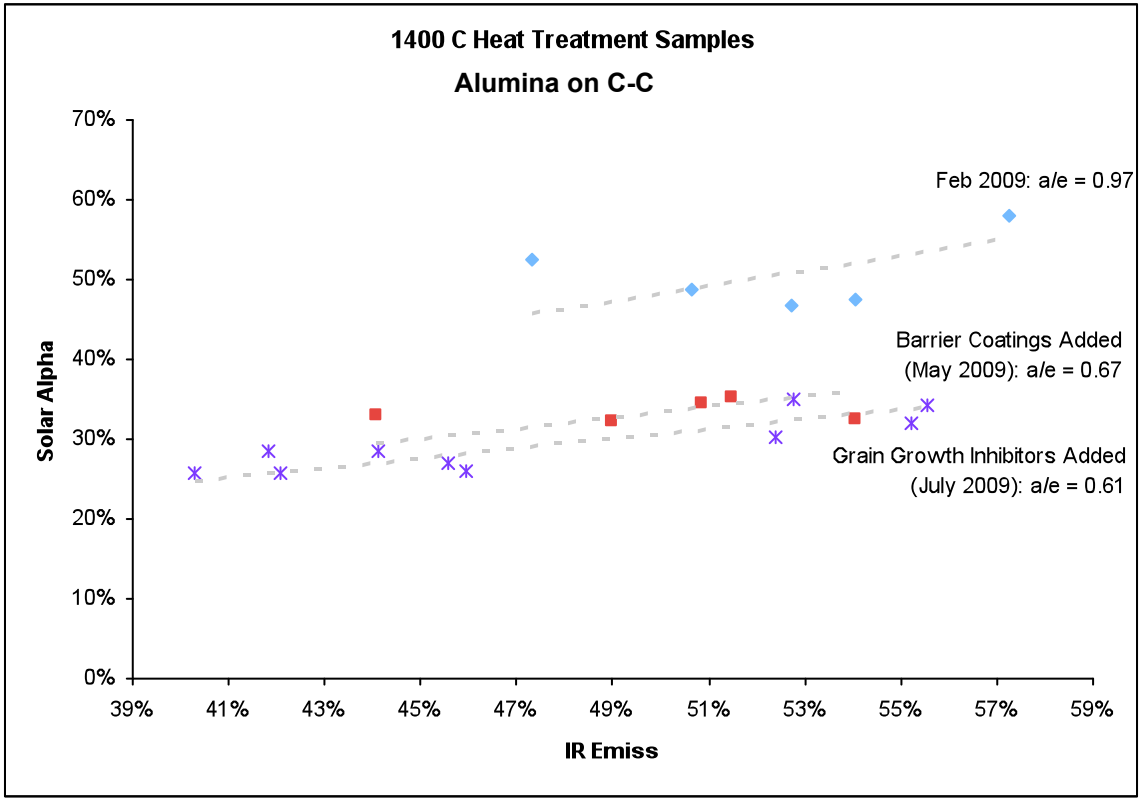
- **Test data, taken at room temperature has shown we are near, but not consistently under the design limit**
- **The effects of coating microstructure are driven by porosity**

Coating options

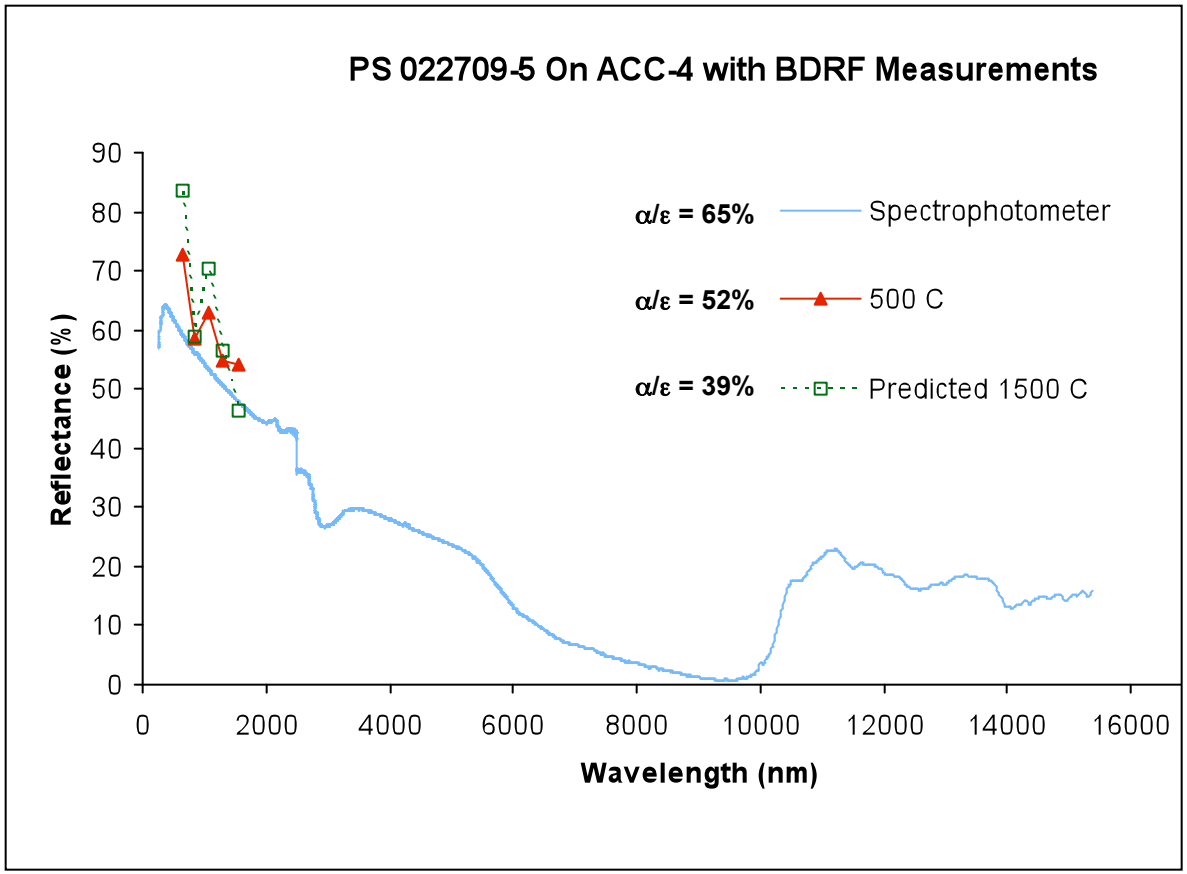
- **Alumina**

and

  - **Yttria**
  - **Encapsulated coatings**
  - **CVD coatings (BN)**
  - **Nextel**

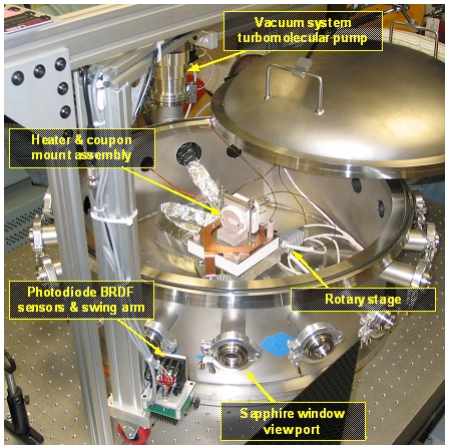


Coatings – Effect of Temperature



HT-BDRF Measurements

- **Currently, test data has been taken at room temperature and 500°C**
- **α_s decreases significantly with temperature**
- **Lower absorptivities increase the coating's emissivity**
- **Continuing to increase BDRF temperature capabilities**



Coatings – Path Forward

Continue Coating Optimization

- Looking at intra-grain v. inter-grain effects with temperature
- Limit pore size to ~ 25 microns
- Add micron scale particles with a different index of refraction

High Temperature Optical Properties

- Improve BRDF temperature capabilities
- ORNL optical property testing

Environmental Exposure

- Thermal cycling
- Radiation exposure
- UV exposure

Update $\alpha_S/\varepsilon_{IR}$ Design Limit

- BOL value - As sprayed
- EOL value – Cumulative effects of environmental exposure

Reference Design - Feasibility

Focus on feasibility

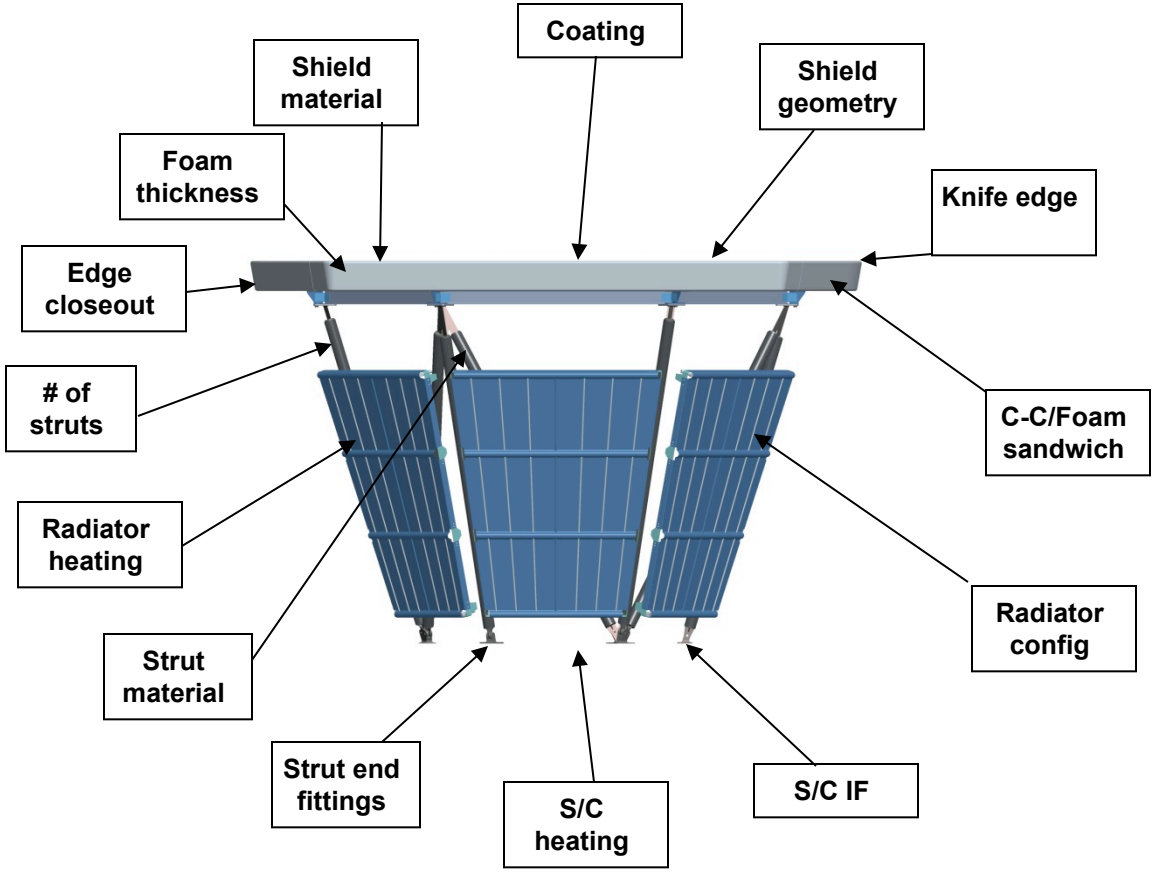
- Reference Design used to show system performance meets working requirements
- Key trade space items identified
- Design options include fall-backs for the flight phase

Reference design is feasible

- The simple design builds on existing TPS heritage
- New technologies are developed in step-wise approach with backups
- The benefit in the sandwich design outweighs its higher risk

Materials testing shows needed property data can be generated

- Completed testing shows materials have required properties
- Use of other existing databases allows pooling of compatible data
- High structural loading of system is at room temperature during launch
- High thermal loading of system is at the closest approach



TPS Peer Review Summary

A TPS Configuration Peer Review was held on 4/30/09

The review team consisted of the following

J. Ercol	APL	Chair	Thermal	SPP Thermal, cooling sys
T. Michalek	NASA		Thermal	external reviewer
K. Hemker	JHU		Mech Eng	external reviewer
R. Dirling	SAIC		C-C	external reviewer (not present at meeting)
M. Lockwood	APL		Systems	SPP S/C System Eng
T. Hartka	APL		Mech Eng	SPP S/C Mech Eng
D. Persons	APL		Structural	APL SD Struct Sec supervisor

Minutes published in SEM-09-4-484, May 12, 2009

Summary comments:

No formal action items were submitted, but suggestions were addressed to the team

The design was noted as being much more mature than the earlier work

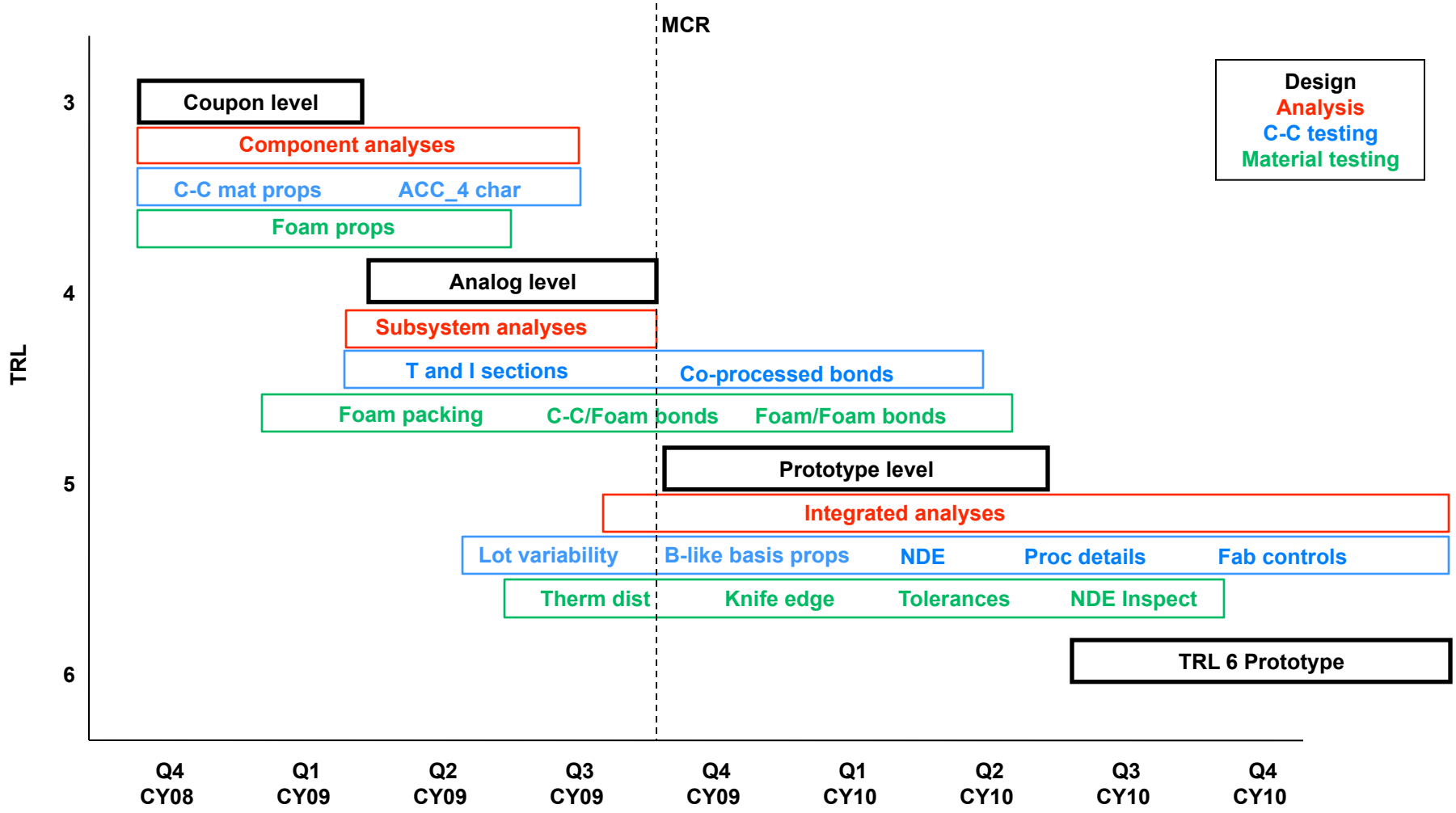
While the Reference Design was noted as requiring more developmental effort, it is also allows for a much simpler fabrication effort

No serious problems were brought up that would challenge the feasibility of the design

Phase A/B Plans

- **Plan to demonstrate TRL 6 by Program PDR**
  - Develop requirements specification
  - Continue detailed design evolution
  - Materials testing program
    - C-C materials testing
    - Carbon foam materials testing
    - C-C/foam bonding
  - Intermediate prototype (91 cm x 91 cm, with edge close out)
    - System flatness
    - Foam-foam bonding
    - Foam machining and tolerances
    - C-C/foam sandwich structure fabrication
    - Straight edge tolerance control
    - Mechanical testing
  - TPS - Mechanical Test Unit (Phase A/B)
    - Full size TPS system testing at GSFC (static load, vib, acoustic)
  - TPS - Thermal Test Unit (Phase A/B)
    - Full size TPS section testing at JSC (thermal vac)

TPS Development Schedule



Solar Probe Plus Mission Concept Review

Solar Array Cooling System

Carl J Ercol

The Johns Hopkins University
APPLIED PHYSICS LABORATORY

Cooling System Outline

- Cooling System Approach
- Options Considered
- Reference Design
- Feasibility and Testing
- Technology Development
- Phase A Plans

Cooling System Approach

The solar array cooling system is the critical design feature that helps to enable the SPP mission. While building off space flight programs, like MESSENGER and space station based liquid cooling systems, the SPP cooling system is a large step forward in the state-of-the-art for space cooling systems. The overall system goal is to demonstrate that the cooling system is at TRL 6 by the Program PDR (8-2011). **The Pre-Phase A activities were aimed at showing the feasibility of an active pump loop cooling system for solar array temperature control. The detailed design and testing of a prototype system will be carried out as part of the flight program, starting in Phase A.**

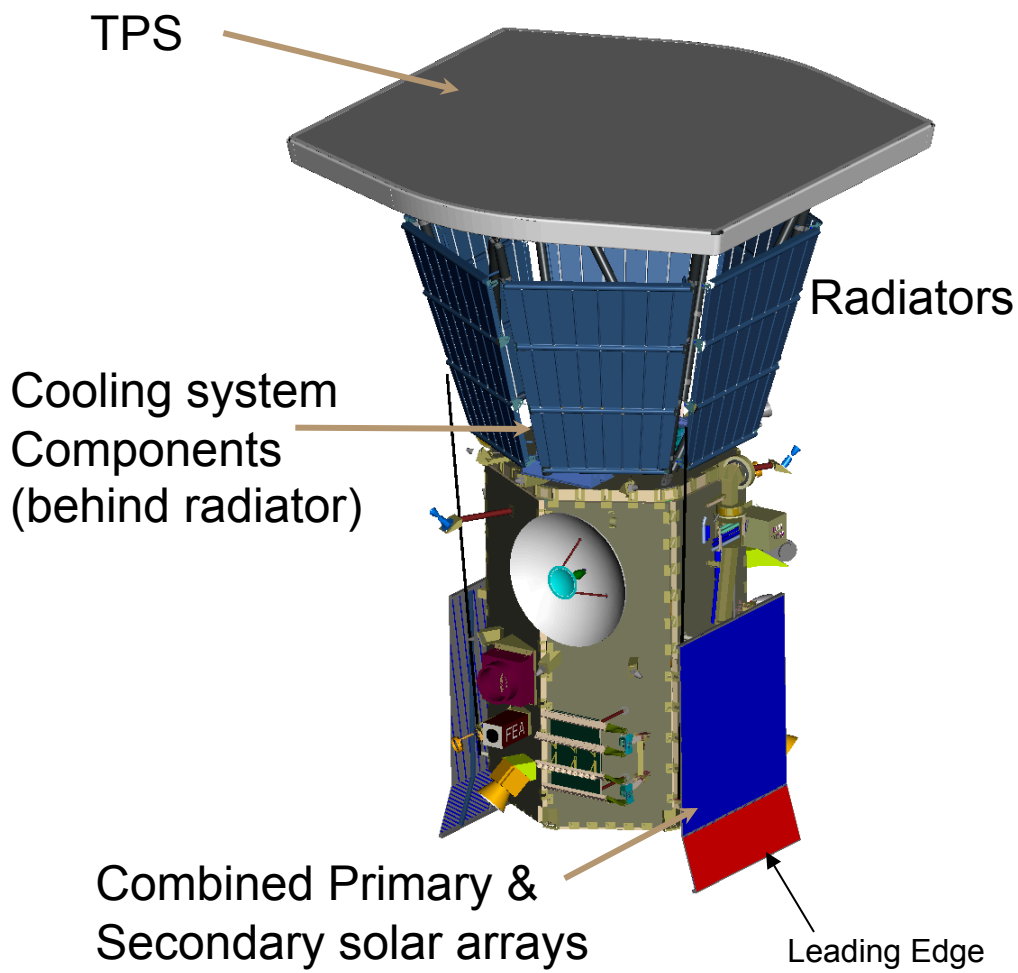
Top Level Requirements

Assumed flow-down requirements:

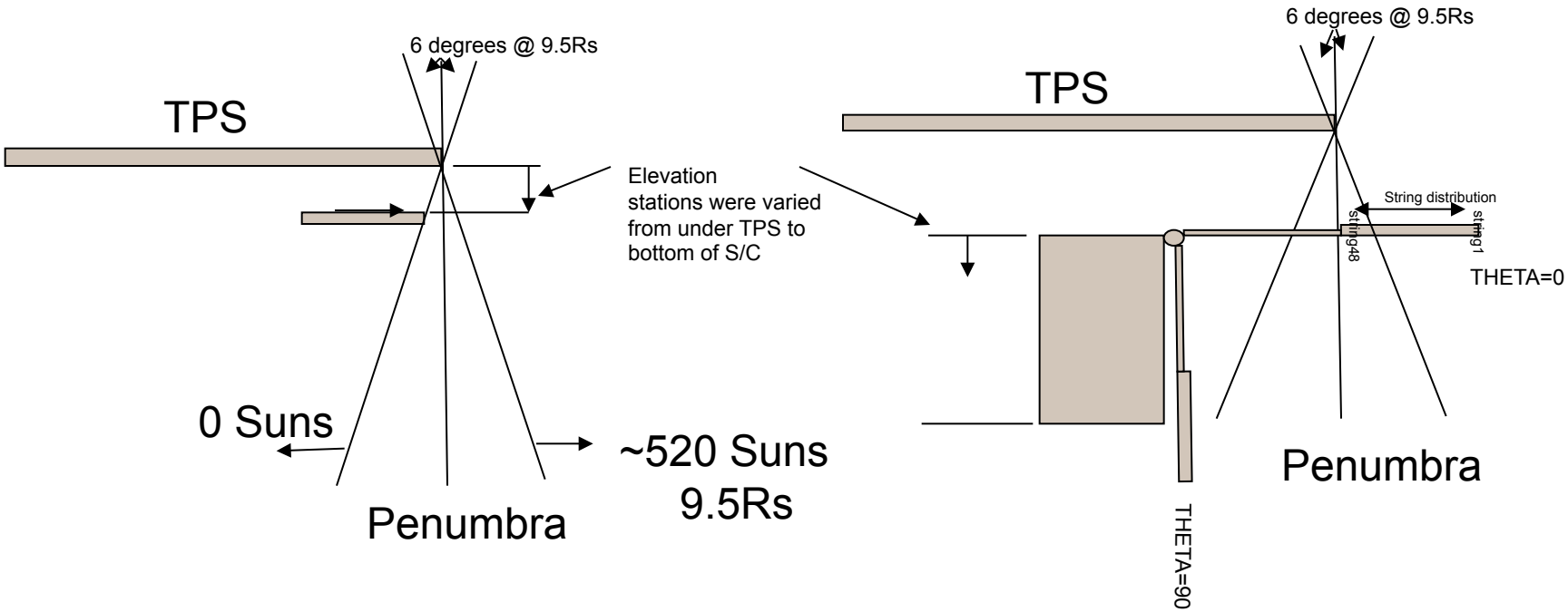
- **Provide primary and secondary solar array cooling over the sun distance range of 0.0442 (9.5 Rs) to 1.02 (215 Rs) AU**
- **Mass Target < 60 Kg**
- **Power Target < 60 W**
- **Comply with the environmental, mission assurance and safety requirements**

Cooling System Overview

- Power provided to spacecraft loads through combined primary and secondary solar array assembly (SAA)
  - Two SAA's make up the power system
- Secondary arrays, denoted as red, use concentrator type cells designed for high illumination applications.
  - Primary array section, denoted as blue, uses a modified conventional space qualified solar cell
- Cells mounted on cooling plate will be rotated downward into the TPS shadow to maintain constant solar flux as SPP approaches the Sun.
- The associated thermal power, 0.8 KW to 2.5 KW per wing, is rejected through remote radiators connected to each wing by the pump cooling loop
  - maintains solar cell temperatures within their allowable operating range



Options Considered in Pre-Phase A Sliding vs. Rotating Concepts



Options Considered in Pre-Phase A

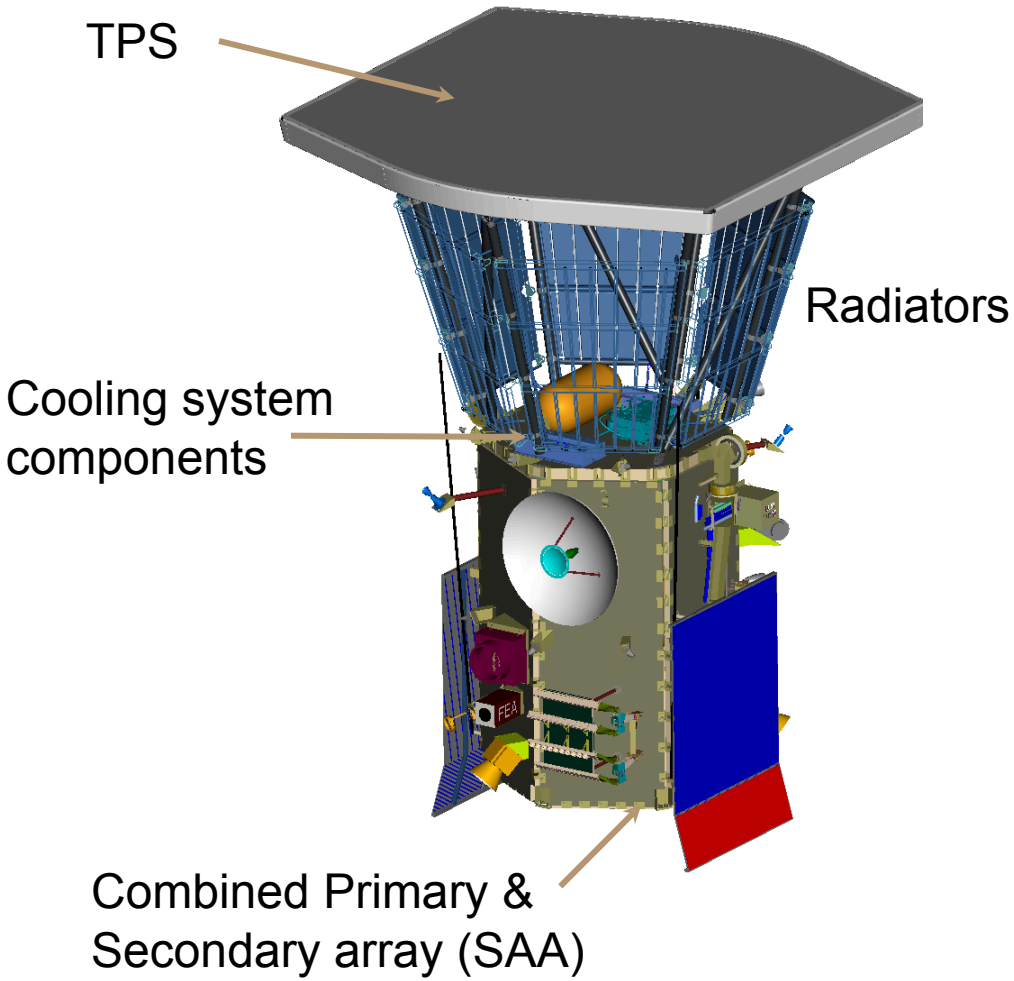
Actively cooled rotating secondary solar array / passively cooled rotating primary solar array ✗

- Sliding verses rotating secondary solar arrays
  - rotating was analytically more robust and insensitive to placement relative to the TPS
  - linear travel had small exposed area / questionable power generation
- required large MESSENGER class primary solar array that combined OSR's with cells
- drove minimum fluid temperature
- complicated in-flight operation and mechanical packaging

Combined secondary and primary solar arrays ✓

- both utilize liquid cooling and share rotation system
- keeps primary array small by eliminating OSR's
- keeps cooling fluid above 0 C

Reference Design



- Combines primary and secondary array into an SAA
  - Entire SAA is actively cooled
  - Eliminates low temperature issues
    - cooling fluid was optimized for hot operation
      - water
  - Utilizes the solar array surface area as part of the radiator system
    - reduces the required radiator under the TPS
  - Simplifies mechanical packaging and orbital operation

- **Chart removed**

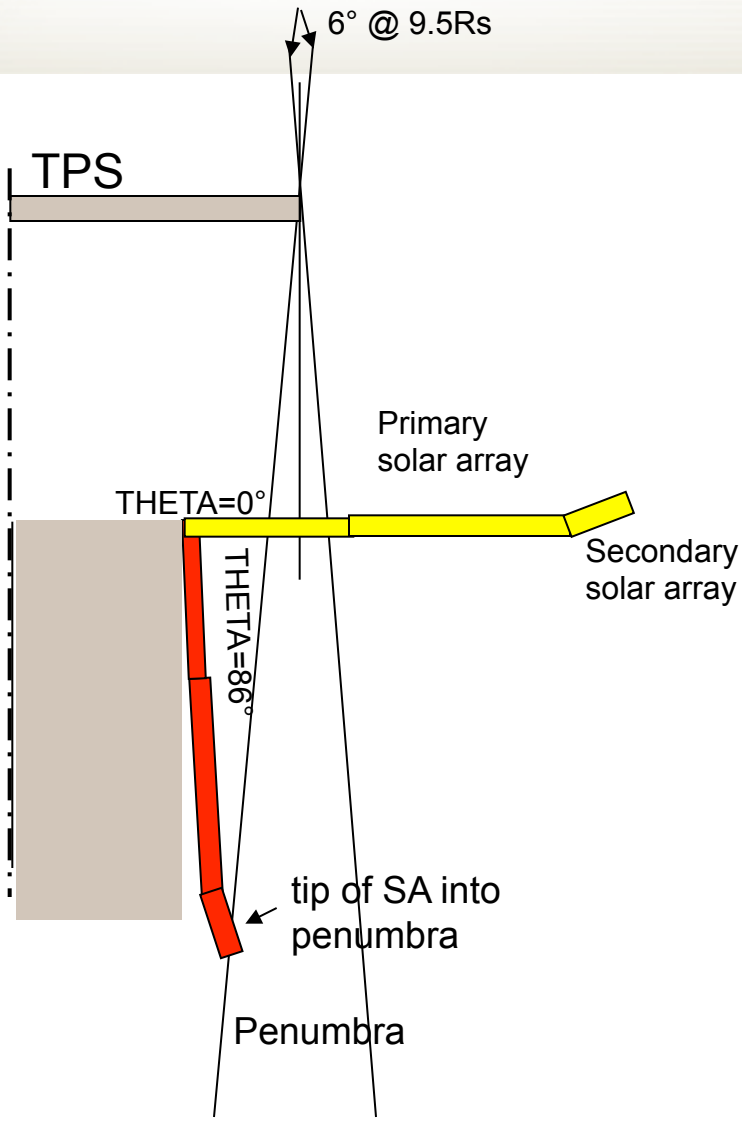
Reference Design
Cooling System Review Summary

- Based on the material presented the board agreed that the concept is feasible. The following bullets represent the major comments that were recorded:
  - the analytical approaches taken to quantify the thermal environments and the cooling system performance was acceptable
  - although a end-to-end system like this has not been built, the concept design utilizes components that have ISS heritage in water and ammonia systems
  - manufacturing of the mini-channel cooling panels and radiators could be an issue
    - Mass
    - Process
  - Freeze protection
    - Launch
  - Dust protection
    - punctures

Reference Design
Cooling System Requirements

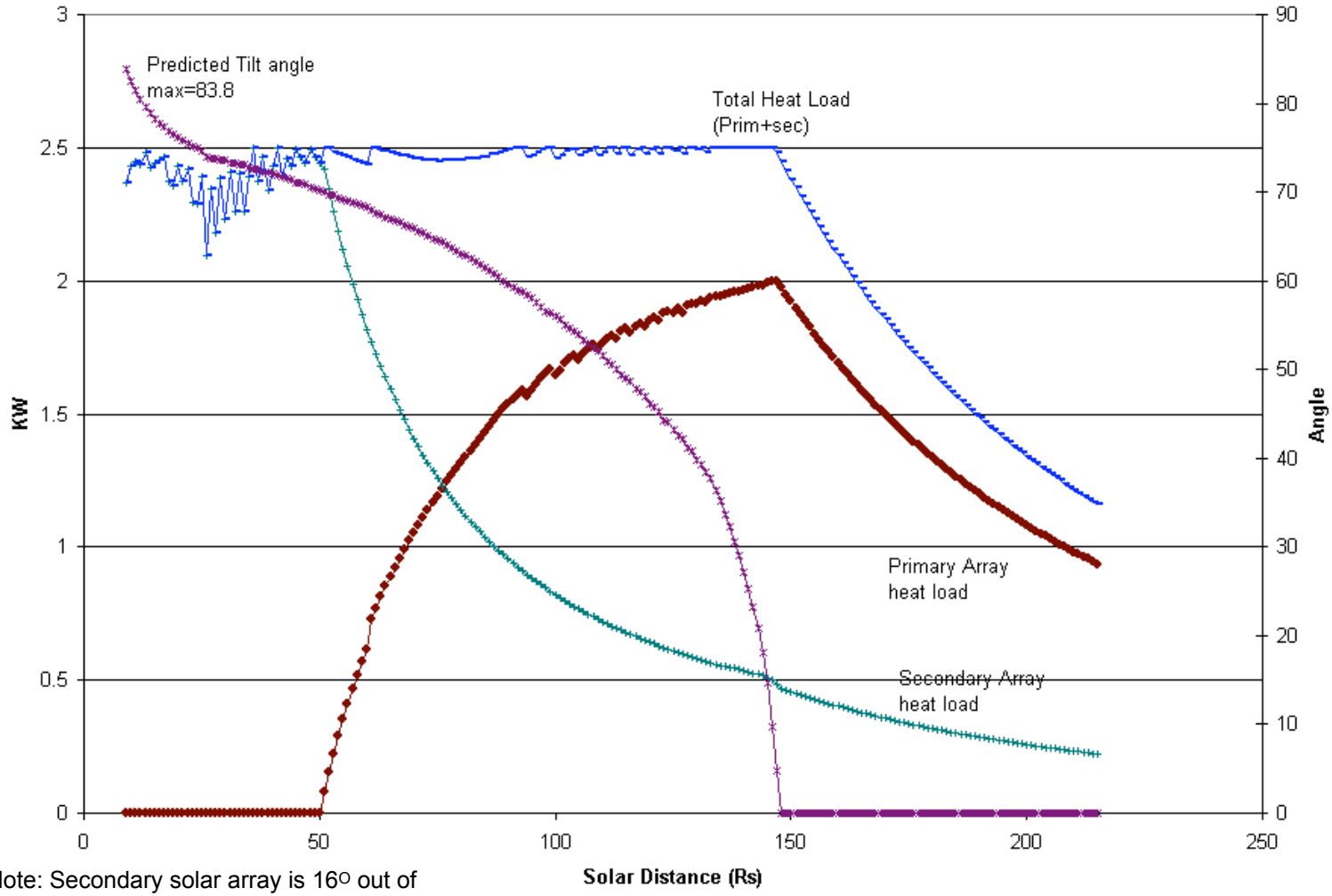
- 150 °C Maximum Solar Array Interface Temperature
- Solar Array Assy (SAA) TCS Budgets:
  - Mass target: 60 kg
  - Power target: 60 W
- Operating Fluid Temperature:
  - 10 °C to 150 °C
- Survival Fluid Temperature:
  - 200 °C
- Heat Load:
  - 2,500 W into each SAA
- 4K Radiator Space Heat Sink
- Physical Dimensions:
  - SAA: 1.1m (44”)(L) x 0.76m (31.5”) (W)
  - Radiator Panels plus SAA’s: Max Total Area = 5.0 m2
- Temperature Gradient:
  - Parallel to Deployment Arm: 50 K/1.1 m (44”)
  - Perpendicular to Deployment Arm: 10 K/0.76 m (31.5”)
- Heat Input Zone:
  - Orbit Dependent per APL input
- Mission Life:
  - 10 years
- TRL6 by PDR 8/2011

Reference Design
Combine Solar Array



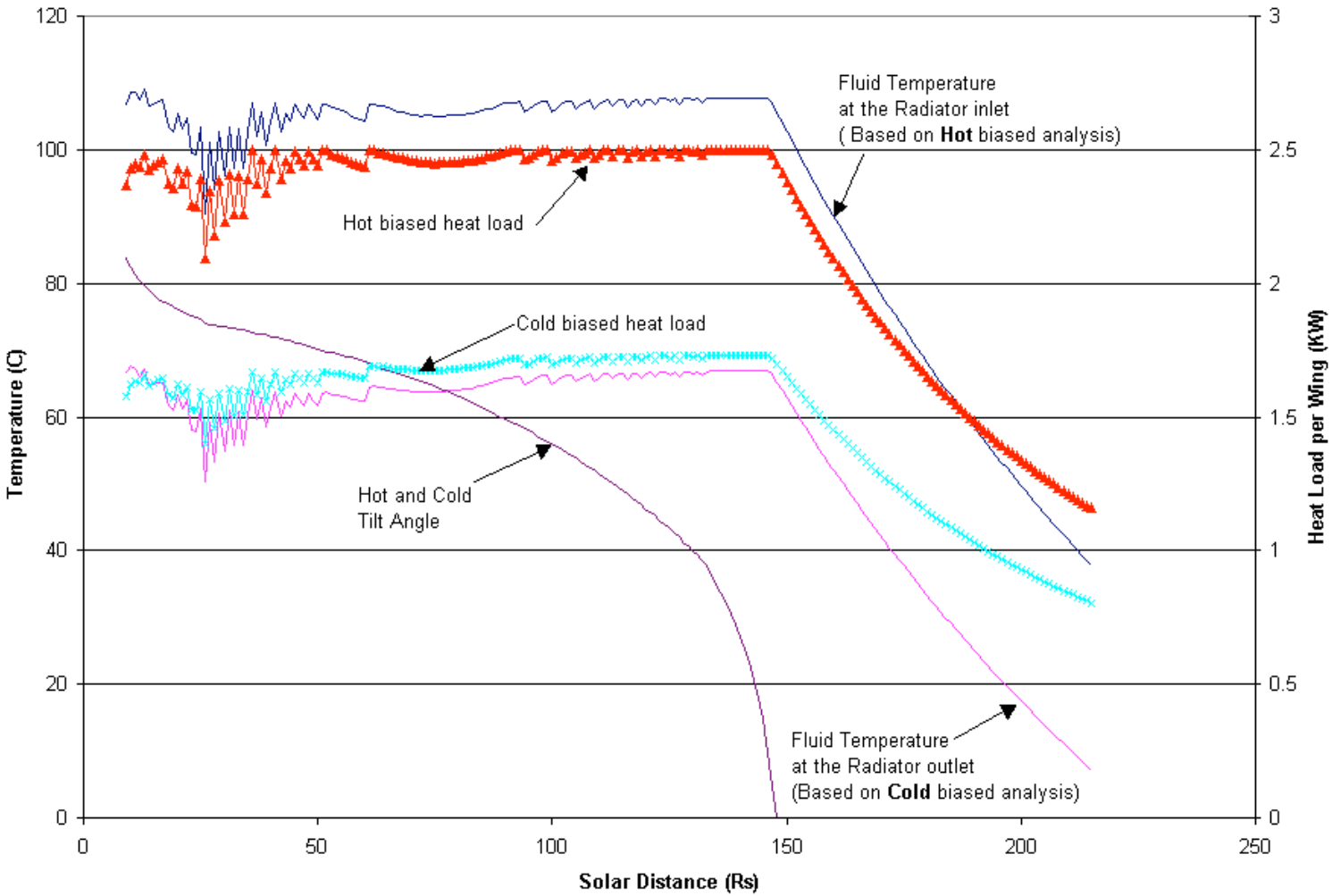
- Thermal Input
  - maintain system maximum thermal input at 5KW or 2.5KW per side
- Geometry
  - ~3.8 sqM of under TPS radiator combined with solar array area
- Peak Suns
  - under nominal conditions keep maximum incident heat flux below 40 Suns
- Maximum solar cell interface temperature
  - varies as a function of heat input
    - analysis done by closed system using radiators to control bulk temperature
- Combined operation
  - 9 to 215 Rs

Reference Design
Heat Load Analysis

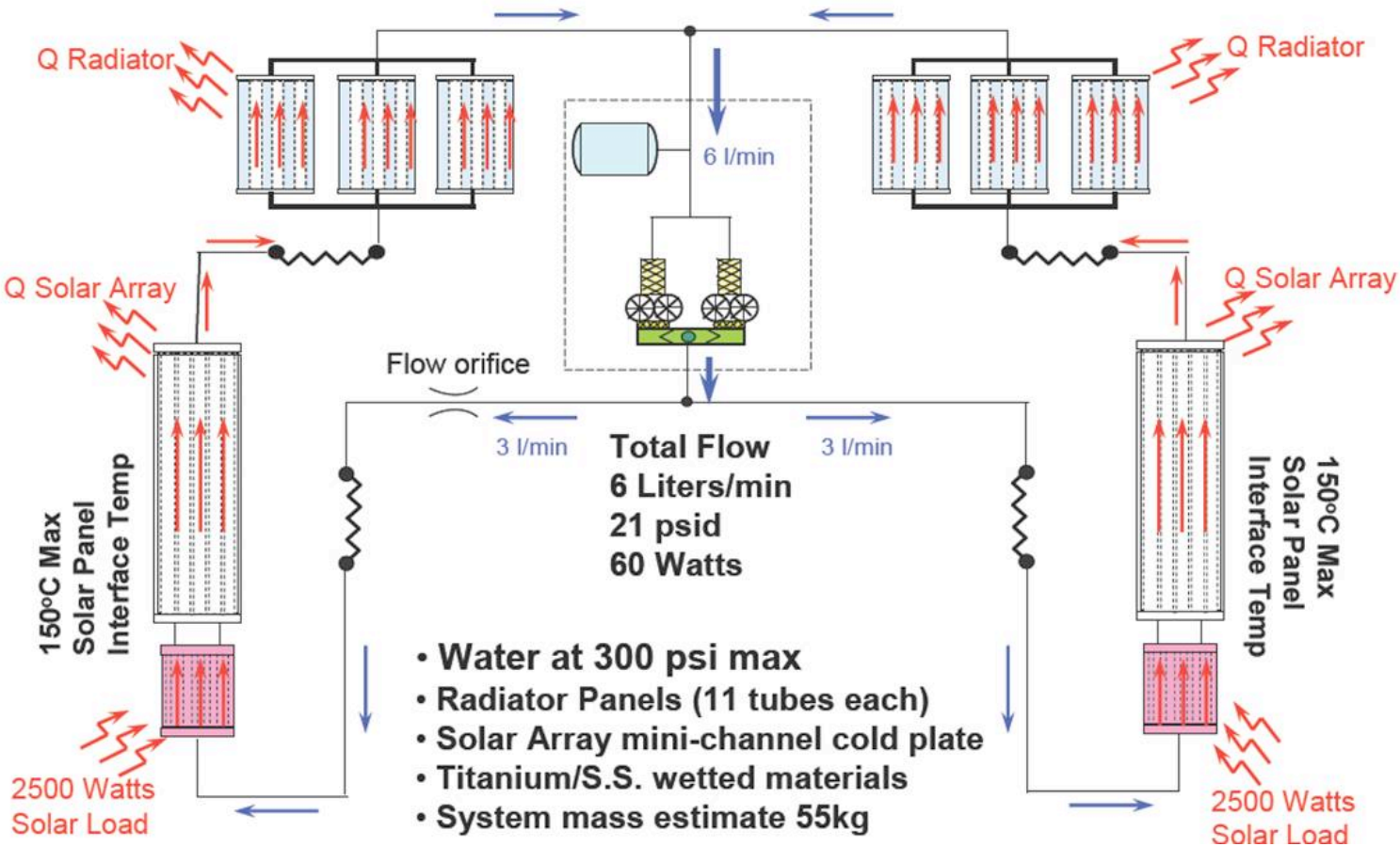


Note: Secondary solar array is 16° out of plane with the primary solar array

Reference Design
Mission Performance Analysis



Reference Design
Cooling System Schematic

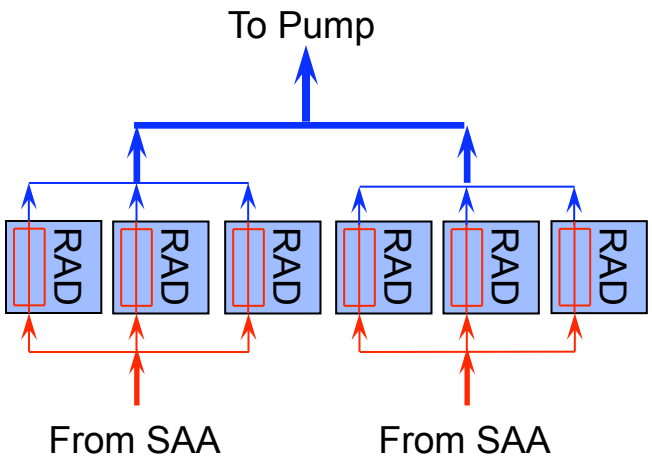


Reference Design
Cooling System Summary

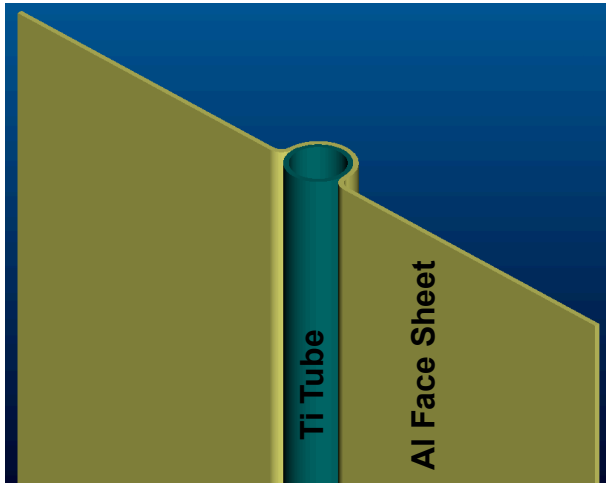
- A single phase pump loop system was chosen as the most efficient technology to remove and transport solar array heat
  - 18 different active and passive TCS configurations were evaluated and the results are summarized in the Cooling System Report
  - driven by target mass, power and solar cell interface temperature requirements
    - mini-channel cold plate design
      - balances hydraulic and thermal performance
    - water chosen as the working fluid
      - titanium and stainless steel materials for compatibility with wetted surfaces
- Development plans are focused to raise the TCS to TRL-6 by Spacecraft PDR
  - August 2011

Reference Design
Main Radiator Schematic

- **Heat Rejecting Capability (6 panels): 3200 W @100 °C**
  - **5 KW nominal thermal input**
  - **Total Surface Area: 3.8 m2**
  - **View Factor to Space: 1**
- **Panel Face Sheet: 0.51 mm (0.02”) Thick Aluminum**
- **Panel Emissivity: 0.8**

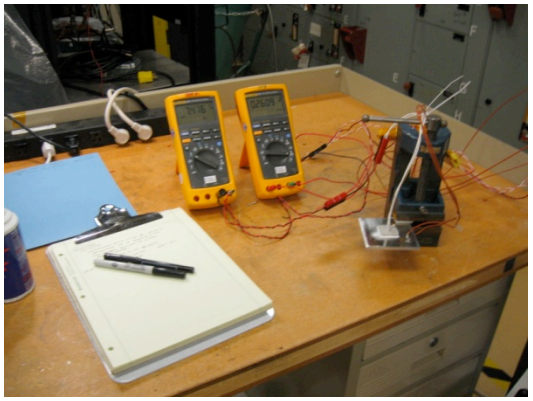


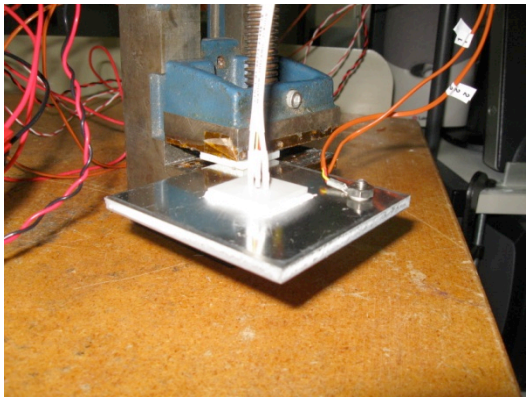
Coolant Flow Arrangement



Panel Structure

High Flux Heater and Solar Cell Adhesive Testing





- Purpose
  - to verify capability of test heaters to be used in the larger scale heat exchanger test
    - aluminum nitride technology allows heaters to operate at high power and temperature
      - 500 W/in^2 and 400 ^{o}C
  - collect test data on various high temperature adhesives that could be used for bonding solar cells.
    - verify high temperature capability and measure thermal conductivity as a function of power and temperature
- Test Configuration
  - 1” X 1” heater bonded with five different high temperature silicon adhesives
- Test Input Parameters
  - steady state operation at seven power settings
    - Equivalent Suns of 2,5,10,20,30,40, and 50 were simulated for each adhesive
- Test Results
  - steady temperatures measured as a function of simulated Suns

Technology Development Phase A Plans

- System-Level Modeling
  - plumbing freeze protection
  - asymmetrical heat load to solar arrays
  - overall system architecture and system level thermal and hydraulic performance based on breadboard test
- 1/6 Scale Cooling system breadboard and Solar Panel Platen Development
  - evaluate candidate manufacturing techniques for the platen by fabricating scale assemblies
    - Mini-channel sub-scale performance testing
      - 1/3 width x full length cooling area
  - 1 full scale radiator panel
  - representative hydraulic hardware
- Pump Testing
  - fabricate a test pump head end assembly for model correlation and testing

Phase A/B Plans

- Use Phase A data from breadboard testing to refine system model for the proto type system
  - Breadboard cooling system (thermal/hydraulic)
  - Structural radiator
- Design and fabrication a full scale proto-type cooling system to demonstrate TRL-6 by August 2011
  - hydraulic and thermal performance
    - Electronics and motor components
    - Specialty materials
    - cold plate fabrication

Solar Probe Plus
Mission Concept Review

Power Subsystem

Lew Roufberg
Power Subsystem Lead Engineer

APL
The Johns Hopkins University
APPLIED PHYSICS LABORATORY

Power Subsystem Outline

- Requirements
- Options Considered
- Reference Design
- Feasibility
- Technology Development
- Phase A Plans and Trades

Working Requirements

Assumed flow-down requirements:

- **Provide bus voltage within the range of 22 V to 35 V for spacecraft bus and instrument components**
- **Provide load power over the sun distance range of 0.0442 AU (9.5 Rs) to 1.02 AU**
- **Provide load power in accordance with the load power budget for all specified modes throughout the mission**
- **Receive commands and send telemetry from/to the avionics over redundant serial digital interfaces**
- **Comply with the environmental, mission assurance and safety requirements**

Options Considered in Pre-Phase A

Solar Cell Assembly for Secondary Section of Array

- Space vs. terrestrial (used with concentrators) solar cell assembly
- Both options are being pursued

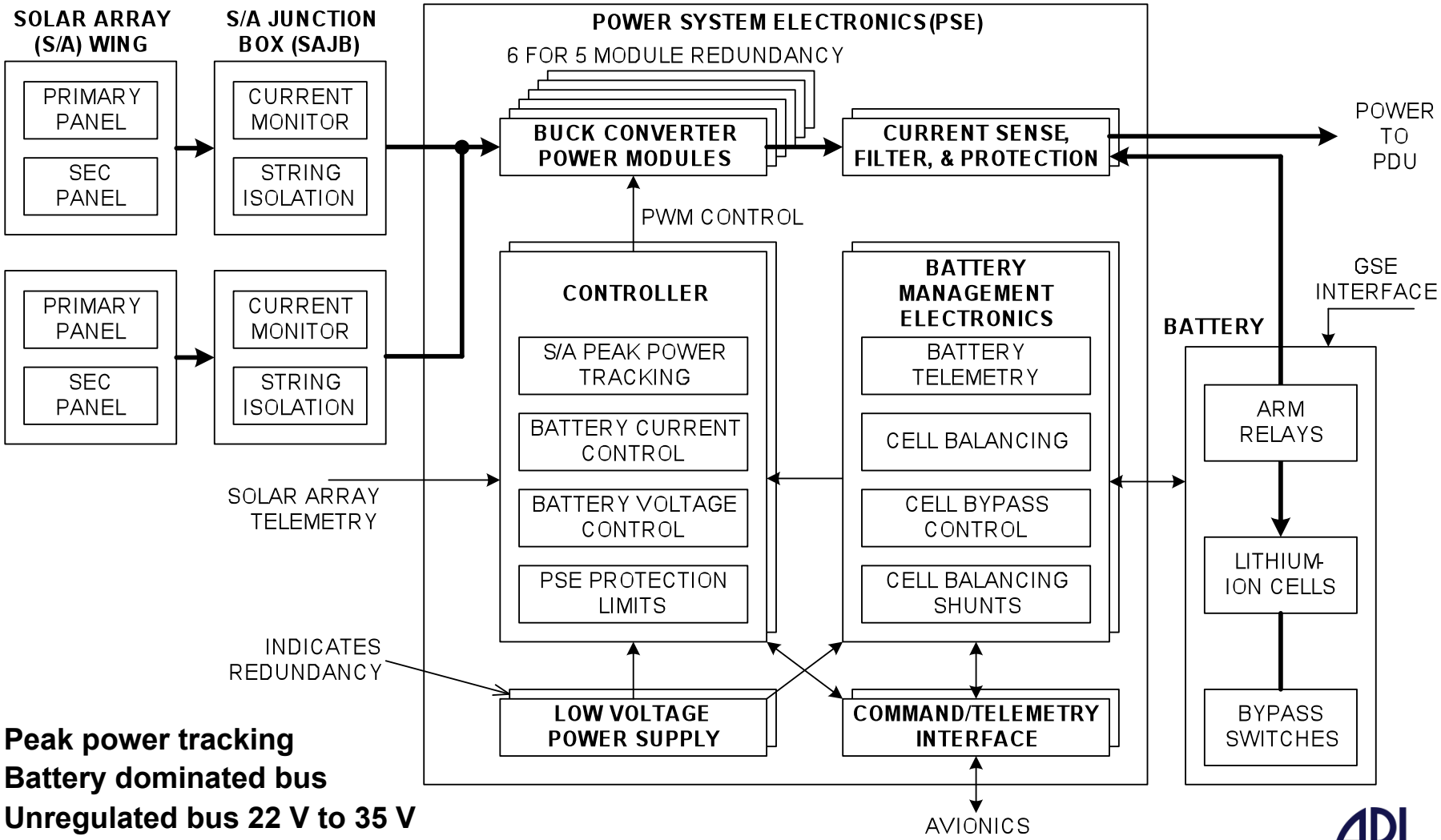
Battery

- Lithium ion has been selected to minimize mass
- Lithium ion has significant space flight heritage

Power Electronics

- Peak power tracking (PPT) versus direct energy transfer (DET)
- PPT assumed for power calculations based on MESSENGER heritage
- Trade study to be conducted during Phase A

Power Subsystem Functional Block Diagram



- **Peak power tracking**
- **Battery dominated bus**
- **Unregulated bus 22 V to 35 V**

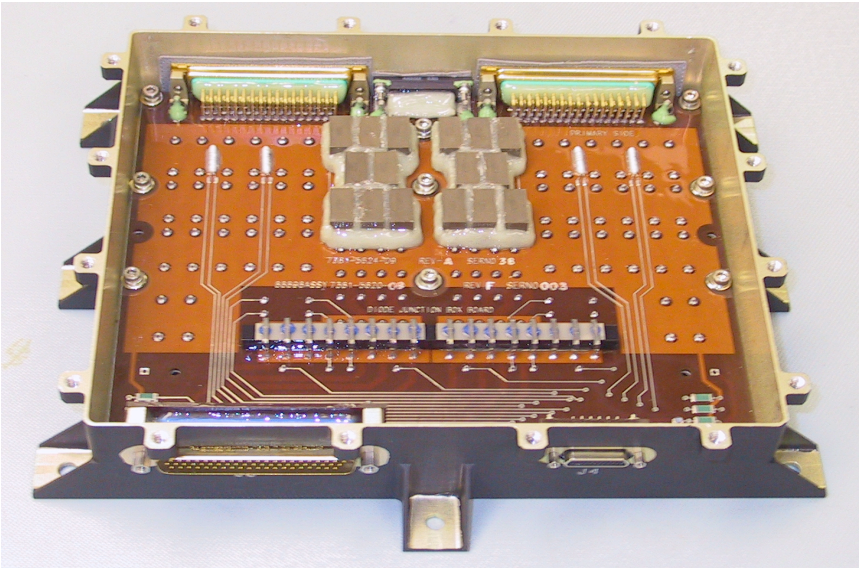
Reference Design
Power System Electronics (PSE)

- Function: solar array peak power tracking and battery charge control
- Upgrade of MESSENGER and STEREO PSE design
- Features
  - Redundant command/telemetry interfaces and housekeeping power supplies
  - Direct command/telemetry serial digital interface with avionics
  - Solar array power processing (buck converter) 6-for-5 module redundancy
  - Average output power per module: 92 W (existing design)
    - Goal is to increase output capability
  - Battery charge current and voltage limits
  - Battery cell balancing and bypass control
- Mass, Size, Power
  - Mass 13 kg (CBE)
  - Size 51 cm x 24 cm x 15 cm (LxWxH)
  - Power consumption 16 W (CBE)
    - Used in load power budget
  - Power throughput loss 40 W max
    - Accounted for in S/A sizing
  - Power dissipation 53 W max
    - Used for thermal analysis

STEREO PSE

Reference Design
Solar Array Junction Box (SAJB)

- Receives power from solar array; feeds power to PSE
- Quantity: 2 SAJBs
  - Accommodates power from primary and secondary sections (on 2 wings)
- Upgrade of MESSENGER and STEREO SAJB design
  - Larger string isolation diodes
  - Accommodate higher string currents from secondary S/A sections
- Features
  - Fault-tolerant design
  - Solar array string isolation diodes
  - Solar array bussing
  - S/A current monitoring
  - Buck converter input fuses
- Mass, Size, Power (each SAJB)
  - Mass 1.2 kg (CBE)
  - Size 26 cm x 21 cm x 5 cm (LxWxH)
  - Power dissipation 4 W max

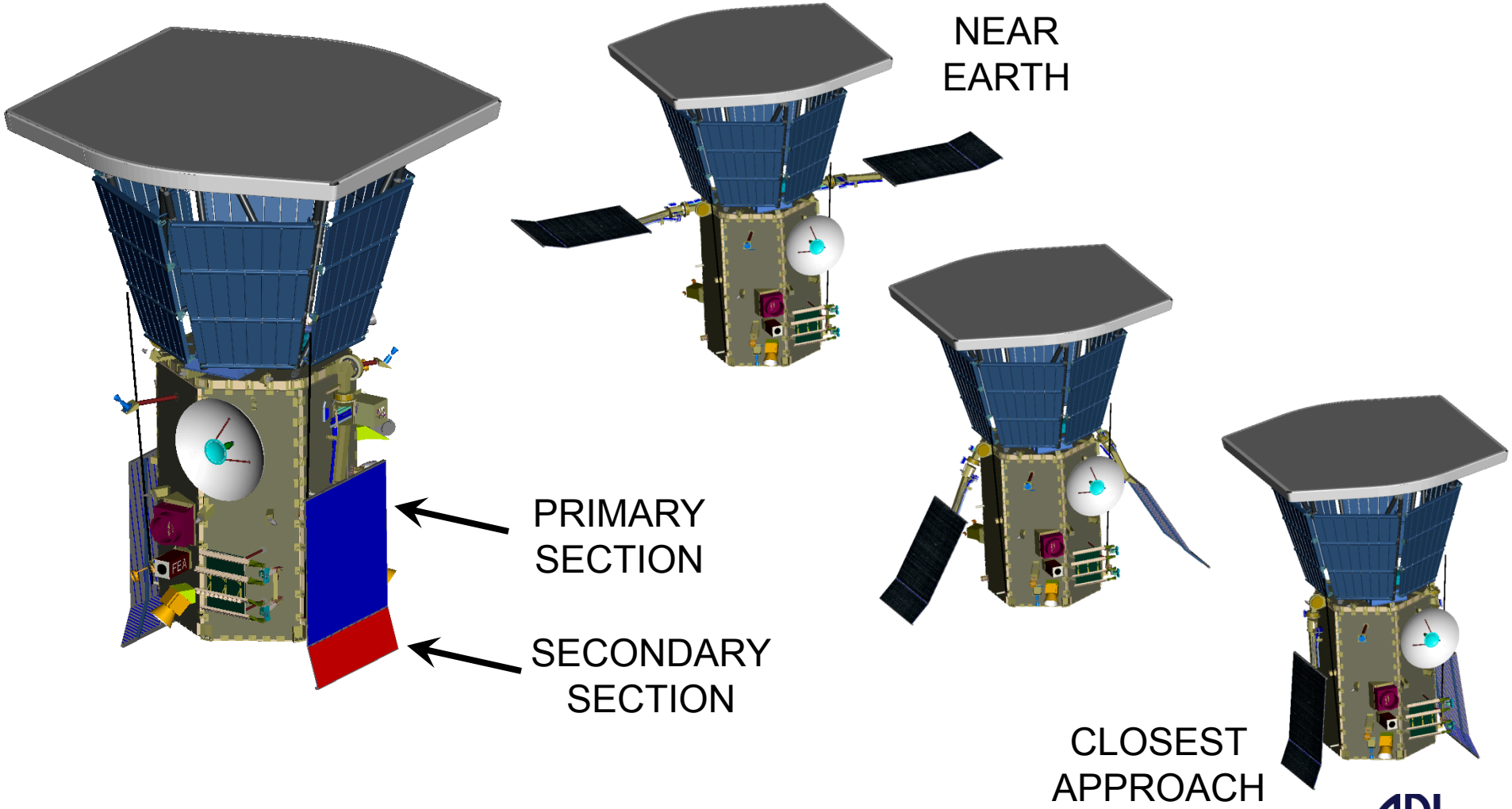


STEREO SAJB

Reference Design
Battery

- 20 Ah nameplate capacity battery supports:
  - Launch
  - Post-launch radiator heating
  - Venus gravity assist eclipse
  - Off-pointing during maneuvers
  - Peak load during communications
  - Short-duration power demands during solar array angle adjustments
  - Fuse clearing
- Features
  - Eight lithium-ion cells in series
  - Individual cell voltage monitoring
  - Cell bypass switches
  - Fault-tolerant design
- Energy margin
  - Maximum depth of discharge (DOD) is approx 75% following launch
- Mass, Size, Power Dissipation
  - Mass 15 kg (CBE)
  - Size 32 cm x 27 cm x 19 cm (LxWxH)
  - Power dissipation typical < 6 W
  - Power dissipation peak 20 W max during discharge

Reference Design
Solar Array



Reference Design
Solar Array

Solar Array Design Drivers and Approach

- **Primary S/A section drives total S/A size**
  - **Optimize for 1-AU operation (1 sun intensity)**
  - **Use standard solar cell design**
  - **Qualify for SPP environments**

- **Secondary S/A section drives cooling system size**
  - **Optimize for 9 Rs operation (40 suns maximum operational intensity)**
  - **Maximize ratio of electrical output power to absorbed thermal power**
  - **Design to thermally accommodate 80-sun survival case**
  - **Design solar cell to accommodate high current and radiation**
  - **Design solar cell laydown stack to minimize thermal gradients**
  - **Qualify for SPP environments**

Reference Design
Solar Array -- Primary Section

- Standard space solar cells (optimized for 1 sun intensity)
- Bypass diode located in cell corner crop area (same as MESSENGER)
- No OSRs (due to use of active panel cooling, unlike MESSENGER)
- Cell packing factor: 86% (ratio of cell active area to total panel area

Reference Design
Solar Array -- Secondary Section

✓ = Desirable for SPP
✗ = Not applicable for SPP

- Typically, space solar cells are optimized for…
  - Space (air-mass-zero) solar spectrum ✓
  - Charged particle radiation ✓
  - Only one sun intensity ✗
- Typically, terrestrial concentrator photovoltaic cells are optimized for…
  - Earth-surface (air-mass-1.5) solar spectrum ✗
  - No charged particle radiation ✗
  - High intensity ✓
- SPP cell performance must be optimized for…
  - Space (air-mass-zero) solar spectrum ✓
  - Charged particle radiation ✓
  - High intensity (40 suns, maximum operational) ✓
- SPP solar cell design…
  - Is a hybrid between space solar cell and terrestrial concentrator cell
  - Uses same materials (GaInP/GaAs/Ge) as space qualified solar cells
  - Uses enhanced metallization to accommodate high current

Reference Design
Solar Array -- Secondary Section

- Solar cell laydown approach: two options being pursued
  - Standard space cell stack-up with thermally enhanced cell bonding adhesive
    - Initial adhesive testing has been performed successfully
  - Modified terrestrial concentrator solar cell assemblies
    - Thermal cycling/shock tests have been performed
    - Pull tests and x-rays indicate no loss of adhesion or other anomalies

- Low-profile bypass diodes (next to cell) covered with reflective layer (OSR)
  - Existing space diodes have been tested to equivalent survival-level current

- Coverglass and adhesive have space flight heritage
  - Ceria-stabilized borosilicate microsheet (CMG) 0.5 mm (20 mils) thick
  - Coverglass adhesive: DC 93-500 (or equivalent)

- Cell size and layout
  - One electrical string of cells per row on panel (due to intensity gradients)

S/A Power Analysis

- Power analysis uses inputs from thermal analysis
  - Includes sun intensity and shading analysis

- For each value of sun distance, inputs include:
  - S/A tilt angle
  - Equivalent sun intensity for each string of cells
  - Maximum cell temperatures for primary and secondary sections

- Total S/A power is calculated for each sun distance
  - Power from each string is calculated and summed
  - Estimated power is referenced to end of life conditions

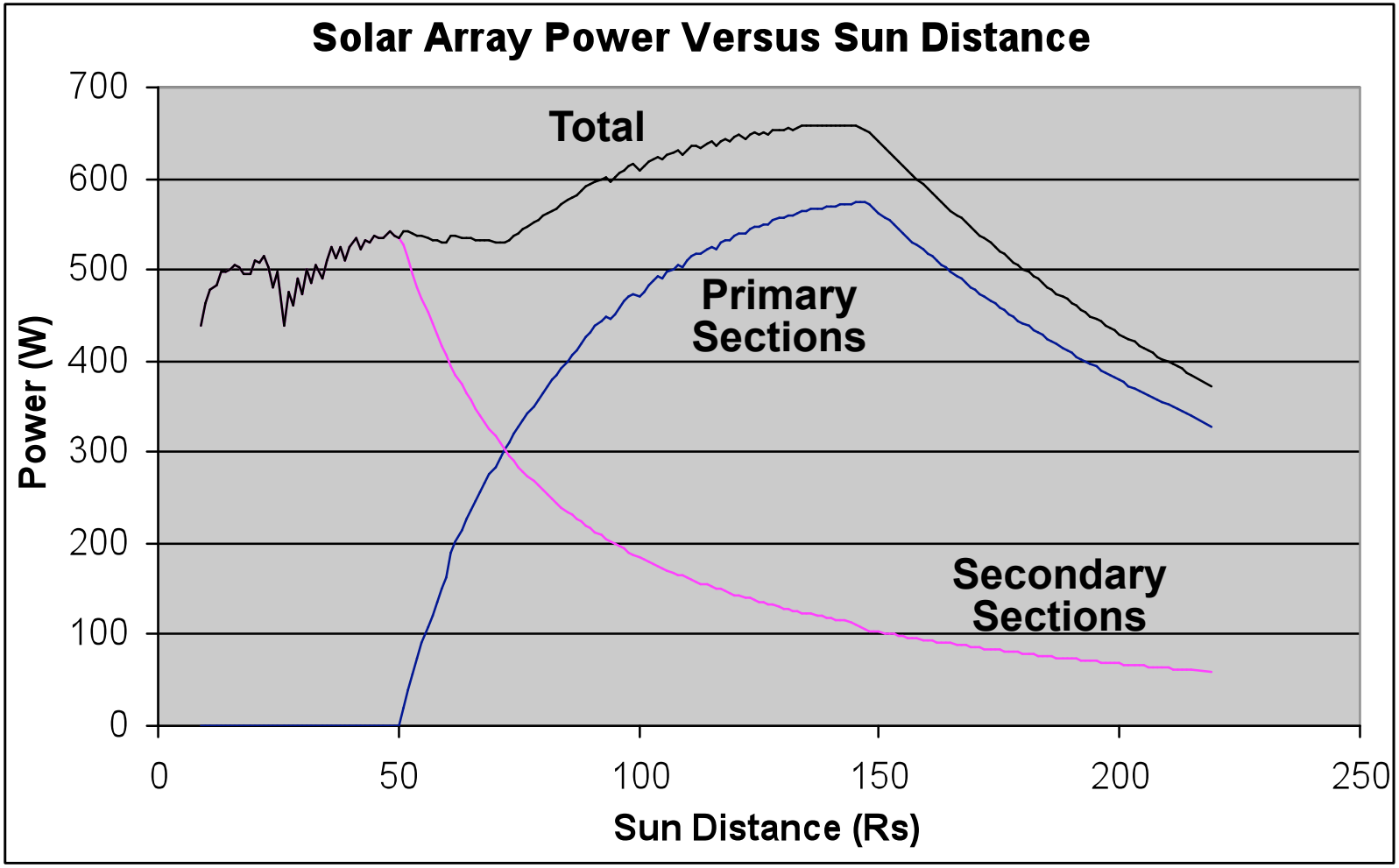
- Results of abnormal conditions are presented in Flight System section
  - Shadow skew (where shadow line is not parallel to strings of cells)
  - String failure

S/A Power Analysis

Degradation and other power loss factors considered in the analysis:

- **SPP-Unique:**
  - **Obscuration due to enhanced cell gridlines (optimized for 40 suns)**
  - **Long term semiconductor instability at temperature**
  - **S/A primary/secondary section voltage mismatch**
- **Typical:**
  - **Assembly voltage and current mismatch loss**
  - **Coverglass loss and optical mismatch**
  - **Measurement uncertainty**
  - **Ultraviolet radiation optical degradation**
  - **Charged particle radiation degradation**
  - **Micrometeoroids / dust and contamination**
  - **Intensity and temperature effects**
  - **Incidence angle effects (cosine, Fresnel reflection, other)**
  - **Power electronics efficiency and wiring/diode loss**
  - **Peak power tracking accuracy**
  - **Power distribution loss**

Solar Array Power Output



Feasibility – Solar Array Intensity and Temperature

- Terrestrial concentrator photovoltaic cells tested to > 700 equivalent suns
  - Optimized for 200 to 500 equivalent suns (SPP survival case is 80 suns)

- Solar array platen temperature range and swing from minimum to maximum
  - SPP: -70°C to 150°C operational, 200°C survival (270°C swing)
  - Typical GEO limits: -180°C to 150°C (330°C swing)

- Thermal gradient from coverglass to solar array cooling platen
  - Analysis results range from 14°C to 38°C for 80-sun survival case
  - Results depend on solar cell assembly design and materials
  - Based on initial test results of enhanced thermal conductivity adhesives

Feasibility – Solar Array Temperature (Continued)

- Solar cell electrical temperature coefficient – included in power predictions
  - Approx. 0.3% decrease in power per °C increase in temperature

- Metal contact diffusion into semiconductor junction (long term at high temp)
  - Vendor test data for bare "space" cells indicates less than 2% power loss
  - Calculated for 7 years at 150°C, based on accelerated life tests at > 200°C
  - GRC tested bare terrestrial cells, 227°C, 2500 hours; minimal degradation

- Coverglass adhesive, DC 93-500, rated for 200°C, tested to higher temperature

- MESSENGER performed testing on cells and related materials up to 270°C

Feasibility – Solar Array Analyses Performed

Pre-Phase A solar array preliminary analyses performed

- **Solar panel intensity distribution (from Thermal Engineering)**
- **Electrical effects of non-uniform intensity**
  - **Penumbral shading (by design) and shadow skew (unintended)**
- **Angle of incidence effects**
  - **Used worst combination of test data for glassed space and terrestrial cells**
- **Effects of voltage mismatch between primary and secondary sections**
- **Thermal gradient through solar cell laydown stack**
- **Radiation fluence, shielding, and solar cell degradation**
- **Solar cell metallization optimization and performance predictions**
- **Solar cell layout optimization and packing factor calculations**
- **Solar array power analysis including degradation factors**
- **Solar array materials assessment for SPP environments**

Feasibility – Power Electronics

- No new technology required for power electronics
  - Engineering development only
- PSE output power is less than for MESSENGER
  - Lower load power and no repetitive eclipses
- Max PSE input voltage (= S/A voltage) is less than for MESSENGER
  - Active temperature control of panels reduces cell voltage variation
- Peak power tracking algorithm will be upgraded
  - To avoid tracking on a local maximum that is not global maximum
  - Primary and secondary sections have different Vmp
  - Vmp = voltage at peak power point
- Solar array junction box will be modified
  - Will use blocking diodes rated for higher current than MESSENGER
  - Such diodes are readily available in larger package size
- Box-level temperature and radiation requirements are not unusual

Feasibility – Battery

- Battery temperature will typically be maintained 10°C to 30°C
- Relatively few discharges of significant depth of discharge (DOD)
- May be periods when battery is subjected to many low-DOD cycles
  - If solar array angle is dithering about optimum control point
  - Near closest approach if power and thermal margins become low
- Such requirements are not unusually stressing for Li-ion cells
- Battery will be tested for high-DOD and low-DOD cycling conditions

Technology Development
Secondary Solar Array – Pre-Phase A

Pre-Phase A tests performed- - demonstrates feasibility, reduces risk

- **Space solar cell samples tested successfully to > 160 suns**
- **Bypass diodes tested successfully to max expected string current**
- **High-temperature long-term exposure of bare cells**
  - **NASA/GRC tested terrestrial cells, 227°C, 2500 hours**
  - **Reviewed previous cell vendor tests for space cells (>150°C)**
  - **Minimal degradation over time for SPP conditions**
- **Thermal cycling of sample terrestrial cell assemblies**
  - **Thermal shock/cycling testing (between LN2 and160°C)**
  - **Pull tests and radiographic inspection performed**
  - **No evidence of loss of adhesion or other anomalies**
- **Solar cell adhesive thermal conductivity testing**
  - **Validates thermal analysis**
- **Solar cell adhesive outgassing testing**
  - **Performed by APL on adhesive used with terrestrial concentrator cells**
  - **DC1-4173 passed ASTM E595 testing**

Technology Development
Secondary Solar Array – Pre-Phase A

Pre-Phase A testing (continued)

- **Prototype cell testing in process:**
  - **Cells designed and fabricated by two cell vendors**
  - **Different design approaches**
  - **NASA/GRC tested the prototype cells up to 40 suns and 150°C**
  - **Charged particle radiation exposure and cell characterization**

- **Testing planned:**
  - **Incidence angle testing of CICs (cell, interconnect, coverglass)**
    - **Including sensitivity to intensity**
  - **Prototype test panel coupons populated with cells (both vendors)**
    - **Thermal cycling of test panels over expected temperature range**
    - **Exposure to survival temperatures**

Technology Development
Secondary Solar Array – Phase A

Phase A testing advances solar array development maturity to TRL-5

- **Long duration, high-intensity, high temperature CIC testing**
- **Bypass diode high temperature, high current exposure**
  - **Diode I-V characterization**
  - **Accelerated life testing**
- **UV degradation at high flux rate in vacuum**
- **Acoustic and/or vibration testing of prototype panel**
- **Micrometeoroid/dust impact testing on CICs**
  - **Post-impact performance testing at high incidence angles**
- **Complete testing started in pre-Phase A, as required**

Technology Development
Secondary Solar Array – Phase B

Phase B testing advances solar array maturity to TRL-6 (by PDR)

- **High intensity, high temperature cell qualification tests**
- **Cell high-altitude flight calibration (if new epitaxial design)**
- **Radiation qualification tests (if new epitaxial design)**
- **High temperature outgassing and TML/CVCM testing as required**
  - **For any materials that have not already been tested**
  - **MESSENGER test environment envelopes SPP requirements**
- **Acoustic/vibration and shock testing of qualification panel**
- **Thermal vacuum testing of qualification panel**
- **Electrical and thermal performance verification of qualification panel**
  - **Operational intensity and temperature at maximum incidence angle**
- **Pre- and post-test electrical verification for all environmental tests**

Phase A Plans

- Generate subsystem requirements document
- Perform trade studies (see next slide)
- Update and refine power analyses
- Continue solar array prototype hardware development
- Complete testing to demonstrate TRL-5 for solar array
- Coordinate with NASA/GRC on solar array test planning
- Perform PSE conceptual design; breadboard circuits where required
- Develop peak power tracking algorithm conceptual design
- Generate preliminary equipment specifications
- Refine resource estimates: mass, power, size, command/telemetry
- Issue requests for proposals for procured items
- Develop cost, schedule, and staff plans for Phases B-E

Phase A Trade Studies

- Solar cell laydown configuration
  - Standard space laydown or terrestrial concentrator approach
  - Coverglass thickness and coating type
  - Cell bonding adhesive
- Peak power tracking and direct energy transfer architectures
  - Present baseline is peak power tracking
- Battery cell type and cell balancing
  - Baseline is lithium ion cells
  - Single string of “large” cells vs. multiple strings of “small” cells
- Power system electronics: local vs. remote digital control
  - Local control in processor or FPGA vs. reliance on S/C central processor
- Solar array junction board/box (SAJB) separate or within PSE
  - Present baseline is separate SAJBs

Solar Probe Plus
Mission Concept Review

Structure and Mechanisms

Ted Hartka
Structures and Mechanisms Lead

The Johns Hopkins University
APPLIED PHYSICS LABORATORY

Outline

- Working Requirements
  - Structure
  - Mechanisms
- Options Investigated
  - Structure
  - Mechanisms
- Reference Design
  - Structure
  - Mechanisms
  - Packaging
- Technology Assessment
- Phase A Plans

Structure Working Requirements

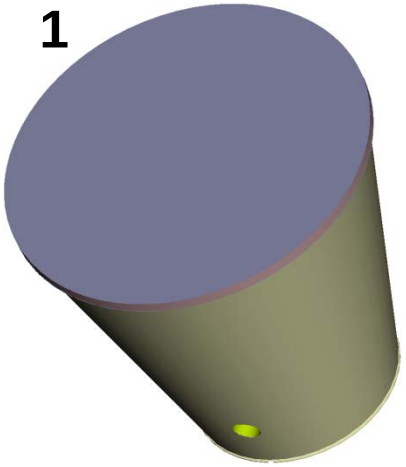
- Accommodate Subsystems and Instruments
- Minimize Size and Mass
- Size to Remain Protected Inside the Umbra
- Atlas V 551
  - Frequency
    - Lateral 8 Hz
    - Axial 15 Hz
  - Design Loads
    - Increased for low mass
    - Launch Vehicle: Axial 7 G's and Lateral 2.5 G's
    - 3rd Stage: Axial 10.6 G's
- Factors of Safety
  - Yield 1.25
  - Ultimate 1.5
  - Buckling 1.5
  - Composites 1.5

Mechanism Working Requirements

- Combined Array
  - 0°- 90° Elevation (flap) from stowed position
  - +/-90° Tilt (feather)
  - 0.01° Pointing Requirement
- High Gain Antenna
  - - 45° to + 45° from stowed position
  - Remain inside umbra in stowed position with margin
- General
  - GSFC-STD-7000, General Environmental Verification Standard
  - Torque Factor of Safety
    - 2x known torques
    - 4x variable torques
    - End of life (EOL) differences determined by life testing

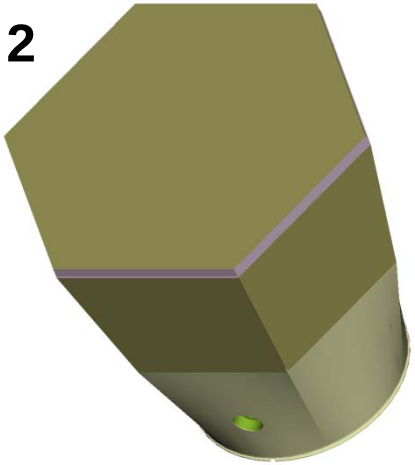
Options Investigated, Structure

1



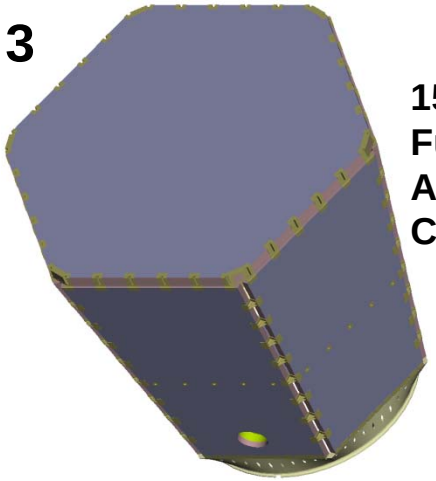
Machined Metal Cone
No panels
Angled (umbra hugger)
Clamp Band

2



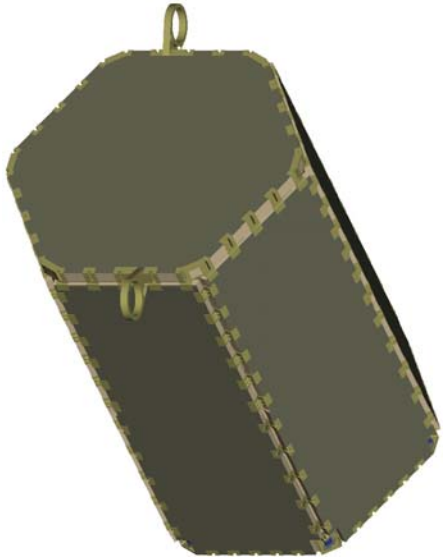
Half Height Round-Hex Adapter
Half Height Panels
Angled (umbra hugger)
Clamp Band

3



15 cm (6”) Round-Hex Adapter
Full Height Panels
Angled (umbra hugger)
Clamp Band

4



Full Height Panels
Right Hexagon
Separation Nuts

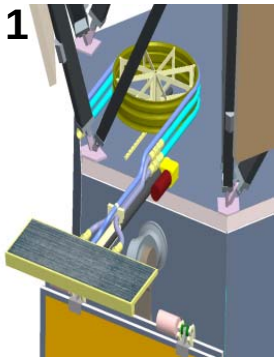
Face Sheet Options:
-Aluminum
-Composite
-AlBemet

Structure Vendor Interaction

- **APL extended an RFI to industry: Lockheed-Martin, Orbital Sciences Corp , ATK/Composite Optics, and Applied Aerospace Structures Corp**
- **All responsive vendors offered mass efficient solutions**
- **Solar Probe Plus requires an integrated, *on-site* design team that exercises control of the design process to ensure critical interfaces match up to technical needs, cost and schedule**
- **APL elected to procure bus structure *components* from industry (panels, fabrication services, etc.) and retain in house, the design, sub-system integration, test and assembly functions acting as the system integrator.**
- **The information obtained during the RFI process gives APL the opportunity to revisit the make/buy decision in Phase A should mass issues arise.**

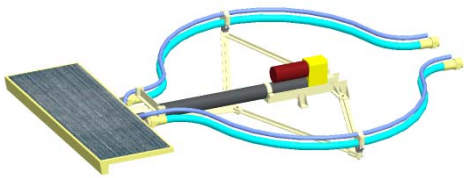
Options Investigated, Secondary Solar Array Mechanism Concepts

1



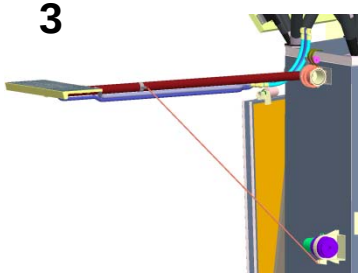
Linear Actuator
with Hose Wheel

2



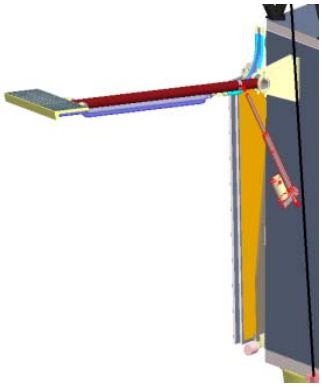
Linear Actuator
with Hose Linkage

3



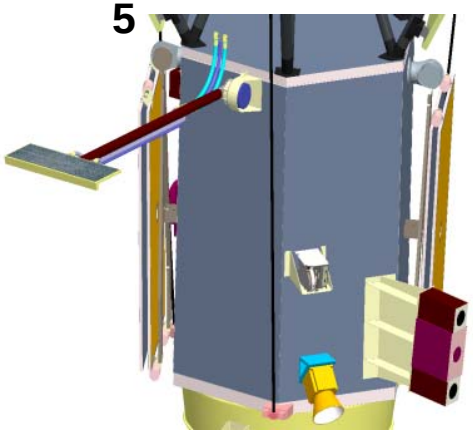
Constant Force Hinge with
Cable Drive & Flex Hose

4



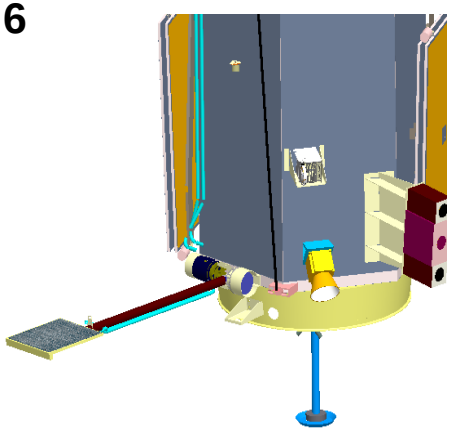
Linear Actuator
with Service Loop

5



Rotary Actuator
with Service Loop

6



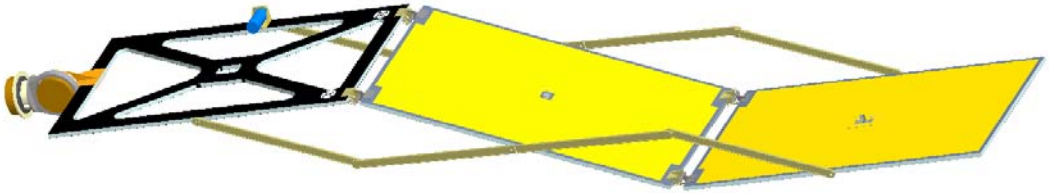
Rotary Actuator
with Rotary Union

7

Combined Array
Dual Gimbal with Service Loop

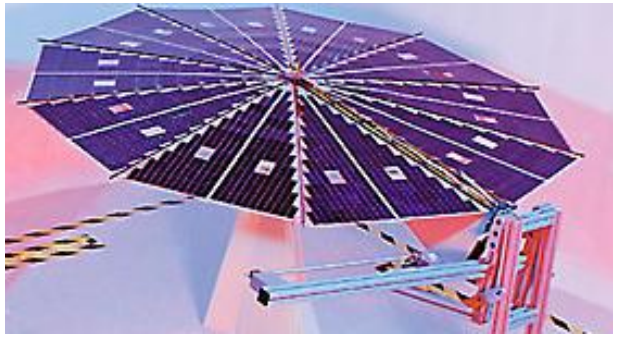
Options Investigated, Primary Solar Array Concepts

1



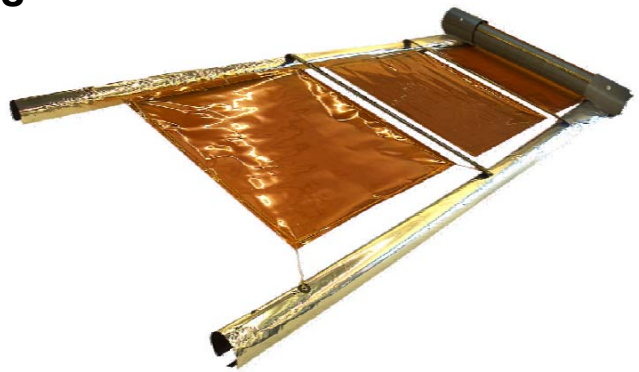
Biaxial Gimbals with Deployment Synchronization Linkage

2



Ultraflex Solar Array

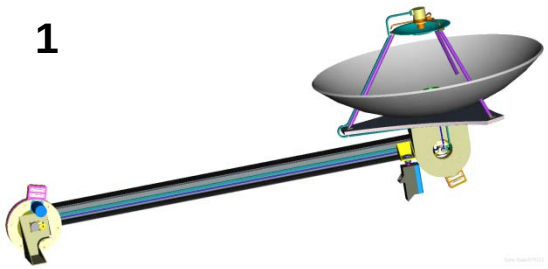
3



Roll-out and Passively Deployed Array (RAPDAR)

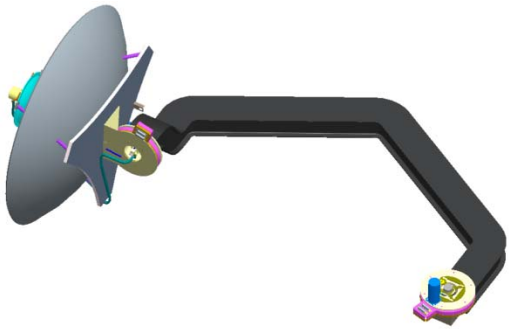
Options Investigated, High Gain Antenna Mechanism Concepts

1



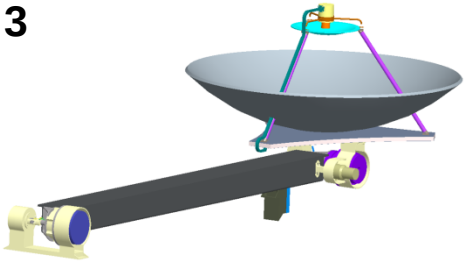
Straight Boom with Internal Rotary Joint
0.8 m (31.5”) Antenna

2



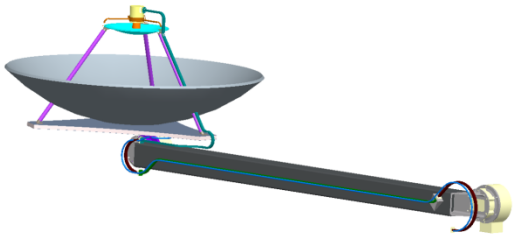
Internal Wave Guide with Curved Boom
0.8 m (31.5”) Antenna

3



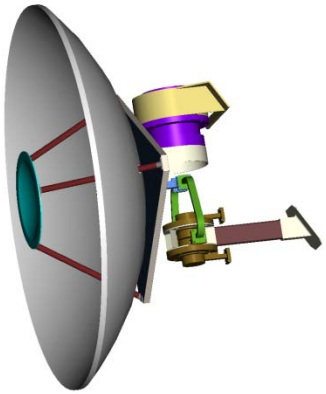
Straight Boom with External Rotary Joint
0.8 m (31.5”) Antenna

4



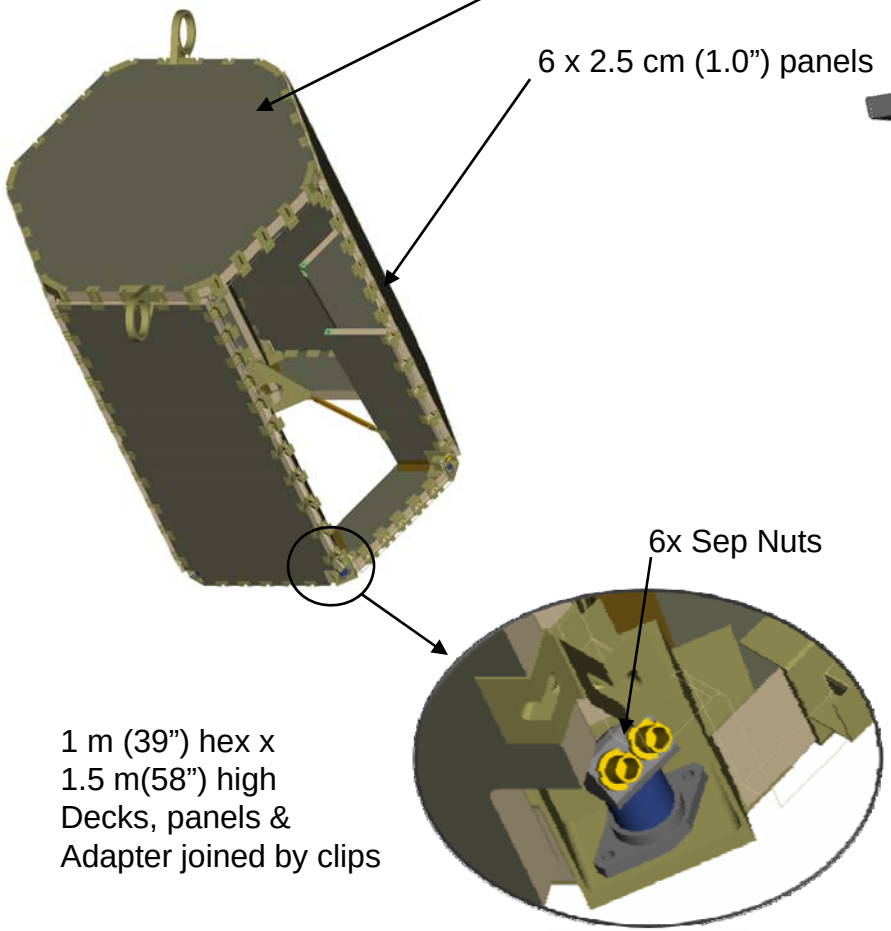
Rotary Actuator with Flexible Wave Guide
0.8 m (31.5”) Antenna

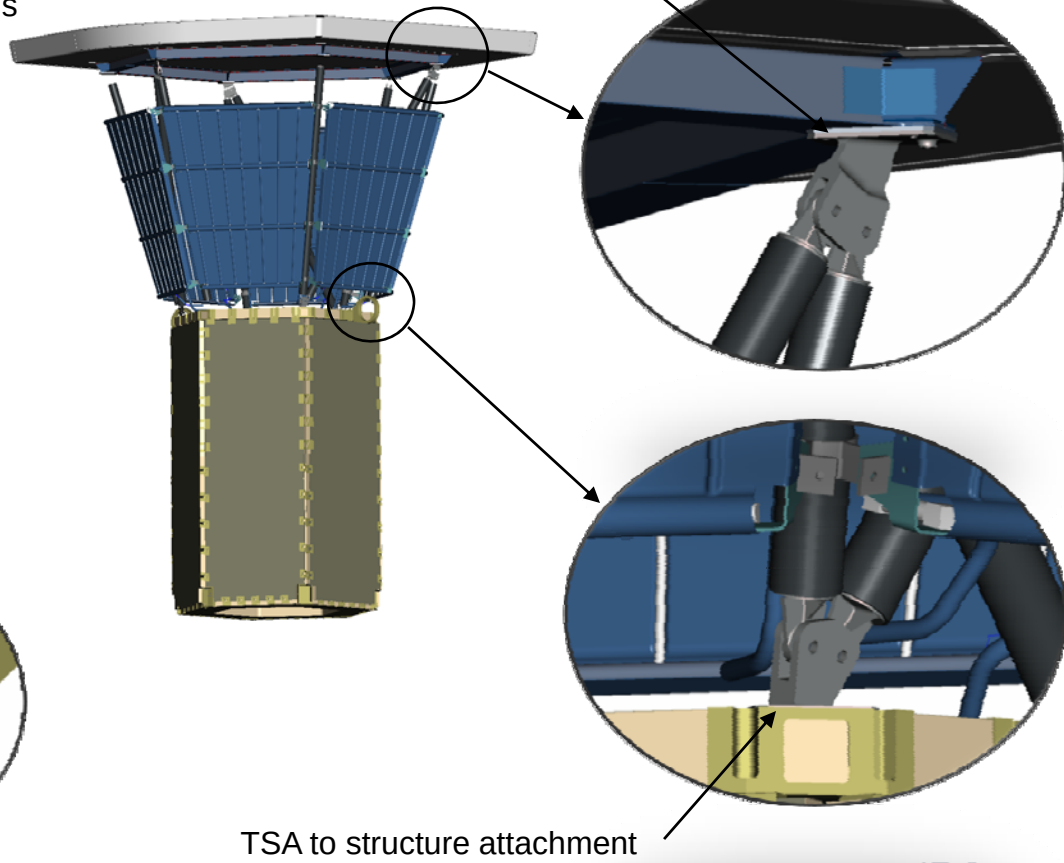
5



Body Mounted with Single Axis Gimbal
0.6 m (23.6”) Antenna

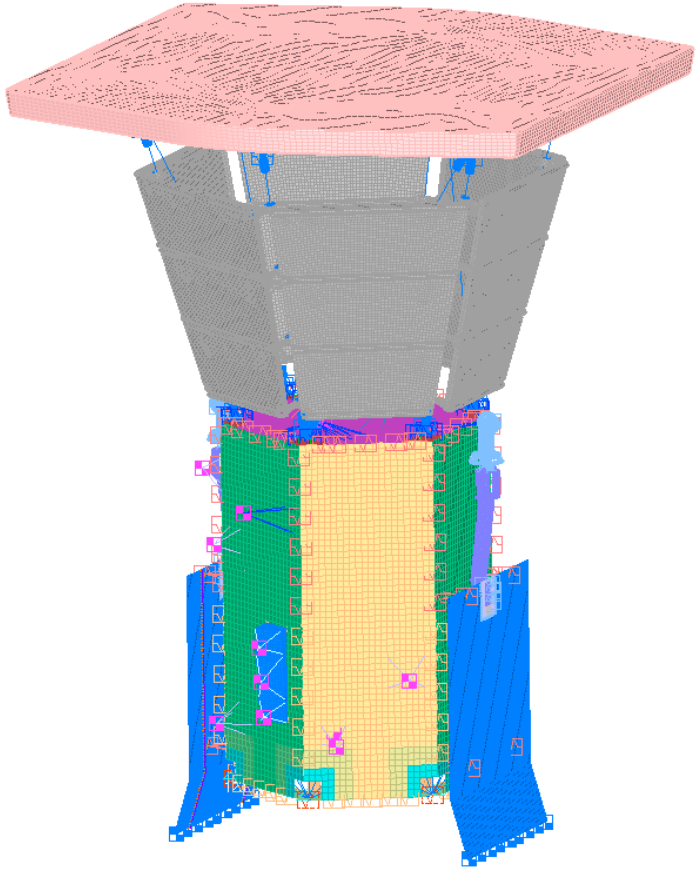
Reference Design, Structure



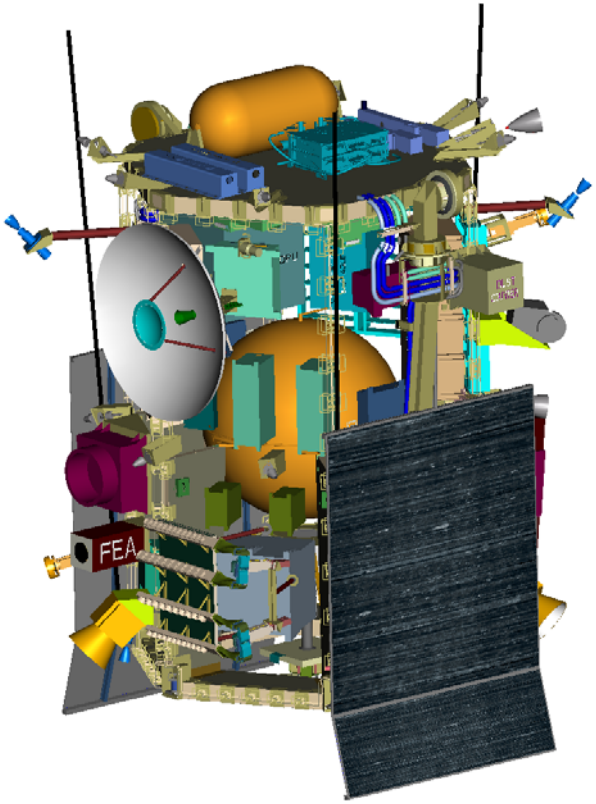


Reference Design Structure FEM

- Highly Detailed FEM
  - Discrete clips
  - Separation nuts
  - Solar arrays
  - TPS/TSA detailed model
  - Mass CBE + 30% (610 kg)
    - Boxes modeled as lumped mass
    - Remaining items smeared appropriately on panels

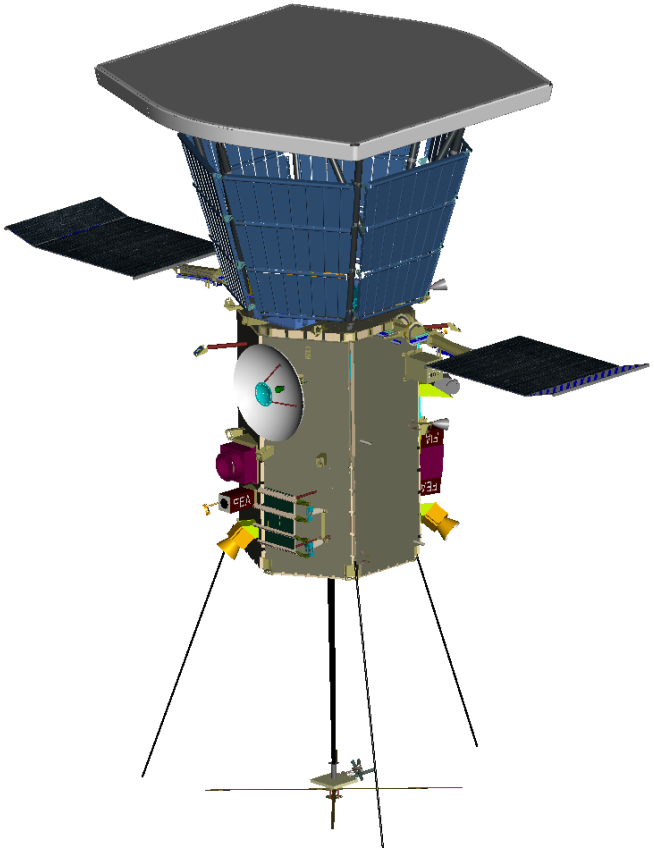


Reference Design, Packaged, In Fairing & Deployed Configuration



Side Panels Removed for Clarity

Atlas V 551



Reference Design, Combined Array

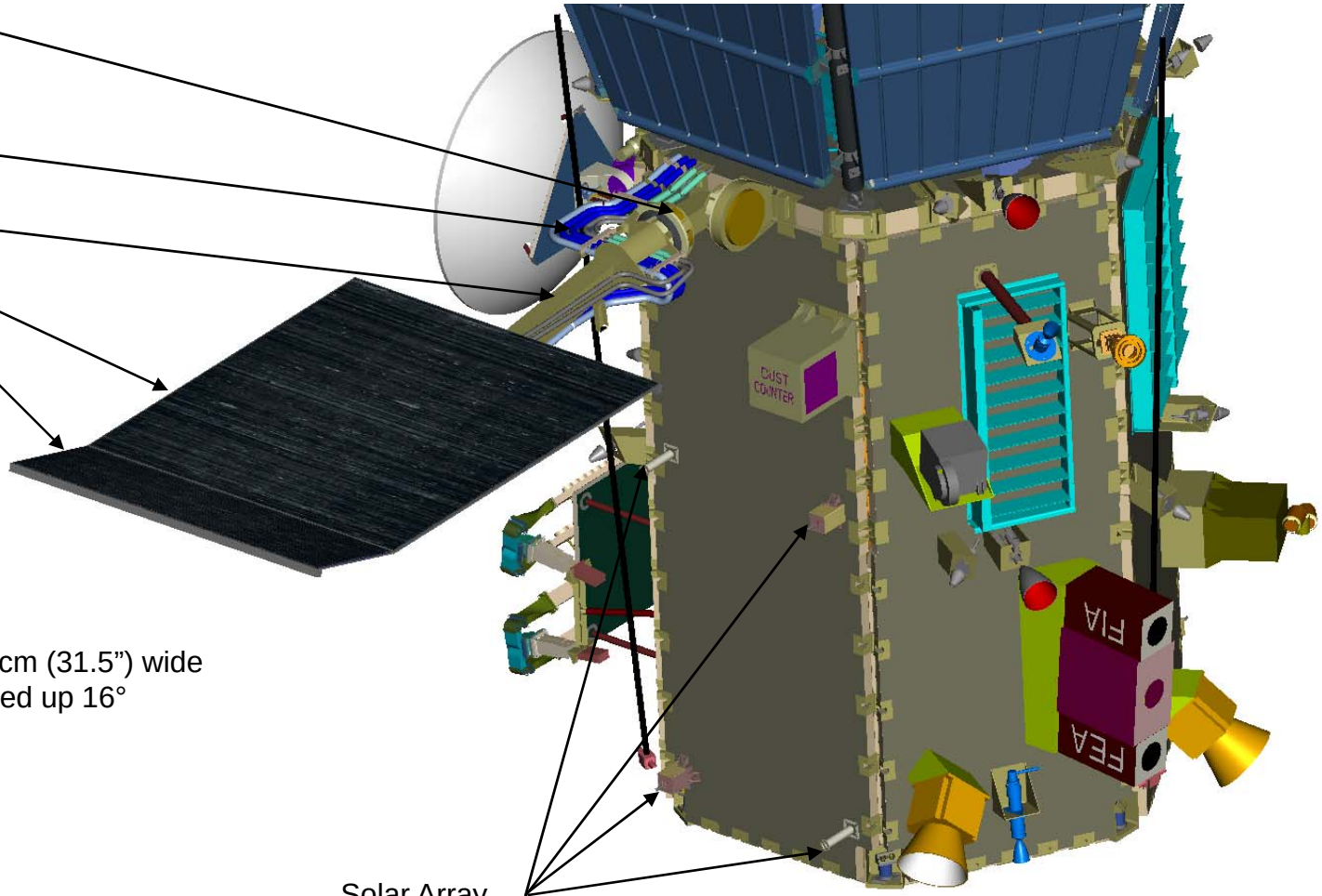
Dual Gimbal Drive
0°-90° Elevation (flap)
+/- 90 Tilt (feather)

Flexible Cooling Lines

Array Boom

Primary Solar Array

Secondary Solar Array



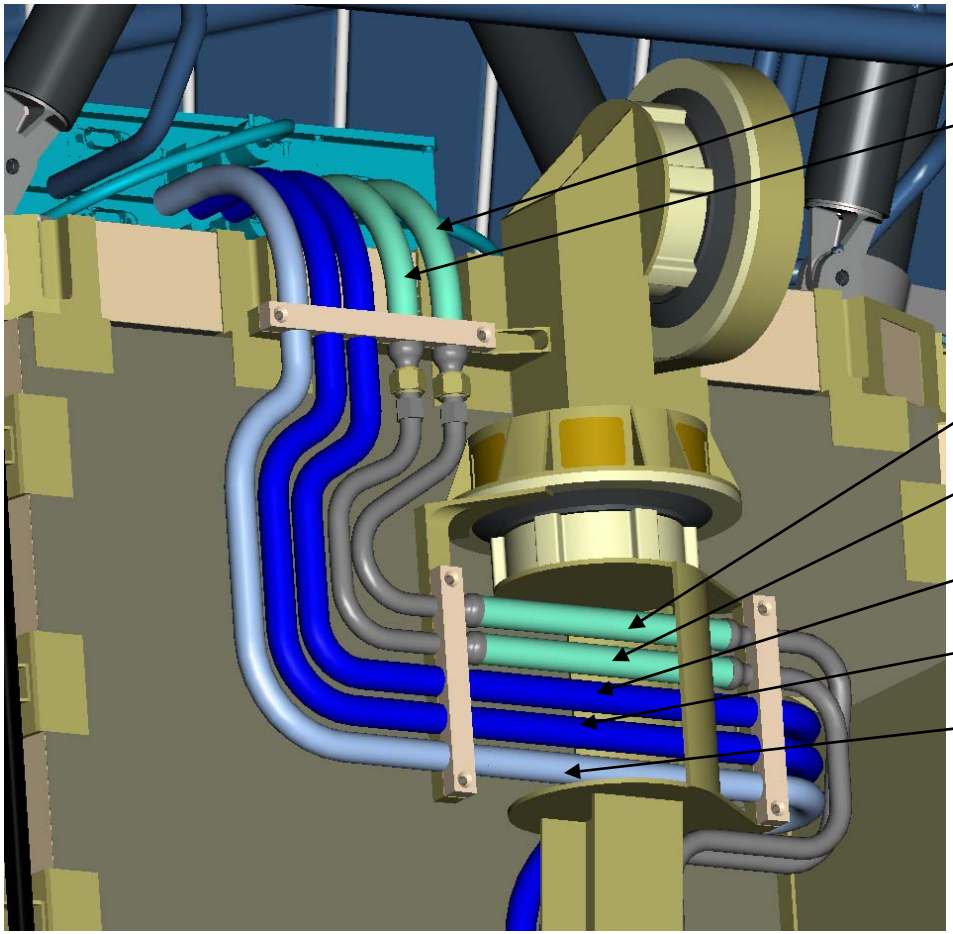
2 Identical Wings
1.1 m (44”) long x 80 cm (31.5”) wide
Secondary Array Tipped up 16°

Solar Array
Restraint/Release

Combined Array Cooling and Harness Routing

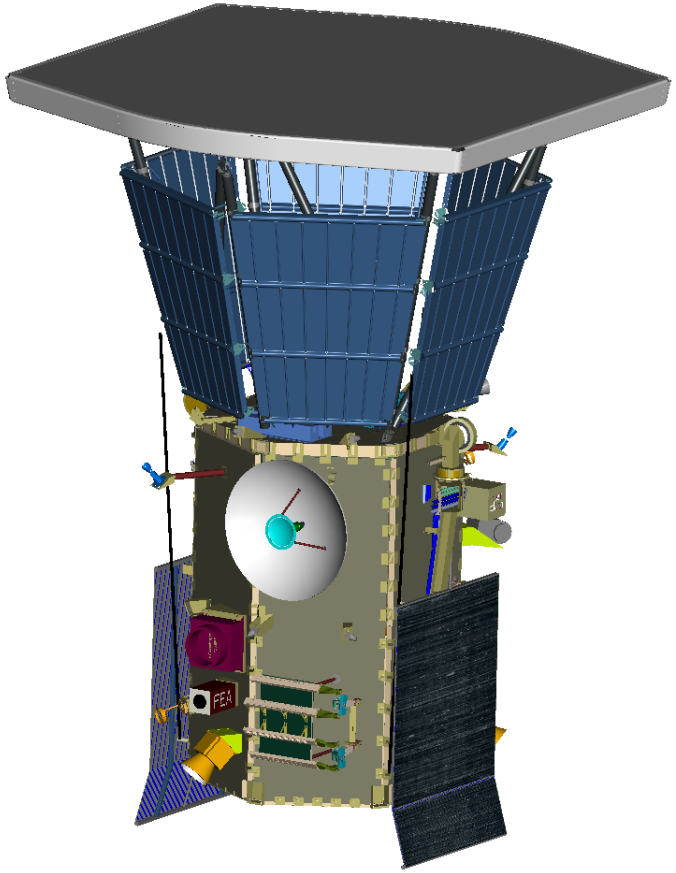
Flex Hose vendors under consideration
- Hosemaster
- Meggitt
- Titeflex
- U.S. Hose
- FMH Corporation

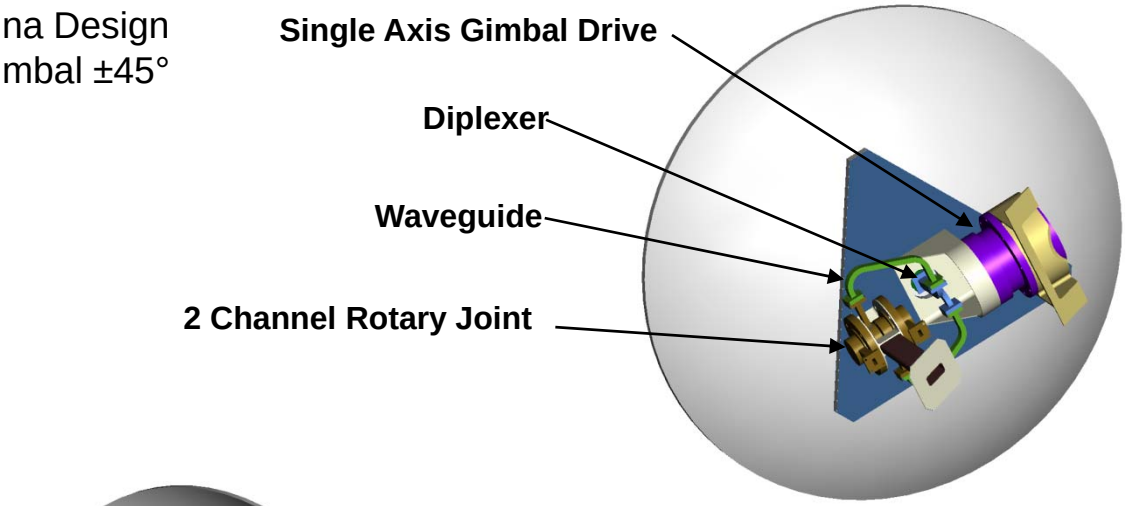
Reference hose
.375” Hosemaster
Extraflex,
Single braid

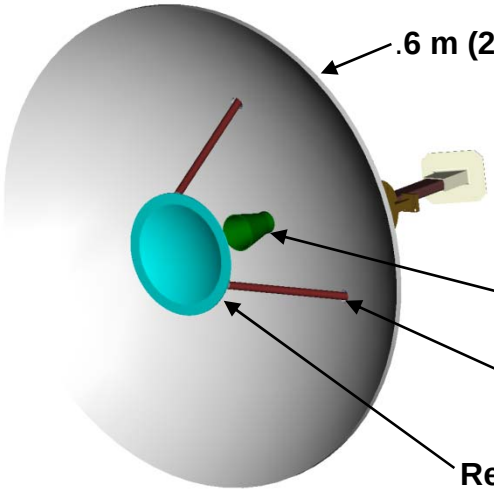


Reference Design High Gain Antenna

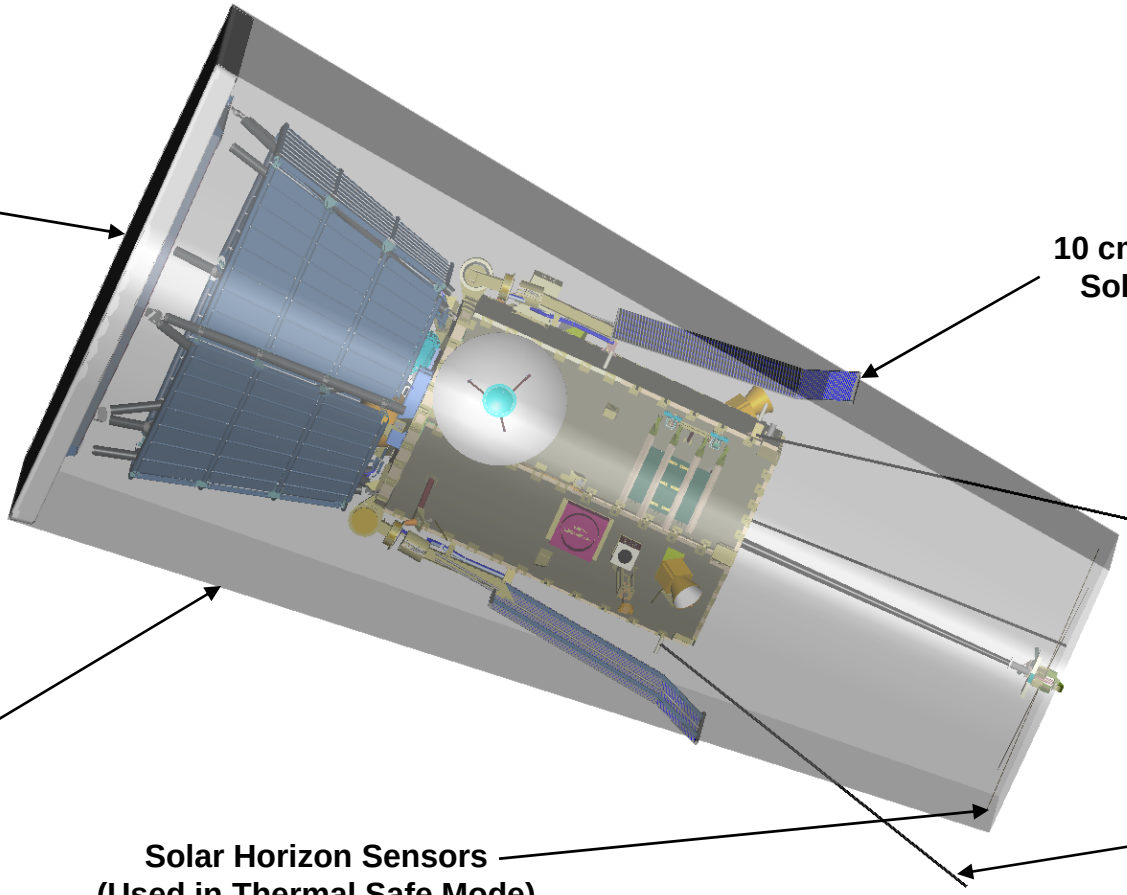
0.6 m (23.6”) Cassegrain Antenna Design
Operates at Ka-band, Single Gimbal ±45°



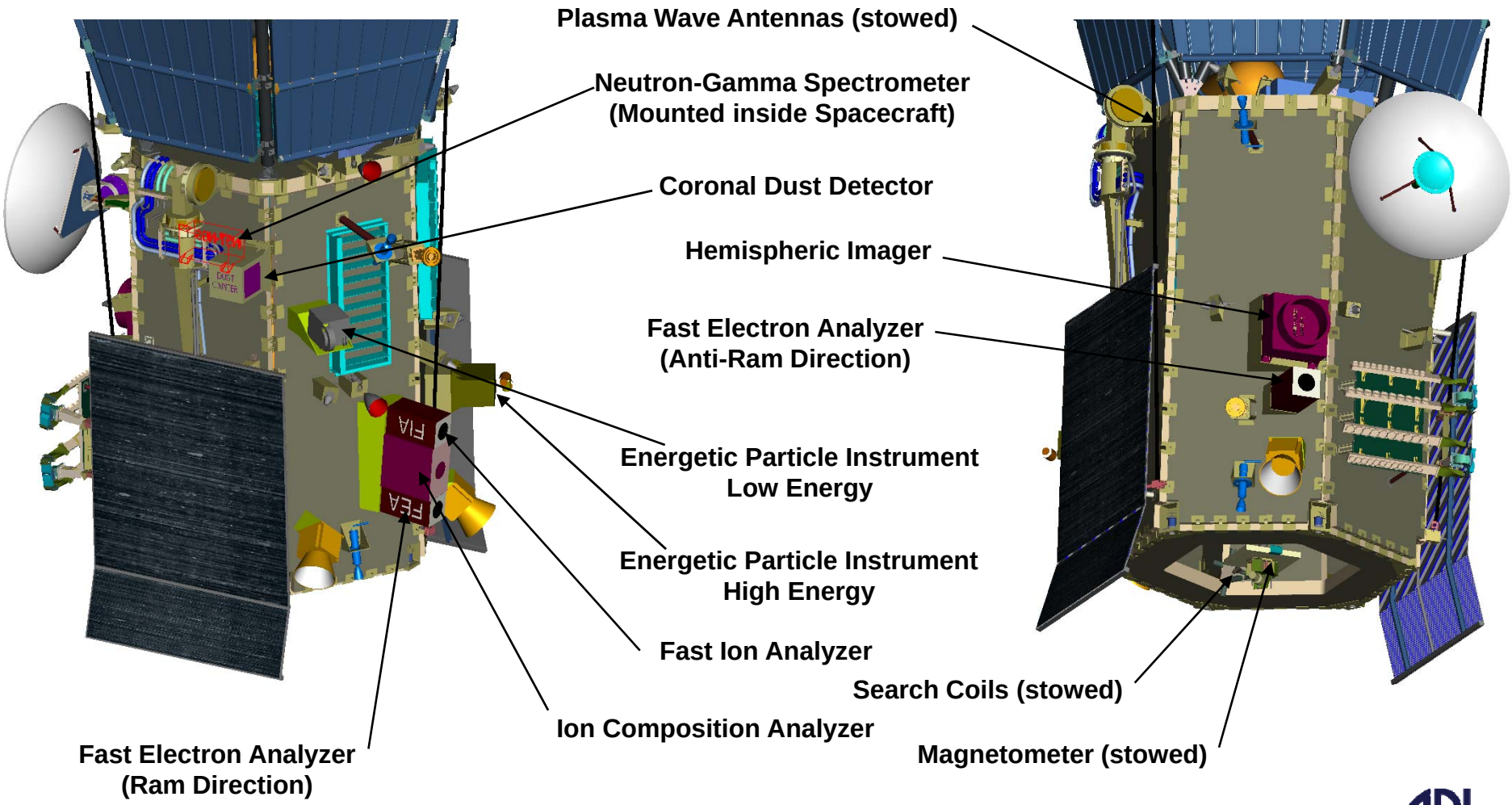




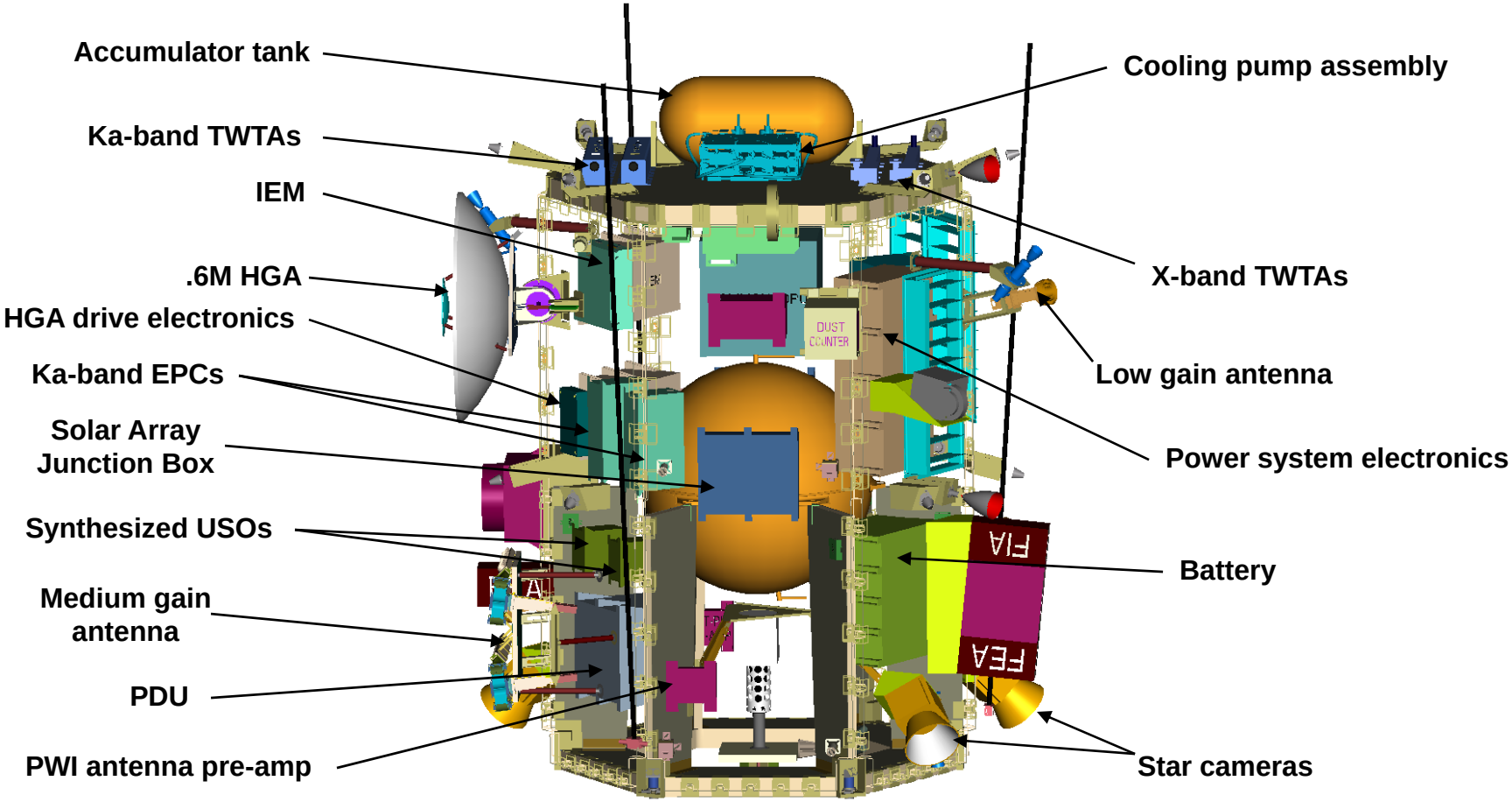
Reference Design, Closest Approach - Umbra



Reference Design, Instrument Packaging

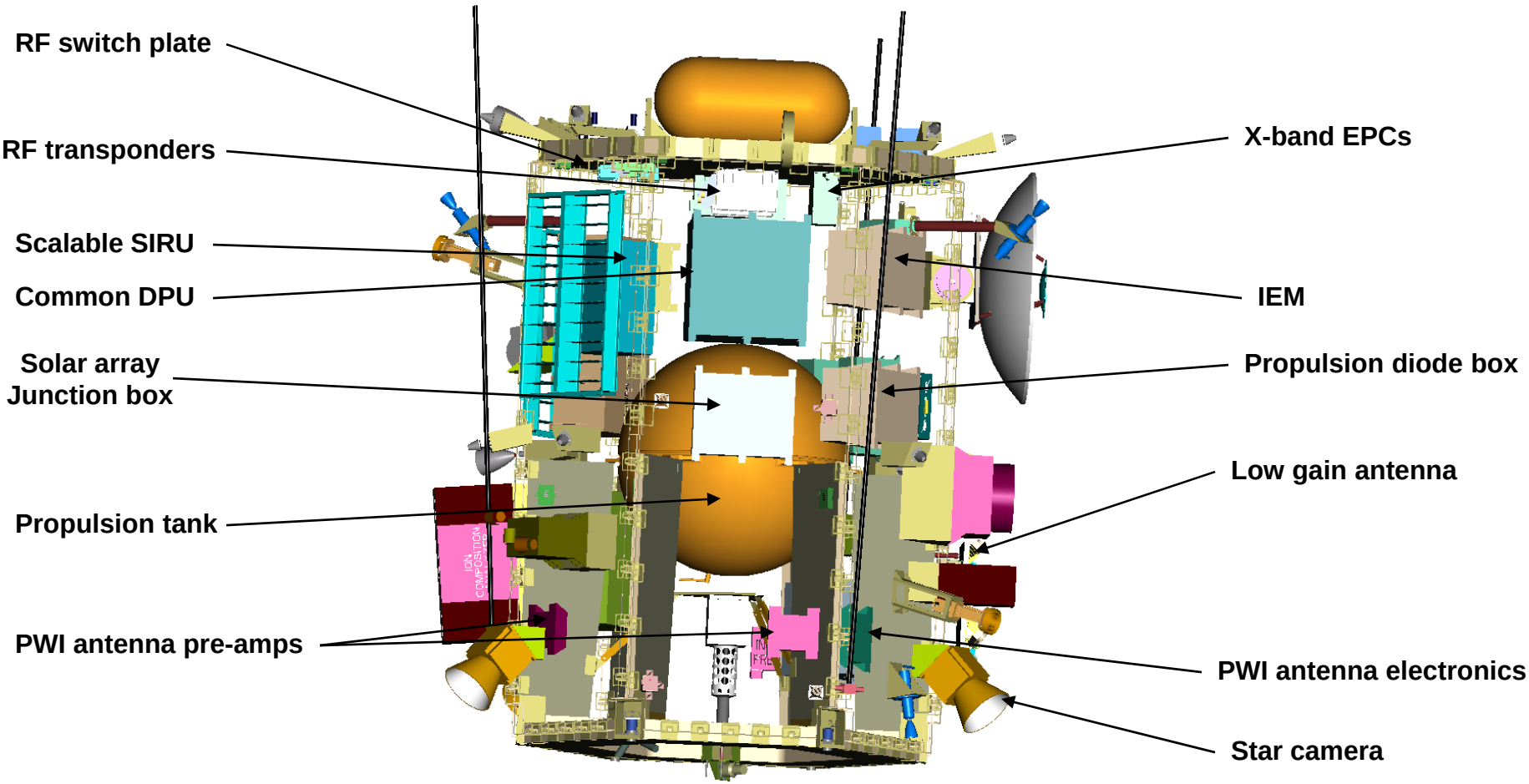


Reference Design Packaging (1 of 2)



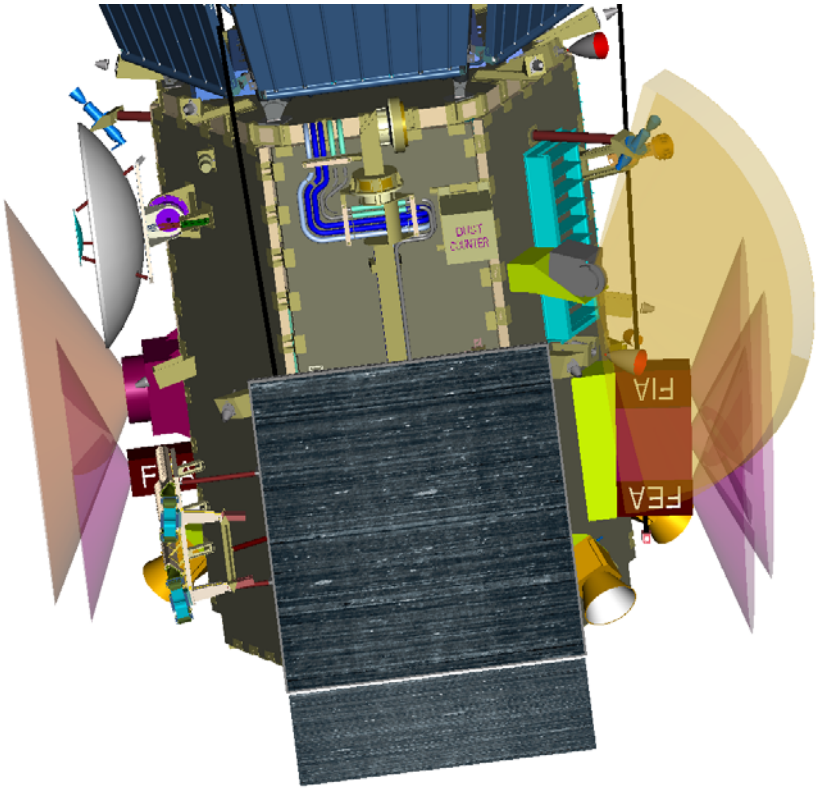
Side Panels Removed for Clarity

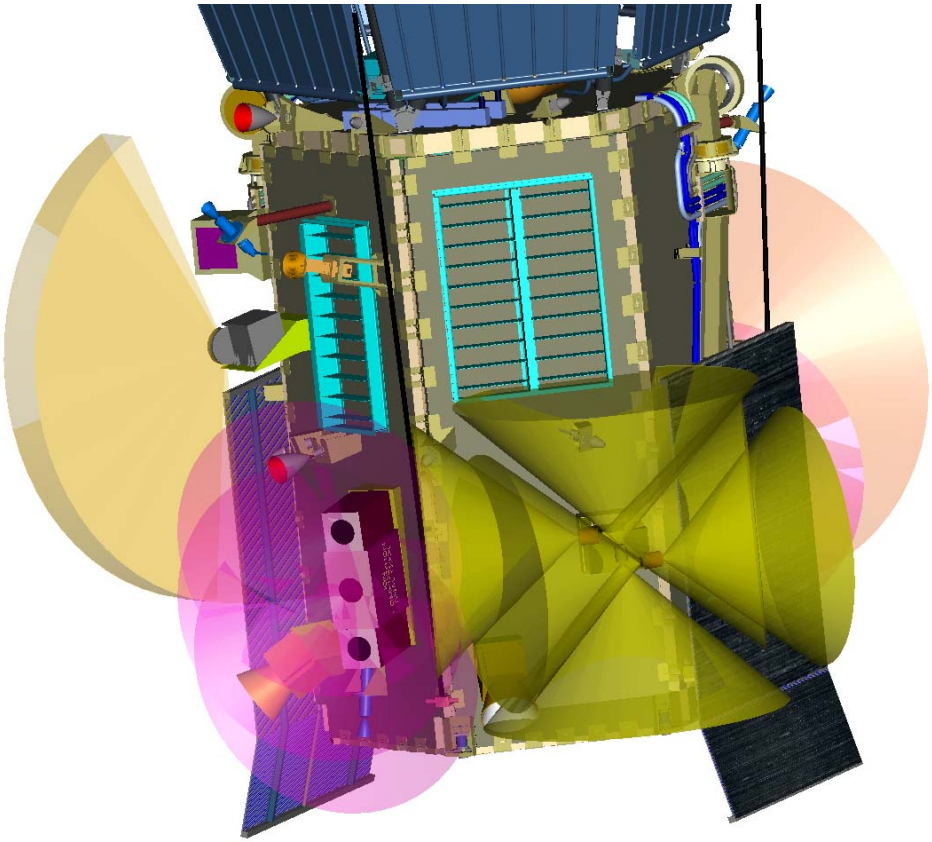
Reference Design Packaging (2 of 2)



Side Panels Removed for Clarity

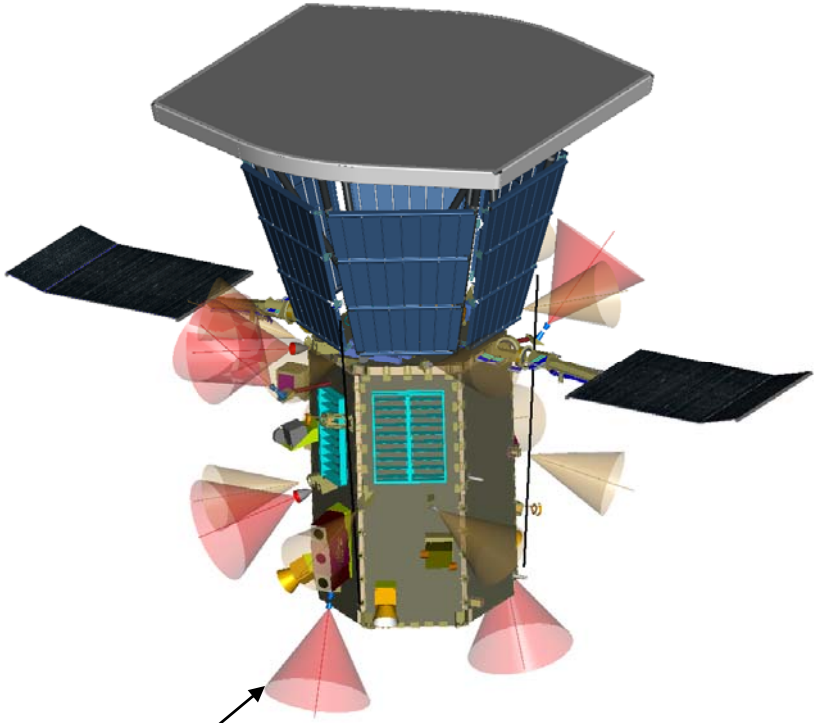
Reference Design, Instrument Fields of View

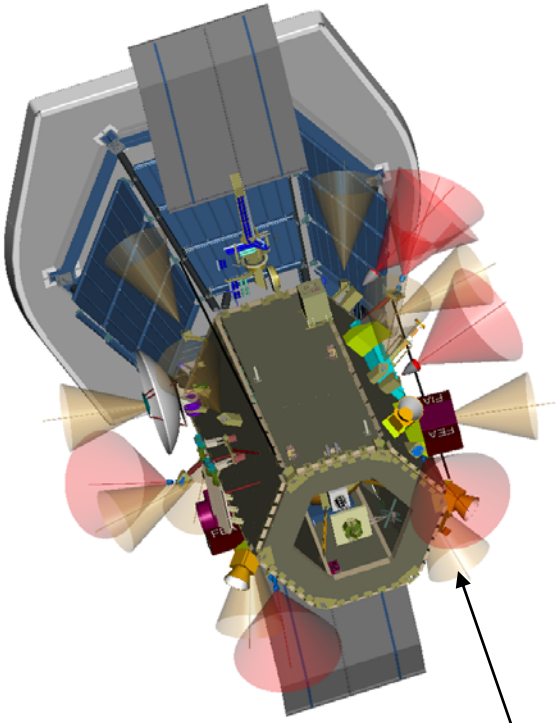




All Fields of Views Meet STDT Requirements

Reference Design, Thruster Plumes



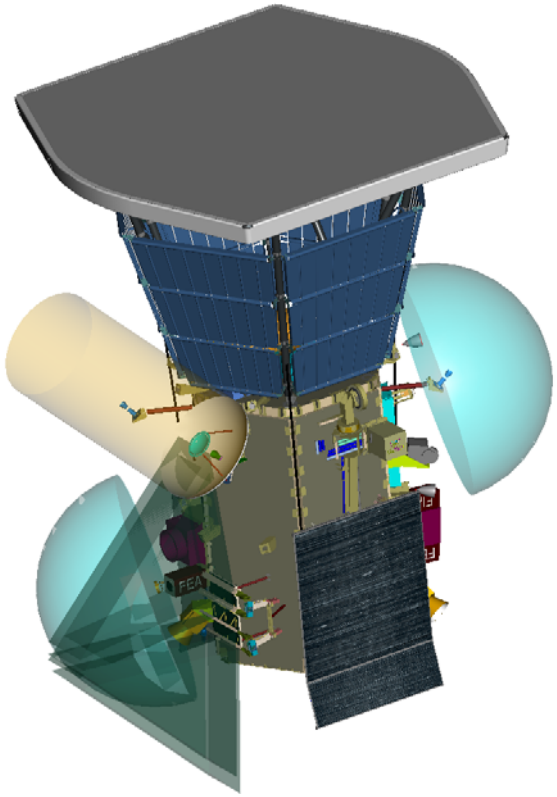


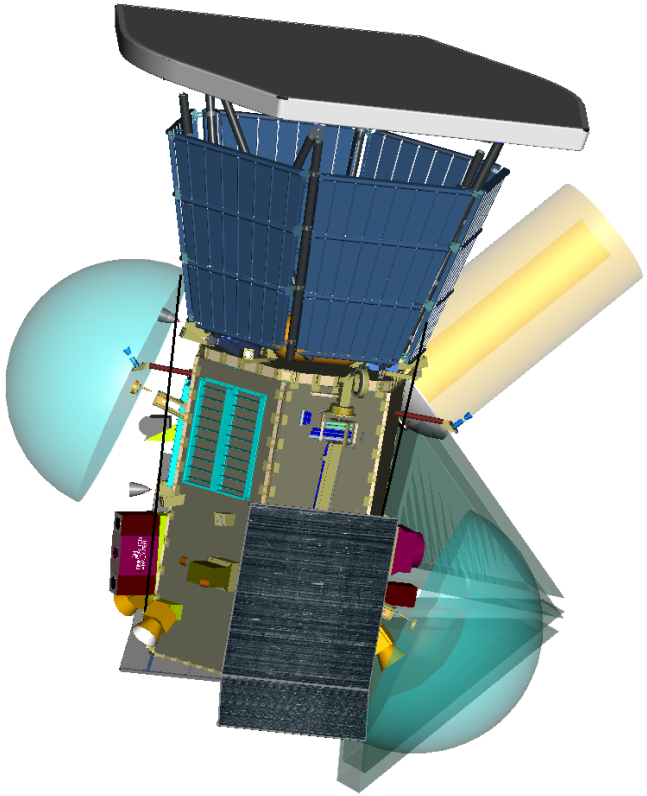
Thruster Plume Studies are Underway

30° half angle red plumes are shown for the MR111C (4N) and MR106E (22N) thrusters

20° half angle beige plumes are shown for the MR103H (1N)

Reference Design, RF Fields of View





All Antenna Fields of View Are Shown

- **HGA Field of View is Clear**
- **Fanbeam and Low Gain Fields of View have been Negotiated with RF**

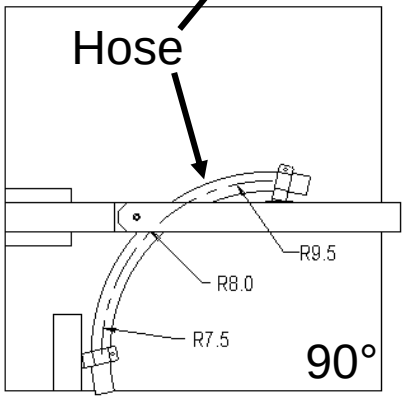
Technology Assessment

- Structure: Honeycomb Panels and Metal Clips
  - TRL 8
  - Similar to Previous APL Spacecraft: New Horizons, ORS, and RBSP
- High Gain Antenna Mechanism: Single Axis Gimbal
  - TRL 8
  - Similar to Previous APL Designs: STEREO
- Combined Array Mechanism: Dual Axis Gimbal
  - TRL 8
  - Similar to Previous APL Designs: MSX

Tests Completed

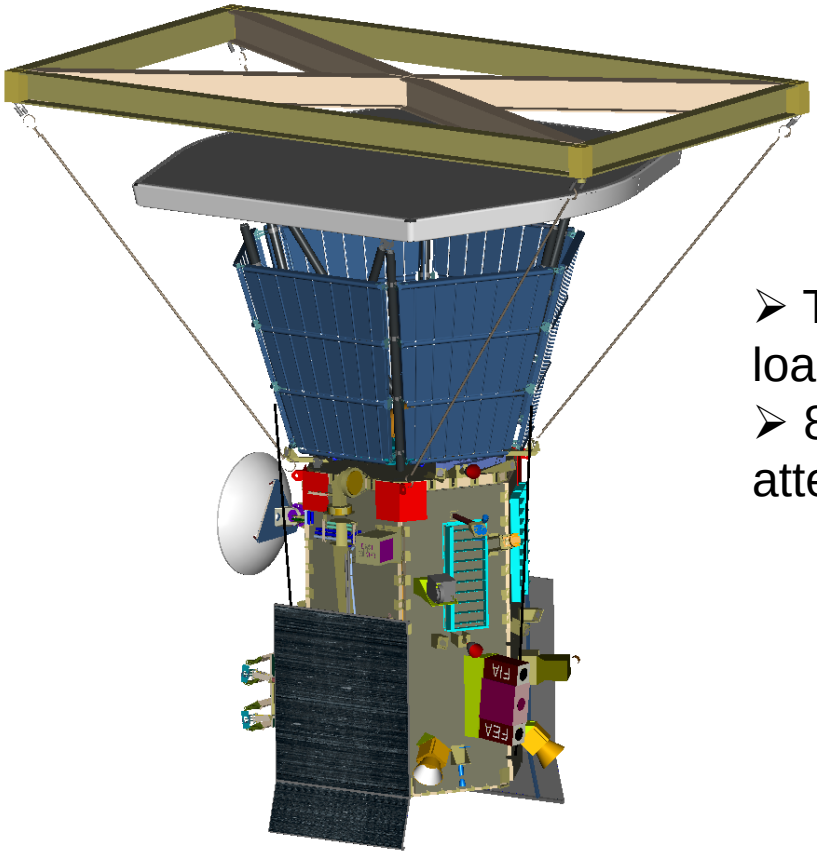
- **Flex Cooling Hose**
  - **Torque Test**
    - **Titeflex S245, 1/2” ID Hose**
    - **Pressurized Hose**
    - **44 in-lbs/hose @ 90°**
  - **Life Test**
    - **Titeflex S245, 1/2” ID Hose**
    - **Pressurized Hose**
    - **100 cycles**
    - **No change in torque noted**
    - **No change between pre and post test Helium leak test**
    - **See backup charts**

Torque Test



Life Test

Spacecraft Handling



- The TPS is not designed to handle lifting loads
- 8 lift configurations were investigated in an attempt to minimize the following:
  - the lift cable angles (39° from vertical)
  - the handling of large GSE components around the Spacecraft (1 kg red brackets)
  - the impact on the Spacecraft structure
  - handling complexity

Phase A Plans

- Mock up non flight, dual gimbal with cooling hoses and harness to better understand upcoming challenges
- Life cycle testing of flexible waveguides as an alternate to rotary joints
- Perform instrument packaging study when instrument details become available
- Refine axial boom deployment mechanism and restraint release mechanism
- Perform honeycomb panel clip test program
- Refine thruster plume study

Backup Slides

Structure Concepts Pros and Cons

Concepts	Pros	Cons
1. Machined Metal Cone	Lightest weight structure	-Brackets to adapt to cone add significant mass -Very large, expensive machining
2. Half Height Panel Umbra Hugger	Excellent propellant tank load path	-Heavier than Concept 4 -Large, expensive machining -Brackets to adapt to cone add significant mass
3. Full Height Panel Umbra Hugger	Provides most packaging volume	-Heavier than Concept 4 -Requires larger TPS
4. Full Height Panel, Straight Hex	-Light weight structure -Smallest bus -Minimizes TPS size	Minimum growth volume

Reference Design Selected: Rationale

- Full Height Panel, Straight Hex: System Mass
- Composite Face Sheets: Mass (AlBemet not available in thin enough sheets)
- Separation Nuts: Mass

Secondary Solar Array Mechanisms Concepts, Pros and Cons

Concepts	Pros	Cons
1. Linear actuator with hose wheel	Pressurized flexible hoses are actively controlled at all times	- Heavy - Takes up significant real estate on top deck - Horizontal sliding array does not meet thermal requirements
2. Linear actuator with hose linkage arms	Pressurized flexible hoses are actively controlled at all times	- Pivot joints of linkage arms introduce additional failure points - Takes up significant real estate on top deck - Horizontal sliding array does not meet thermal requirements
3. Constant force hinge with cable drive and flexible hose	Constant force hinge has high TRL	- Positional control is complicated: it takes two devices (hinge and cable drive) to control array position - Extra device introduces extra weight - Extra moving component introduces extra failure point
4. Linear actuator with service loop	- Light weight	- Limited linear actuator configuration options due to space limitation - Linear actuator has low TRL as SA drive
5. Rotary Actuator with Service Loop – High	- Light weight - No additional mechanism to manage cooling hoses - High TRL for the rotary actuator	- Need to manage and deploy the pressurized hoses
6. Rotary actuator with rotary union	-No pressurized flexible hose to manage - Rotary actuator has high TRL	-Rotary union has low TRL - Rotary union is an additional mechanism to be qualified - Rotary union is heavy
7. Combined Array	- Lowest system mass -Meets thermal and power requirements	-

Primary Solar Array Mechanism Concepts, Pros and Cons

Concepts	Pros	Cons
1. Biaxial Gimbals with Deployment Synchronization Linkage	- Synchronized panels deployment - Simple design with flight heritage	- Linkage arm pivot joints introduce potential failure points
2. Ultraflex SA	Higher packing ratio (stowed volume over deployed volume)	- Deployed geometry is difficult to protect under umbra
3. Roll-out and Passively Deployed Array (RAPDAR)	- Extremely light weight - High packing ratio	- Thermal activated device is not suitable for high temperature environment - Low TRL

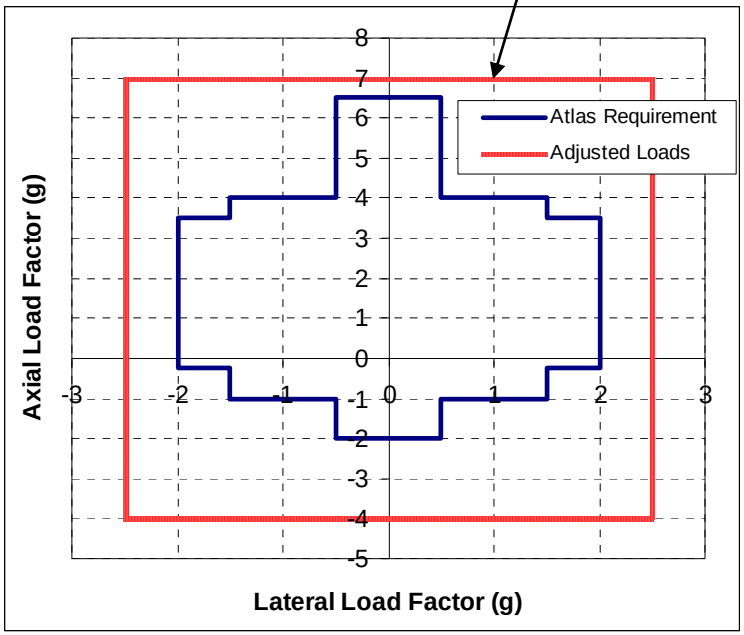
High Gain Antenna Mechanism Concepts Pros and Cons

Concepts	Pros	Cons
1. Straight Boom w/ Internal Rotary Joint, 0.8 m (31.5”) Antenna	Light weight	-Low TRL for the root drive assembly and dish drive assembly (TRL = 5 or below) -Multiple deployments
2. Curved Boom with Internal Rotary Joint, 0.8 m (31.5”) Antenna	More real estate available for arrays and instruments on the S/C	-Low TRL for the root drive assembly and dish drive assembly -Heavier than straight boom -Multiple deployments
3. Straight Boom with External Rotary Joint, 0.8 m (31.5”) Antenna	High TRL for the root drive assembly and dish drive assembly (TRL = 9)	-Heavier than internal rotary joint drive assembly -Multiple deployments
4. Straight Boom with Flexible Waveguide, 0.8 m (31.5”) Antenna	Light weight	-Higher risk due to potential flexible waveguide damage -Multiple deployments
5. Body Mounted with Single Axis Gimbal, 0.6 m (23.6”) Antenna	-Light weight -Simple design -Remains inside umbra at all times	-Requires S/C slew -Requires downlink strategy -Imposes thermal requirements on S/C packaging

Load Cases – Launch Vehicle

- Atlas V 551
  - SPP not within guidelines for Mission Planners Guide (MPG)
    - SPP+Star48V = 2164+610 = 2774 kg
    - MPG valid for 4500 to 19000 kg
  - Adjust design loads for SPP
    - Based off of previous programs
    - Conservative approach
    - Increased Axial from 6.5 to 7.0 G's
    - Increased Lateral from 2 to 2.5 G's

Updated Load Factors for SPP



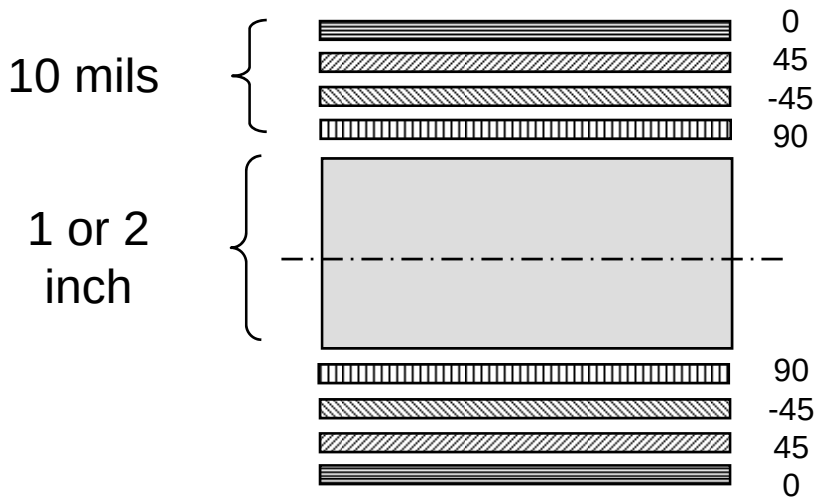
Load Cases - Summary

- **Atlas and STAR-48V**

Load	Source	Axial, G	Lateral, G	
Case		Z	X	Y
1	Atlas / Adjusted	-7.0	2.5	0
2	Atlas / Adjusted	-7.0	0	2.5
3	Atlas / Adjusted	-7.0	1.77	1.77
4	Atlas / Adjusted	-7.0	1.77	-1.77
5	Atlas / Adjusted	4.0	2.5	0
6	Atlas / Adjusted	4.0	0	2.5
7	Star-48V	-10.6	0	0

Primary Structure Panels

- Aggressive light weight design
  - Composite facesheet, M55J
    - Each facesheet is balanced, but <u>not symmetric</u>
    - Laminate is symmetric about center of core
  - Analysis shows positive margins
  - Testing of clips on honeycomb coupons planned



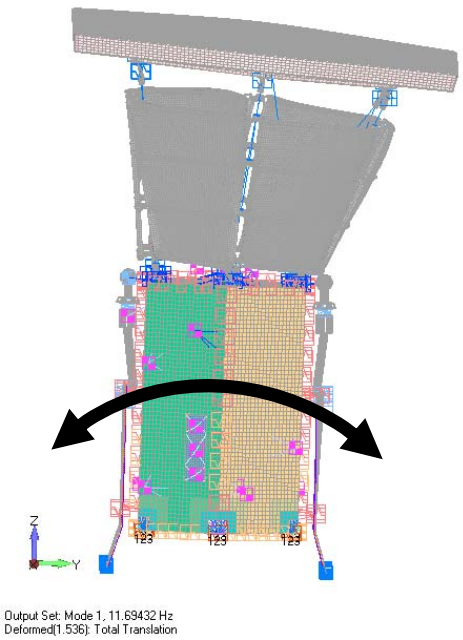
Layup: $[0/45/-45/90]_T$

Core: Al 3.1 pcf

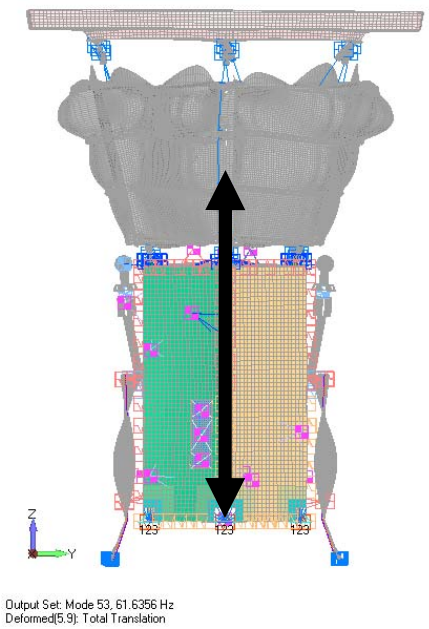
Layup: $[90/-45/45/0]_T$

FEM Results - Frequency

- **Meets Atlas V 551 Requirements**
  - **11.7 Hz meets lateral 8 Hz requirement**
  - **61.6 Hz meets axial 15 Hz requirement**



Mode 1: 11.7 Hz Lateral



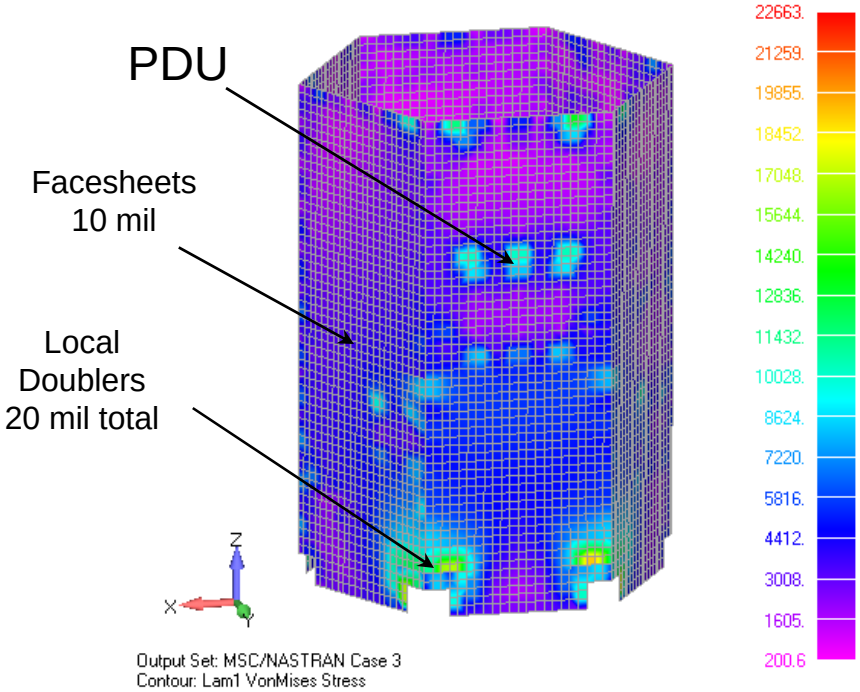
Mode 53: 61.6 Hz Axial

Mass Participation

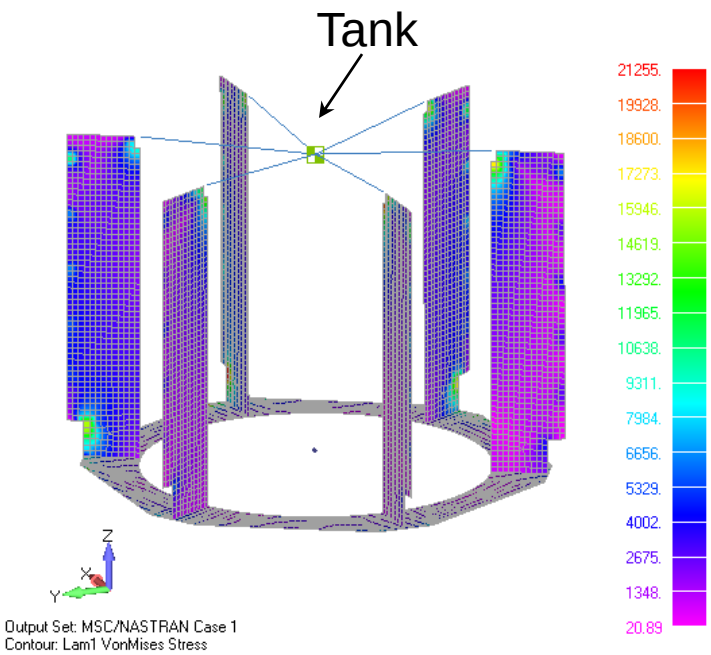
Mode	Freq	Modal Effective Mass Fraction					
No.	Hz	TX%	TY%	TZ%	RX%	RY%	RZ%
1	11.7	3.6%	23.0%	0.0%	67.2%	10.8%	0.0%
2	11.8	22.8%	3.6%	0.0%	10.9%	67.1%	0.0%
3	17.3	0.0%	0.0%	0.0%	0.0%	0.0%	50.4%
4	23.8	0.0%	0.1%	0.0%	0.3%	0.0%	0.0%
5	23.9	0.0%	0.0%	0.1%	0.0%	0.0%	0.0%
6	26.9	0.0%	0.1%	0.0%	0.0%	0.0%	0.0%
7	26.9	0.1%	0.0%	0.0%	0.0%	0.0%	0.0%
8	27.0	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%
9	27.9	0.0%	0.3%	0.0%	0.0%	0.0%	0.0%
10	27.9	0.3%	0.0%	0.0%	0.0%	0.0%	0.0%
11	28.1	0.0%	0.0%	0.3%	0.0%	0.0%	0.0%
12	30.7	0.0%	0.0%	0.0%	0.0%	0.0%	0.1%
13	30.7	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%
14	33.2	46.8%	3.5%	0.0%	1.1%	16.2%	0.1%
15	33.5	3.3%	44.7%	0.0%	15.1%	1.1%	0.0%
16	35.7	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%
17	35.8	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%
50	60.6	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%
51	60.6	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%
52	61.1	0.0%	0.0%	11.5%	0.0%	0.0%	0.4%
53	61.6	0.0%	0.0%	21.1%	0.0%	0.0%	0.1%
54	62.8	0.2%	0.1%	3.9%	0.1%	0.2%	0.0%

FEM Results – Stress (1/2)

- **Driven by Launch load case (7G axial, 2.5 G lateral)**
- **Results shown are Von Mises of smeared M55J properties**

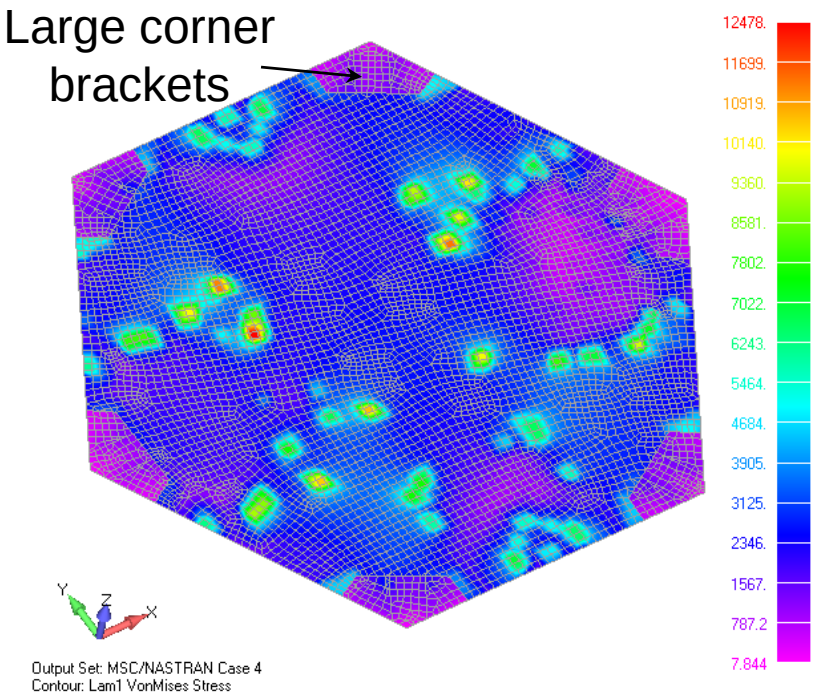


Facesheet Von Mises



Facesheet Von Mises

FEM Results –Stress (2/2)



Facesheet Von Mises

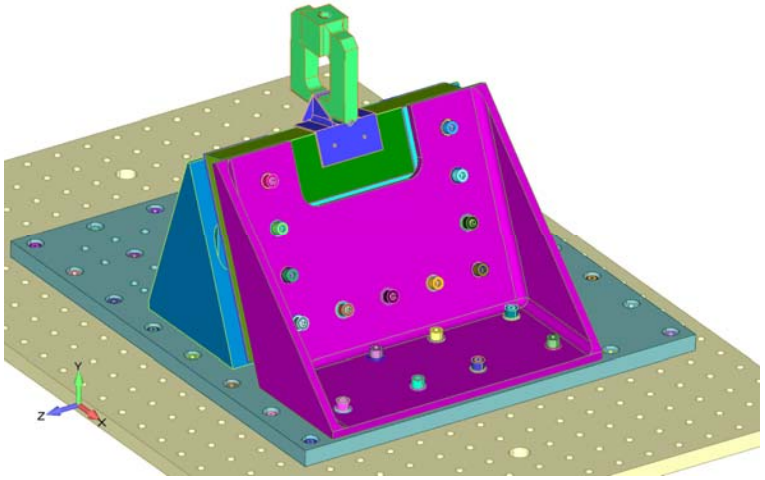
FEM Results – Stress Summary

- **Bus composite facesheet has <u>all positive margins</u>**
  - **Future analysis to include ply by ply analysis**

- **Many light weight opportunities to strengthen as needed as the design matures (brackets, potting area, facesheet doublers)**

Component	Stress Type	Material	Stress ksi	Ult Allowable ksi	Ult FS	Ult MS
Side Panels	Von Mises*	M55J	22.7	35.0	1.5	0.0
Prop Decks	Von Mises*	M55J	21.3	35.0	1.5	0.1
Top Deck	Von Mises*	M55J	12.5	35.0	1.5	0.9

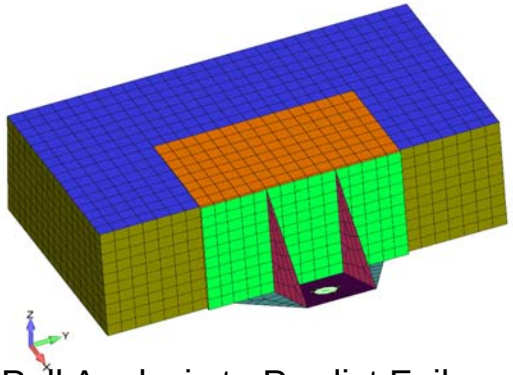
Structure Clip Tests Planned



Coupon Test Assembly, 67 Test Configurations

Drawing Number	Coupon Description					
	Honeycomb core		Facesheet			Panel size
	density / size	thk	gauge	material	layup	
7434-1612-09	1.6 pcf-1/16"-5056Al	1"	0.008"	M55J	[-60/0/60]	8"x6"
7434-1612-19	1.6 pcf-1/16"-5056Al	1"	0.010"	M55J	[-45/90/45/0]	8"x6"
7434-1612-49	3.1 pcf-1/8"-5056Al	1"	0.010"	M55J	[-45/90/45/0]	8"x6"
7434-1612-69	1.6 pcf-1/16"-5056Al	1"	0.010"	M55J	[-45/90/45/0]	8"x6"
			0.010"	M55J	[0/45/90/-45] doubler	
7434-1612-79	3.1 pcf-1/8"-5056Al	1"	0.010"	M55J	[-45/90/45/0]	8"x6"
			0.010"	M55J	[0/45/90/-45] doubler	
7434-1612-89	1.6 pcf-1/16"-5056Al	1"	0.008"	M55J	[-30/90/30]	8"x6"
7434-1615-19	1.6 pcf-1/16"-5056Al	1"	0.010"	2024Al-T81		8"x6"
7434-1615-49	3.1 pcf-1/8"-5056Al	1"	0.010"	2024Al-T81		8"x6"
7434-1624-09	1.6 pcf-1/16"-5056Al	1"	0.008"	M55J	[-60/0/60]	5"x5"
7434-1624-19	1.6 pcf-1/16"-5056Al	1"	0.010"	M55J	[-45/90/45/0]	5"x5"
7434-1624-29	3.1 pcf-1/8"-5056Al	1"	0.010"	M55J	[-45/90/45/0]	5"x5"
7434-1624-39	3.1 pcf-1/8"-5056Al	2"	0.015"	M55J	[-60/0/60]s	5"x5"
7434-1625-09	1.6 pcf-1/16"-5056Al	1"	0.010"	2024Al-T81		5"x5"
7434-1625-19	3.1 pcf-1/8"-5056Al	1"	0.010"	2024Al-T81		5"x5"
7434-1646-09	3.1 pcf-1/8"-5056Al	1"	0.010"	M55J	[-45/90/45/0]	9"x6"
						non-symmetric bolt pattern
7434-1655-09	3.1 pcf-1/8"-5056Al	2"	0.015"	M55J	[-60/0/60]s	8"x6"
						notched, square edges
7434-1657-09	3.1 pcf-1/8"-5056Al	2"	0.015"	M55J	[-60/0/60]s	8"x6"

Panel Coupons, Aluminum and Composite Facesheets, Aluminum Honeycomb Cores

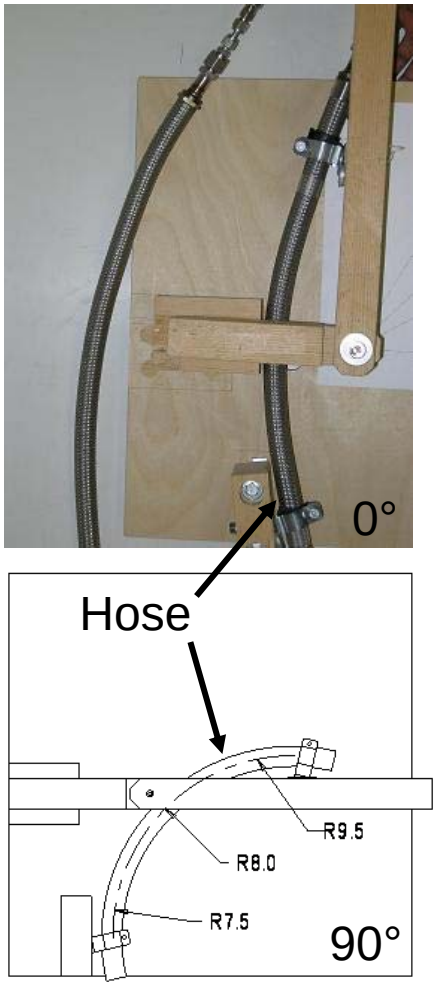


Joint Pull Analysis to Predict Failure Modes and Loads, Including MGSE and Laminate Ply Failure

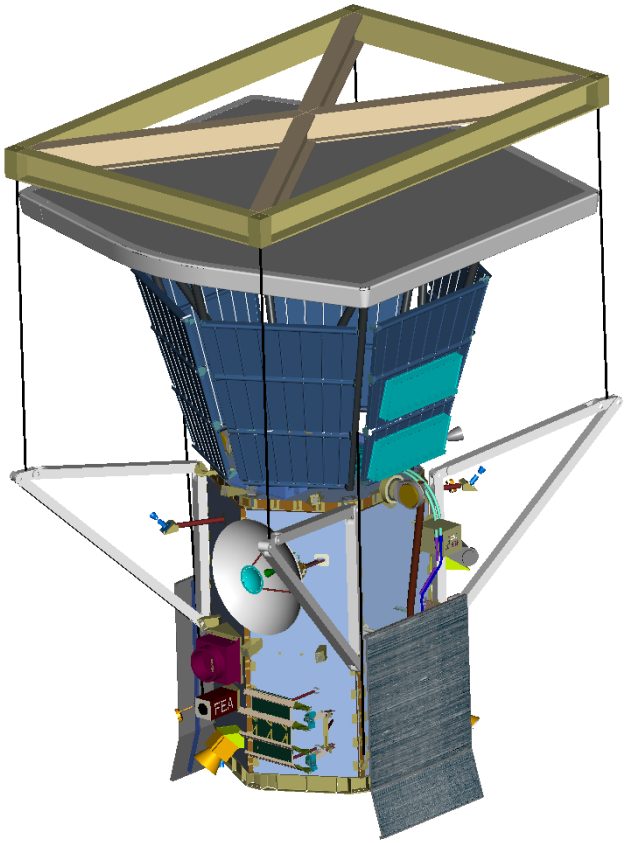
Testing planned for 1st quarter FY2010

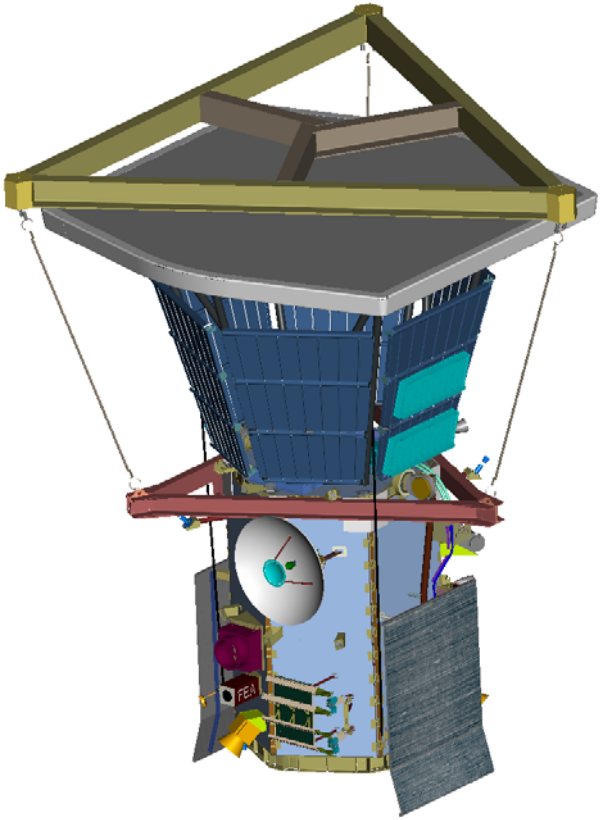
Flex Cooling Hose Life Tests

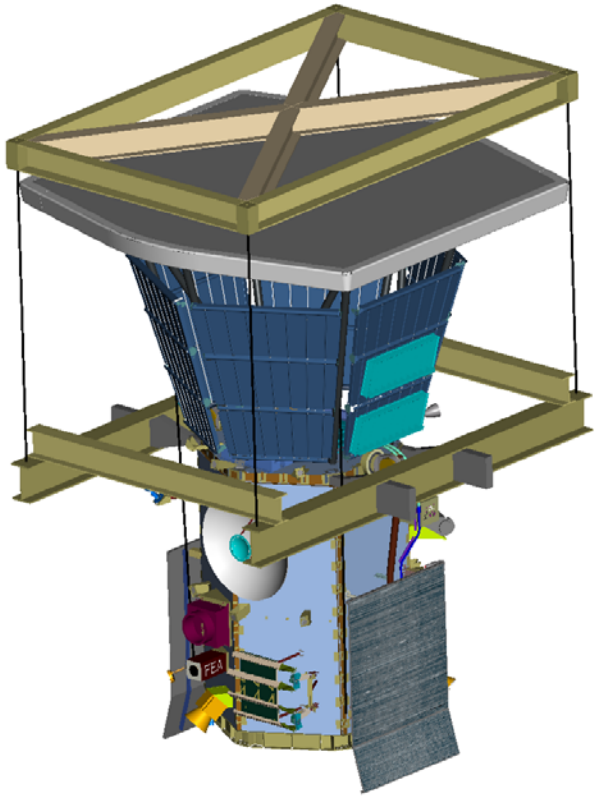
- Development tests were performed to evaluate flexible SS hose for coolant transfer across rotary joints.
- Test Article: Titeflex S245, ½” ID, inner core - SS321 annular convolution, reinforcement - SS321 wire braid, minimum allowable bend radius: 9.0 inch,
- Test setup: Hose filled with water and pressurized to 200 psi with GN2. Hose was attached to a wood hinge assembly
- Resistance torques were measured on several hose configurations while maintaining the minimum bend radius requirement. Max resistance torques on a single hose:
  - 0° to 90° bend: 35 in-lb
  - ±45° bend: 23 in-lb
- Life test: The pressurized hose was subjected to 100 cycles (0°-90°-0°)
  - Resistance torque was monitored and no degradation was observed after 100 cycles.
  - Helium vacuum leak test was performed before and after life test and no leaks were detected, implying absence of defects.
  - Hose did not exhibit any visual signs of damage.
- Results show that SS Flex hose is a viable option. More hoses and configurations will be tested in the next phase.



Examples of Alternate Spacecraft Handling Options







In order to get nearly vertical cables required large lift brackets to be installed on the spacecraft. That risk was deemed too high and these concepts were eliminated.

Solar Probe Plus
Mission Concept Review

Dust Effects

Douglas Mehoke

The Johns Hopkins University
APPLIED PHYSICS LABORATORY

Outline

- **Requirements**
  - SPP Dust Environment
- **Task Efforts**
  - S/C Fluence Predictions
  - Impact Modeling
  - EOS Development
  - CTH Modeling
- **Reference Design Feasibility**
  - Preliminary Damage Assessment
    - TPS
    - Solar Arrays
  - Peer Review summary
  - ST Illumination Study
- **Phase A/B plans**

SPP Dust Study Objectives

Study Objective

- **Develop a dust and meteoroid environment for the SPP Mission and assess its effects on the spacecraft**

Approach

- **Develop dust model and method for predicting impacts on specific spacecraft surfaces**
  - **Develop methodology for analyzing particle impacts at velocities up to 350 km/s**
    - **Develop the computer modeling technique to analyze these impacts**
    - **Generate material equations of state (EOS) for very high energy impacts**
    - **Develop methodology for assessing the damage away from the crater zone**
  - **Demonstrate mission feasibility by making a preliminary damage estimate for key surfaces**
- **Develop model predicting effect of reflected light on ST performance**

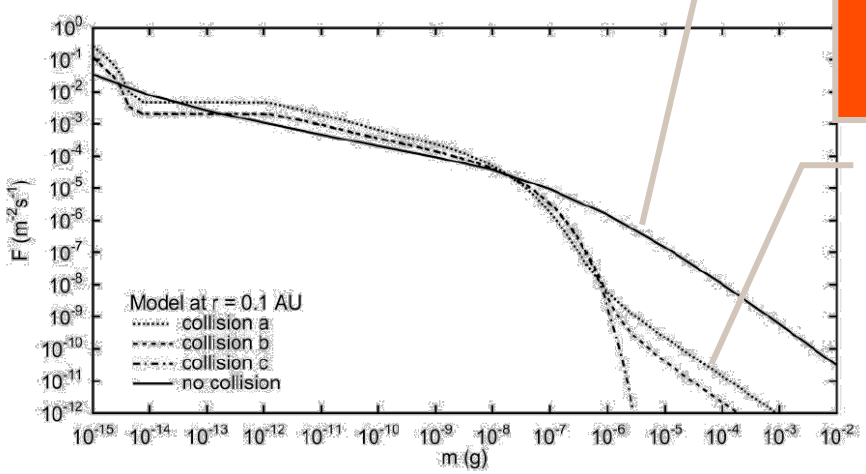
SPP Dust Study Team

Douglas Mehoke	Study lead	APL
Mary Kae Lockwood	Spacecraft Systems Engineer	APL
Dust Impact Analysis		
Cesar Carrasco	Dust model & fluence predictions	UTEP
P. K. Swaminathan	CTH modeling	APL
Gerald Kerley	EOS	KTS
Robert Brown	CTH modeling	APL
Kyle Manning	CTH modeling	APL
Ramesh Batra	Constitutive damage modeling	Va Tech
Testing		
Andy Finchum	Impact testing facility	MSFC
Brian Wells	Impact test data	Auburn U
ST illumination Analysis		
Tom Strikwerda	ST performance	APL
Shadrian Strong	Dust illumination modeling	APL

The Dust Environment

- **Dr. Carrasco of UTEP has developed a tool that uses the Mann dust environment and the S/C geometry to predict the impact profile**
- **The Mann model has been reviewed by Dr. David Lario of the SPP science team**

- **Within 1AU of the Sun:**
  - Collision model "a" from Mann *et al.* 2004 used
  - Collision effects on flux assumed to be zero at 1 AU and increase linearly from 1AU to the values shown in the figure below at 0.1AU
  - For radii less than 0.1AU collision effects assumed to be those at 0.1AU



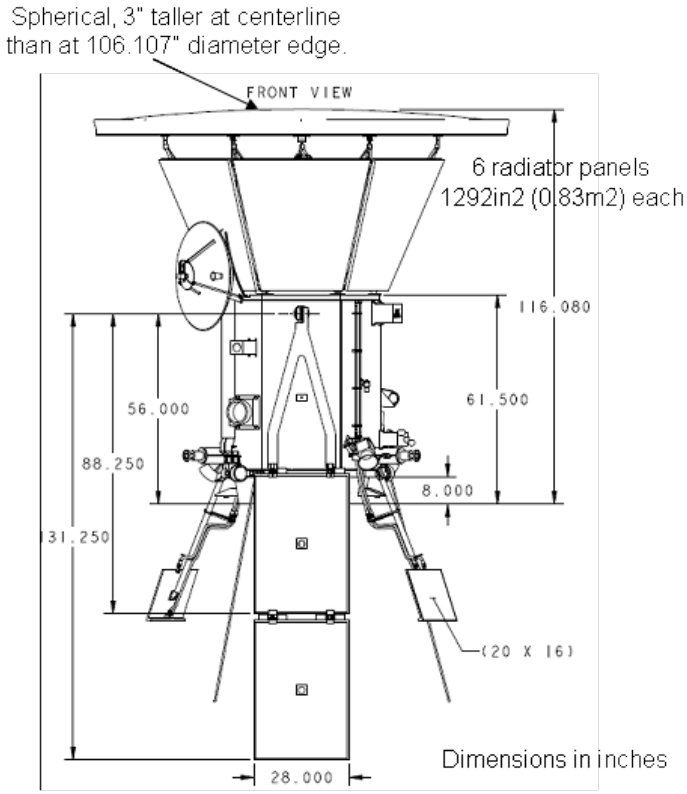
Dust model used for shield design (most conservative)

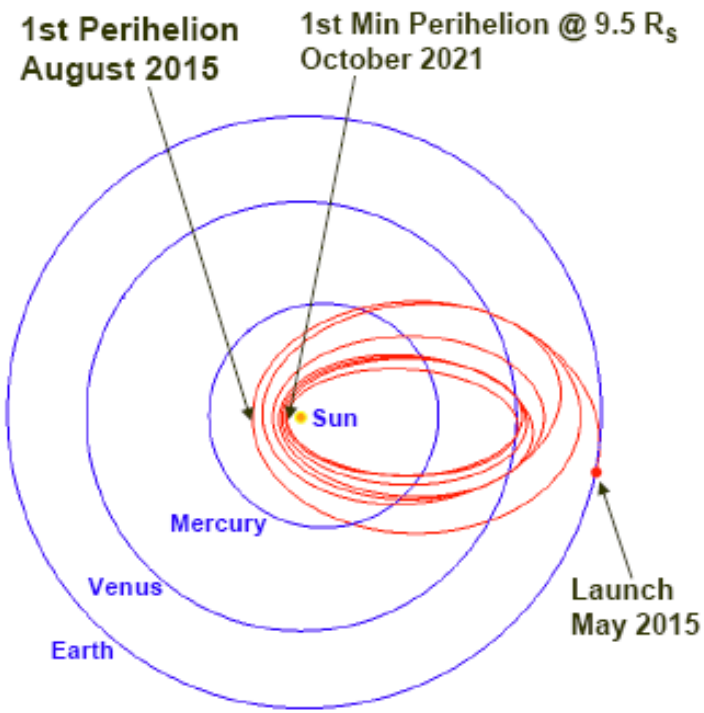
Collision model used

From Mann *et al.* 2004

The Dust Environment – S/C Model

UTEP developed a model that used the S/C geometry, the mission trajectory and the dust model to predict particle fluxes on the different S/C surfaces





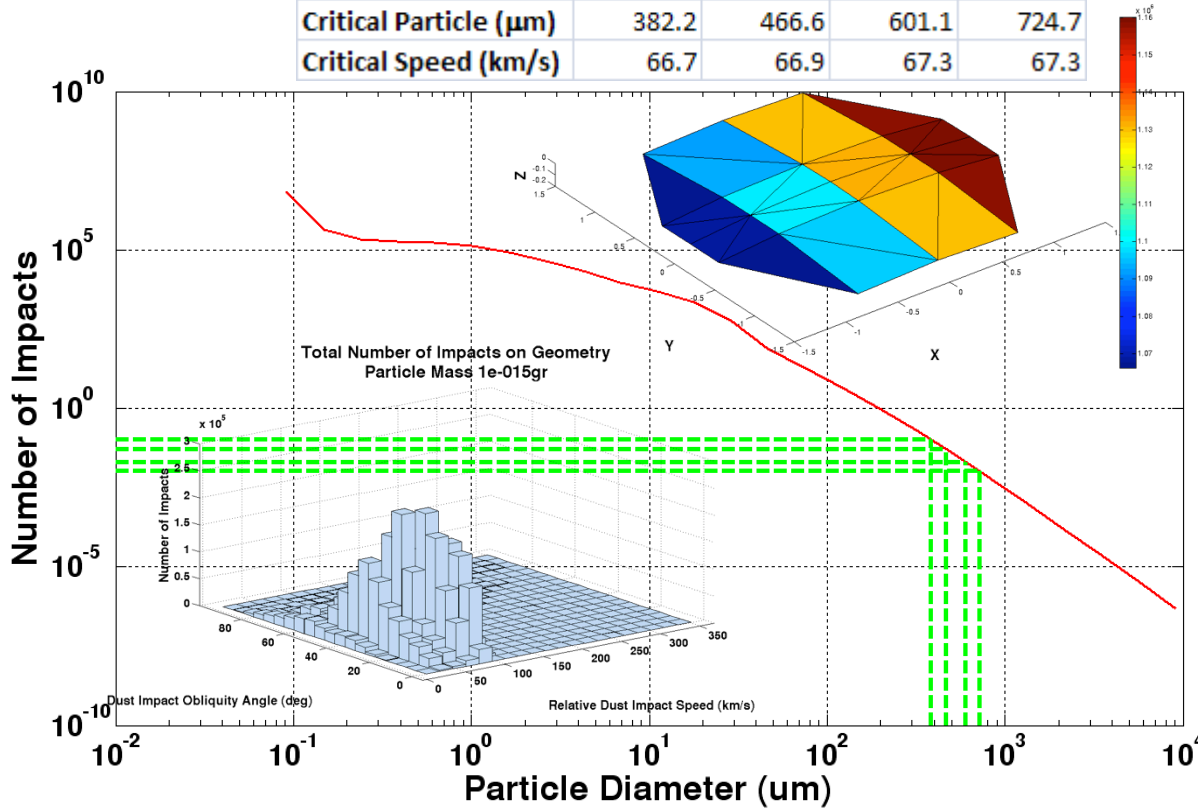
Note: January 09 spacecraft version used for original fluence predictions

The Dust Environment – Fluence Predictions

- **Dust modeling involves two cases**
  - Case 1: Probability of No Impact for the critical particle size
  - Case 2: Cumulative impacts of many small particles

- **Challenge: SPP will be exposed to particle speeds up to 350 km/s**
  - Damage predictions will need to rely on analysis

	Probability of No Impact (%)			
	90	95	98	99
Critical Particle (µm)	382.2	466.6	601.1	724.7
Critical Speed (km/s)	66.7	66.9	67.3	67.3

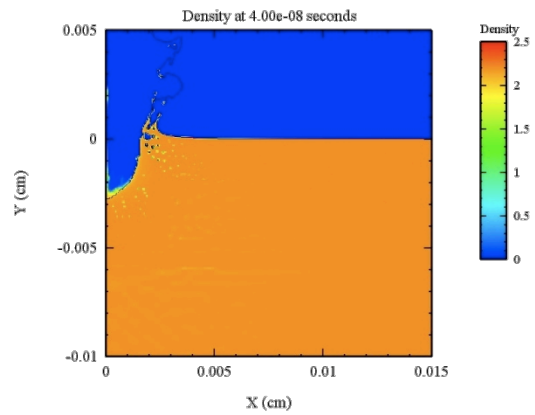


Impact Modeling – CTH Analysis

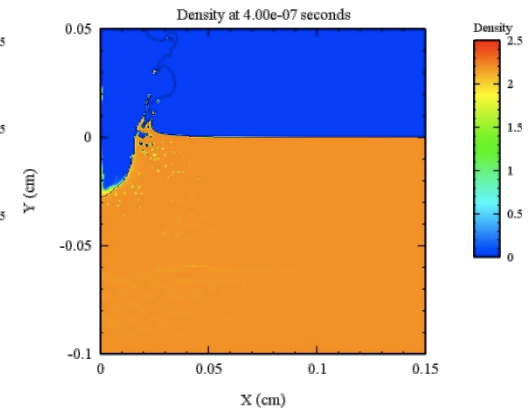
- One effort has focused on validating the CTH modeling results at high velocities
- This effort is critical because of the limited test capabilities
- Code checking
  - Semi-infinite slab models used to check velocity dependence
  - Zoning and mesh run control parameters checked
- Complicated material damage model
  - CTH is not typically used to predict structural damage in the lower energy shock regime
  - Correlate model to AU data to define spall zone

Semi infinite model

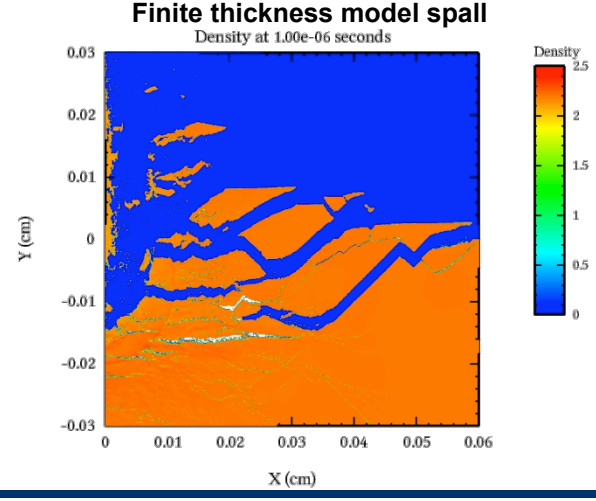
10 micron at 10km/s



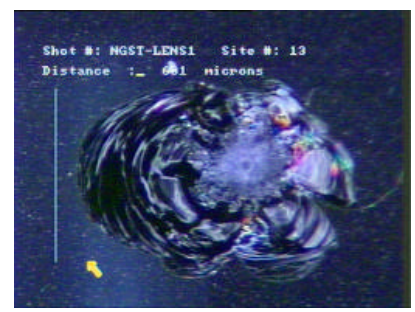
100 micron at 10km/s



Finite thickness model spall



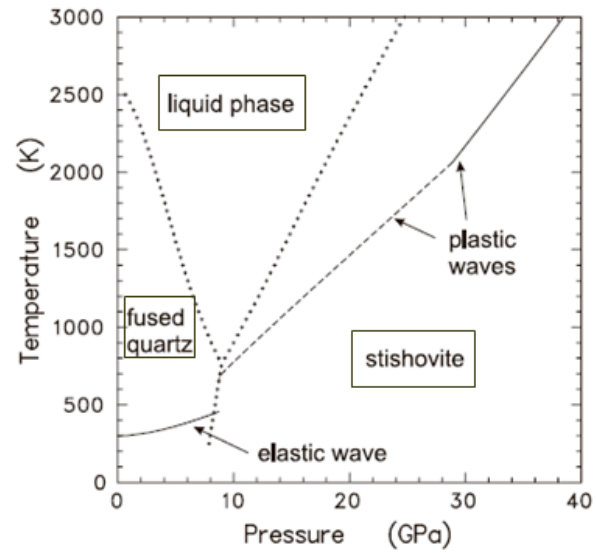
AU test results



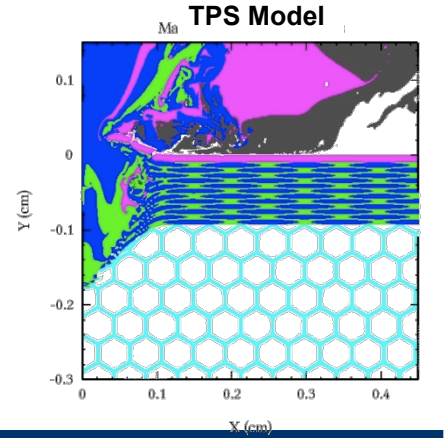
Impact Modeling - EOS

- High energy impact analyses predict damage by following the dissipation of the impulsive shock in the target material
- A significant part of the dust study focused on extending material EOS to the very high energies needed
  - The failure of a material and even the properties prior to failure all depend on the thermal state of the material which is defined by its equation of state (EOS)
  - Standard temperature/energy corrections used are not correct for melt effects, which are a strong function of the pressure
  - Also, EOS were generated for the inhomogeneous materials in the TPS
- Dr. Kerley (Kerley Technical Services) is an expert in developing these material properties

Fused Silica EOS Model



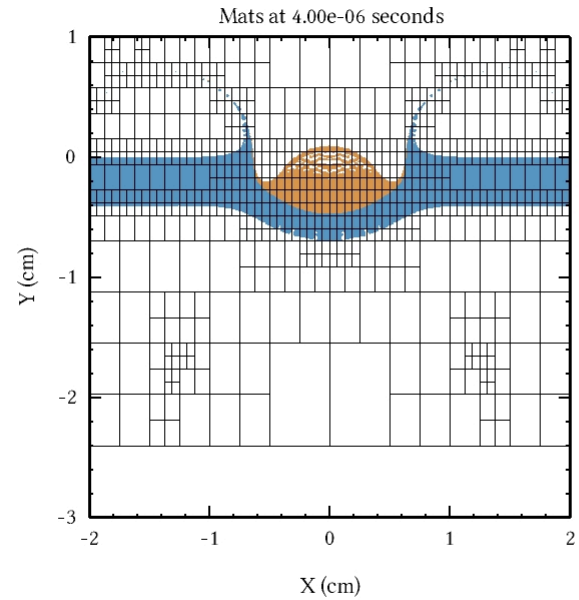
TPS Model



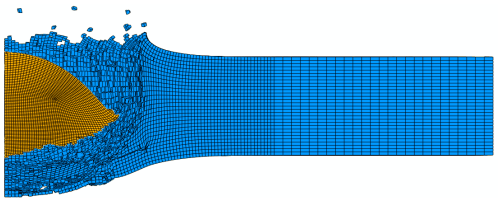
Impact Modeling – Low Shock Area Damage

- Due to its limited damage model, CTH does not predict damage away from the crater zone reliably
- To address this limitation, Prof. Batra (Va Tech) is attempting to couple the CTH runs to a FE code with a higher fidelity damage model
- Using a CTH run to develop initial conditions for the system at some time, the analysis is shifted to a FE code (ABAQUS) to complete the run and predict damage away from the crater zone
- This coupling is a new analytical approach to the problem
- The hand-off needs to put the stress and displacement state of the original model into a form that can be recognized by the FE model
- The models are being checked by running the two codes independently for sample problems, and comparing those results to the coupled model

CTH run at 4 microsec

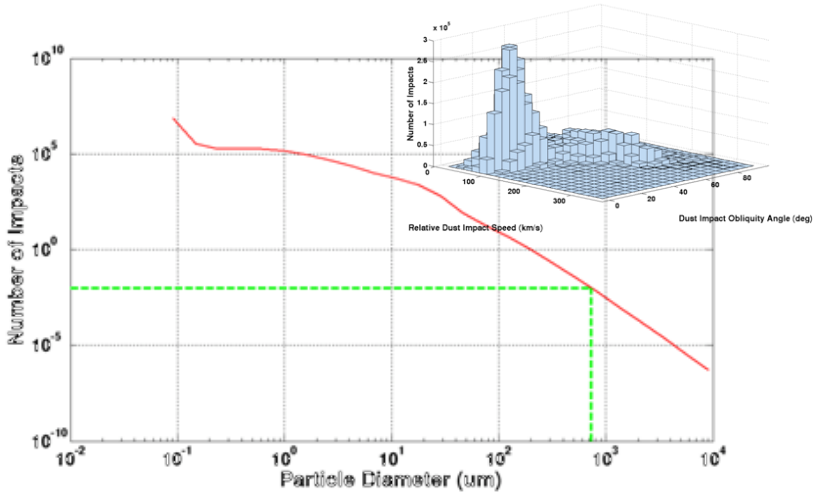


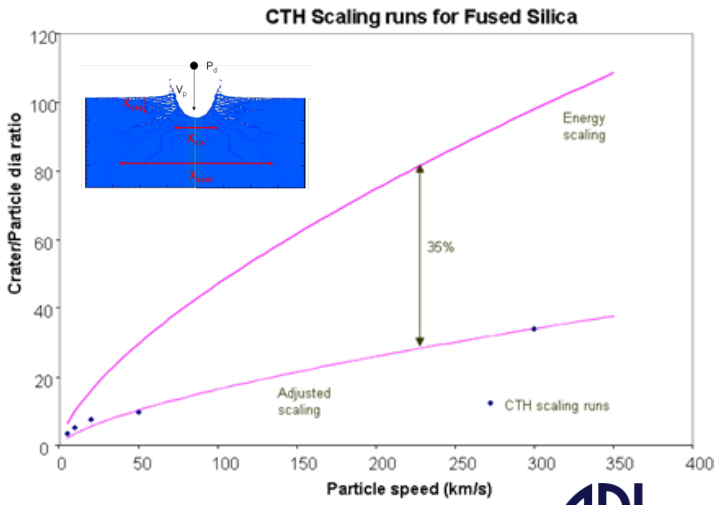
ABAQUS run at 4 microsec



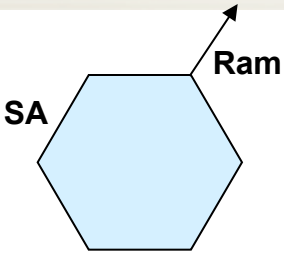
Reference Design Preliminary Damage Assessment

- The damage assessment aims at quantifying the results from the accumulated particle impacts, including the predicted particle sizes, velocities, and angles
- The particle flux is grouped into 25 particle mass bins, 20 velocity bins, and 20 angle bins
- Normal velocities are calculated for all particles
- Three damage thresholds are set
  - No damage
  - Minimal damage
  - Penetration
- The damage is based on energy scaling which is conservative

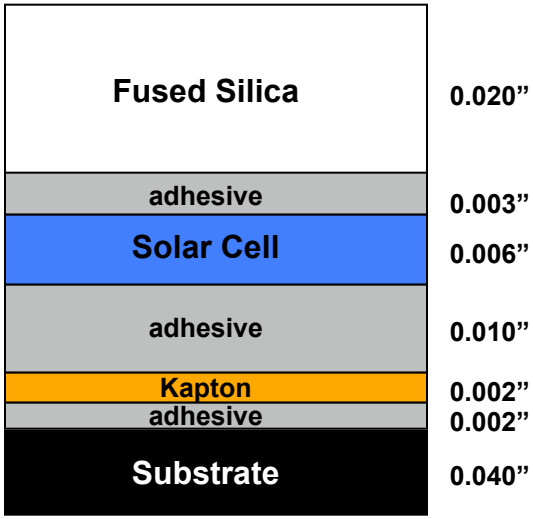


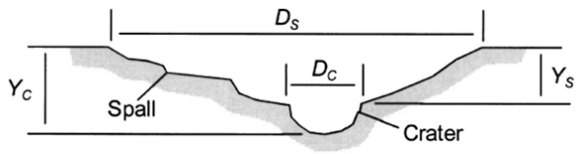


Preliminary Damage Assessment – Cumulative damage to the SA

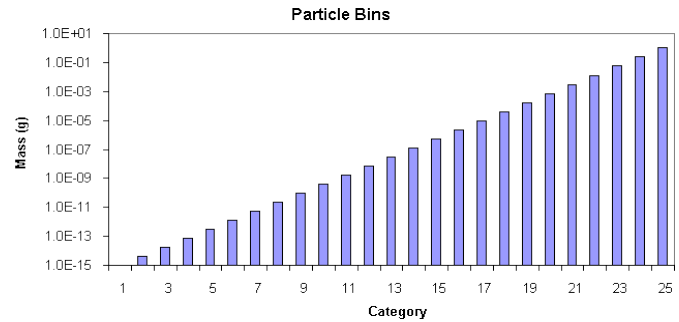


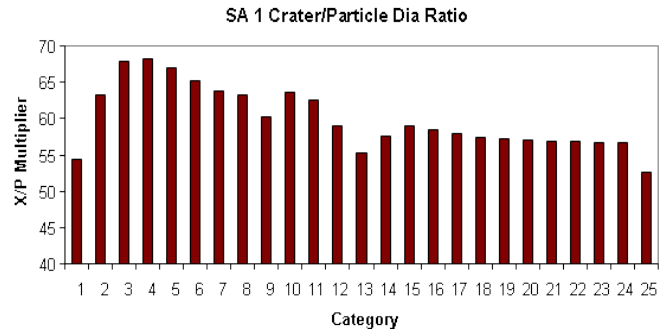
Note: damage assessment includes estimate of fluence in Ref Des array position, from Jan 09 geometry runs

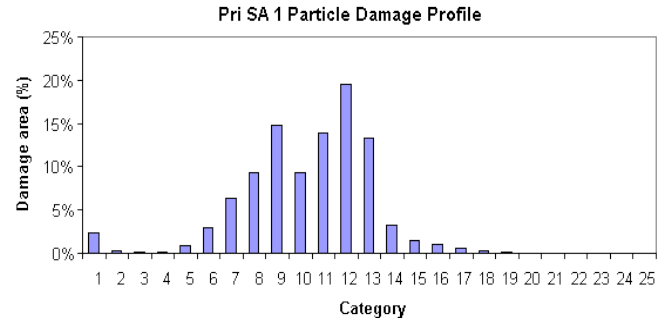




SA 1 total crater area	52 cm^2
Hits – no damage	0
min damage	1,655,553
penetration	698







For the small solar cell crater sizes, vendor damage expectation is only area loss

Vendor expectation is a low probability for loss of the cell or string

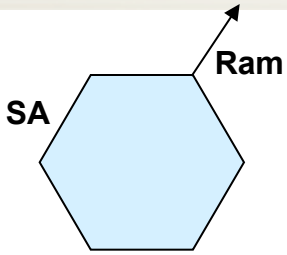
Predicted damage	0.75%
Limit	1%

Testing is planned in Phase A

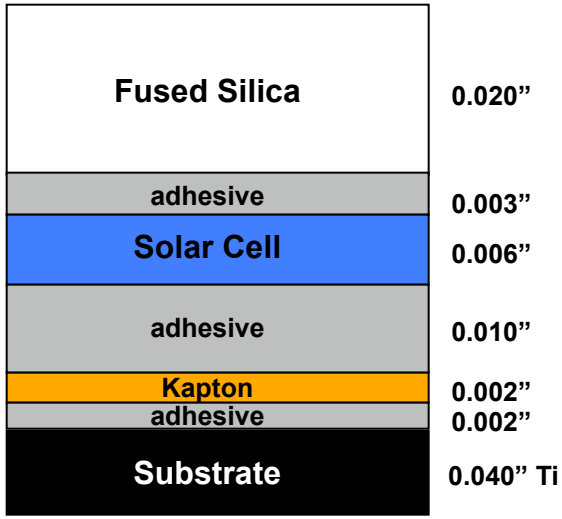
Damage thresholds

No damage	Xdep < 0.1 micron
Minimal damage	0.1 micron < Xdep < 0.023”
Penetration	Xdep > 0.023”

Preliminary Damage Assessment – Critical Particle Damage to the SA



Note: damage assessment includes estimate of fluence in Ref Des array position, from Jan 09 geometry runs



UTEP Critical Particle Information

	Probability of No Impact (%)			
	90	95	98	99
Critical Particle (μm)	270.3	332.6	431.9	523.0
Critical Speed (km/s)	124.1	125.6	127.6	128.3

Energy scaling analysis

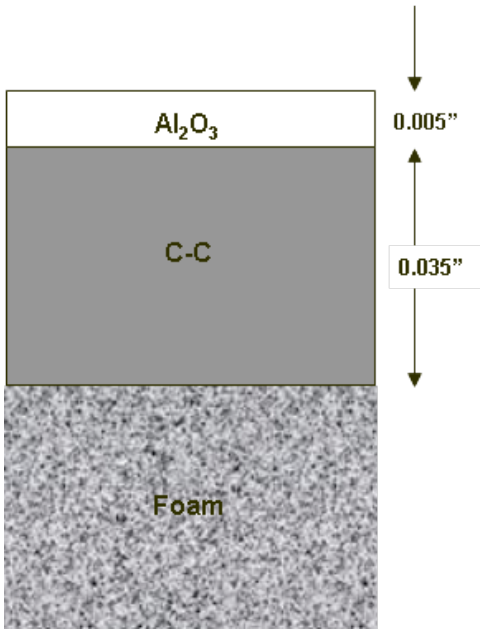
crater depth (inches)	0.058	0.072	0.094	0.114

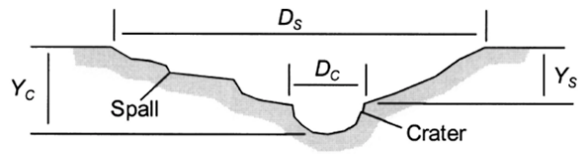
MSFC test data comparison

vel (km/s)	5.6	
xd / Pd =	6.1	
Pd	400	
xd (inches)	0.096	predicted
xd (inches)	0.012	actual
ratio (predicted/actual)	8.3	

Based on the predicted value and the factor of 8 between the actual and predicted carter depth, the present substrate thickness is in the right range

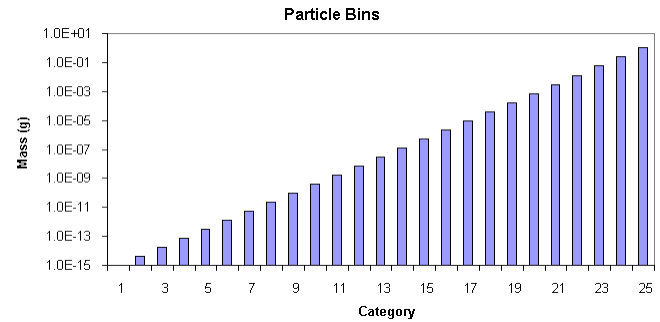
Preliminary Damage Assessment – Cumulative Damage to the TPS

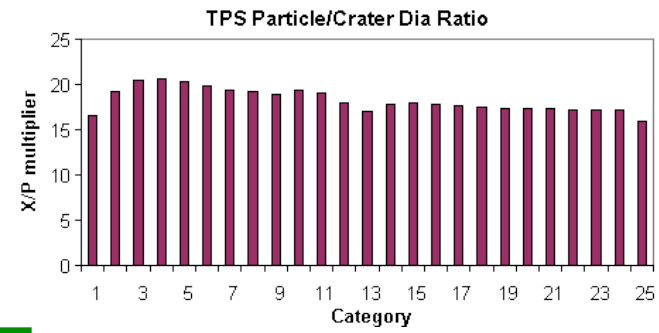


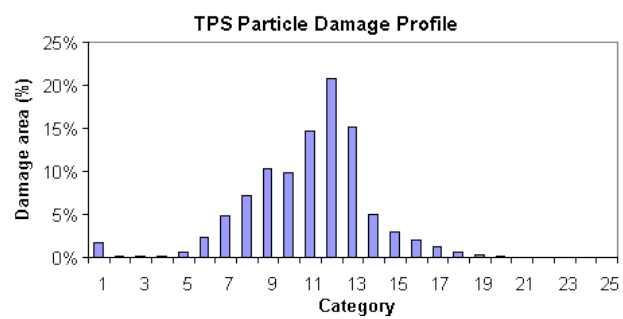


Total crater area	6.2 cm^2 (1)
Hits – no damage	6473247
min damage	6762
penetration	25

(1) Manolis Georgoulis is an independent analysis looking at mass loss from the TPS calculated ~ 6 cm^2 total affected area on the TPS







Damage thresholds

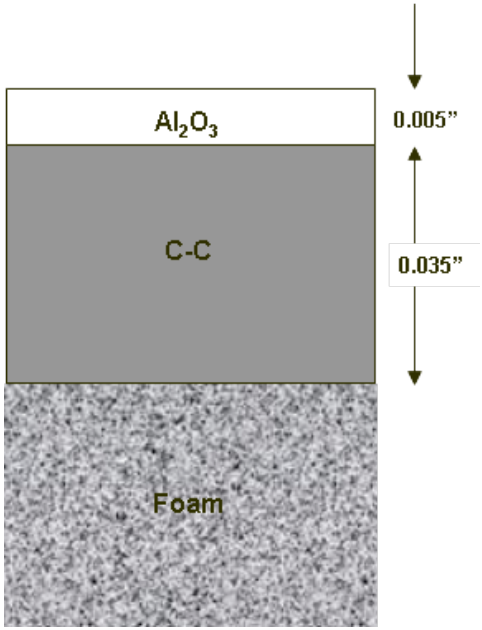
No damage	Xdep < 0.005”
Minimal damage	0.005” < Xdep < 0.040”
Penetration	Xdep > 0.040”

The preliminary damage involves thermal effects that scale on area loss

Predicted damage	0.01%
Limit	10%

Testing is planned in Phase A

Preliminary Damage Assessment – Critical Particle Damage to the TPS



Damage thresholds

No damage **Xdep < 0.005”**

Minimal damage **0.005” < Xdep < 0.040”**

Penetration **Xdep > 0.040”**

UTEP Critical Particle Information

	Probability of No Impact (%)			
	90	**95**	**98**	**99**
Critical Particle (μm)	382.2	466.6	601.1	724.7
Critical Speed (km/s)	66.7	66.9	67.3	67.3

Energy scaling analysis

crater depth (inches)	0.253	0.309	0.400	0.482

MSFC test data comparison

vel (km/s)	5.5	
xd/Pd	3.2	
Pd (microns)	400	
xd / Pd =	16.8	predicted
xd / Pd =	3.18	actual
ratio (predicted/actual)	5.28	

Test results

crater depth (mm)	8.97	
crater depth (in)	0.359	1.895 scaled penetration for crit part case

Peer Review Summary

A Dust Study Peer Review was held on 5/13/09

The review team consisted of the following

K. Hemker	JHU	Mech Eng	external reviewer
K.T. Ramesh	JHU	Mech Eng	external reviewer
H. Brandhorst	AU/SRI	Env effects	external reviewer

Summary comments:

- The big areas were covered
- Concern is on the accuracy of the Mann model
- It is important to simplify the complex problem
- Effort is focused correctly on science behind impact analysis
- CTH damage model is limited

ST Illumination Study – Problem Description and Model

- **What is the nature of the SPP dust environment?**
  - Impact on star tracker (ST) observations
  - Reliance on dust models to characterize environment
  - Are special ST modifications required?
- **How can it be modeled:**
  - Assume: optically thin dust distribution
  - Mie Scattering: dominant source of scattered irradiance for 0.4 – 1 micron operational wavelength
  - Rayleigh scattering: important for smallest dust masses, minimal net contribution
  - Adopt Mann et al. cumulative dust spatial density, mass range, dust flux (nominal model)
  - Compare with D. Lario APL conservative case (modified Mann density distribution)
  - Determine net solar scattered irradiance observed at the aperture of a ST within a field-of-view cone.

ST Illumination Study – Conclusions

- Star trackers manufactured tolerance:
  1.49e-9 W/cm^2 (25° FOV)
  Modeled irradiance:
  1.48e-8 W/cm^2 (25° FOV)
- Modeled dust irradiance at 10 solar radii is in excess of predicted star tracker tolerance
- ST vendor (Sodern) says their tracker will operate in spec for the background level modeled

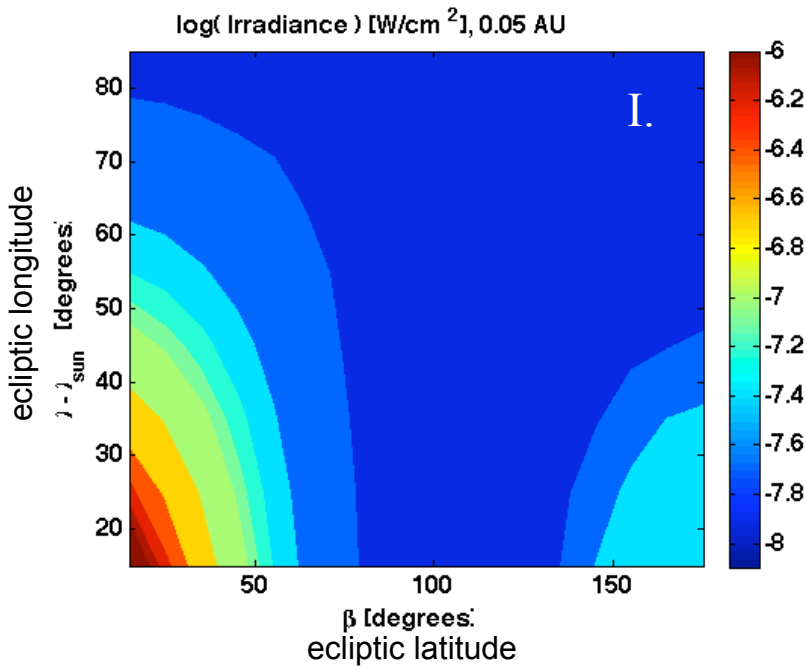


Figure 2. *A model predicted irradiance map within the ecliptic plane at 10 solar radii (nominal model I.).* This map indicates that star trackers placed at ecliptic longitudes > 80 degrees & latitudes between 70 and 125 degrees, receive the least scattered irradiance.

Phase A/B Plans

- **The feasibility effort has been directed at developing the high energy impact analysis methodology and doing a preliminary assessment for those things that could not be shielded**
- **Phase A/B plans**
  - Fluence predictions for detailed S/C surfaces — UTEP
  - Material EOS for flight materials — KTS
  - Complete coupled FE/CTH analysis — Va Tech, APL
  - Develop mitigation strategies for other S/C systems — APL
    - Design requirements (MLI spacing, wall thickness, etc)
  - Perform impact testing to validate damage models — MSFC, AU
  - Develop mitigation analysis/plans for ST — APL

Solar Probe Plus
Mission Concept Review

Avionics

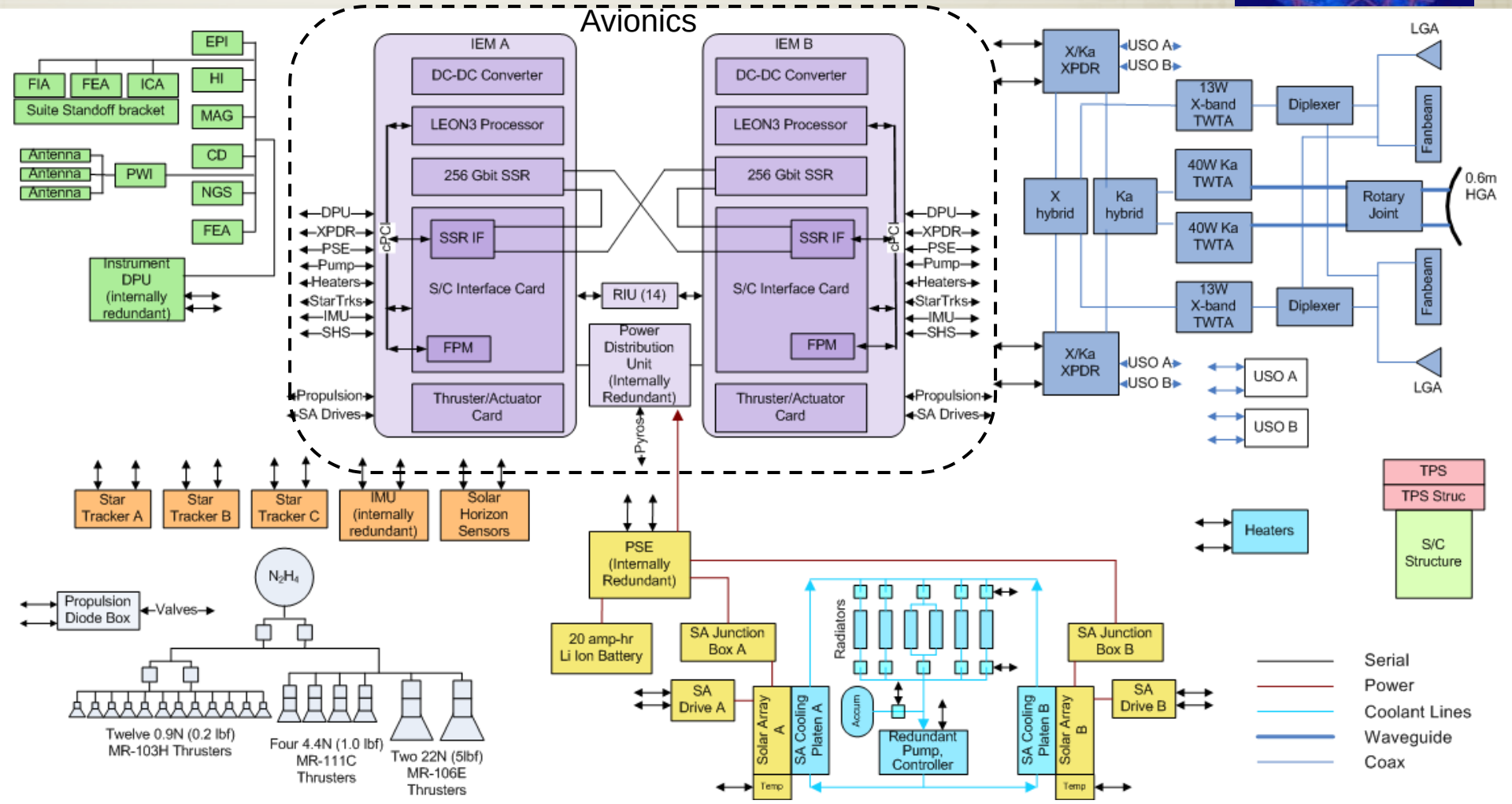
Dan Rodriguez
Avionics Subsystem Lead

The Johns Hopkins University
APPLIED PHYSICS LABORATORY

Avionics Subsystem Topical Outline

- Requirements
- Reference Design
- Redundancy Architecture
- Physical Characteristics
- Studies & Trades
- Technology Development / LEON3 Processor Risk Mitigation

Spacecraft Block Diagram



Working Requirements

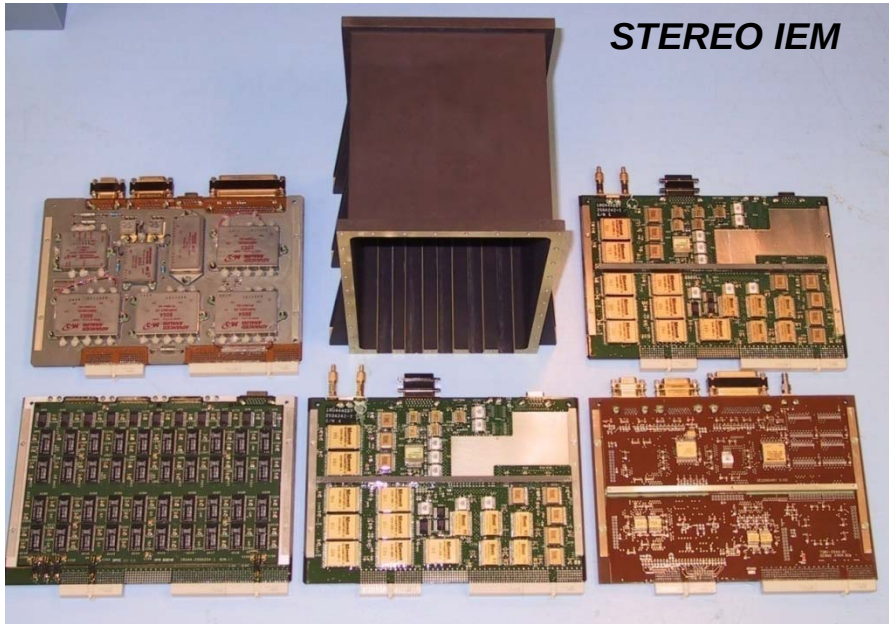
- **Command and Data Handling**
  - Instrument data collection at 123kbps
  - 256Gbit data storage – to accommodate data collection objectives
  - Downlink up to 850kbps
- **Guidance and Control**
  - 50Hz control loop
  - Thruster control to 2-msec
  - Actuator & HGA gimbal control
- **Thermal Control**
  - Active cooling system control
  - Solar Array position control
- **Power Distribution**
  - Load switching
  - Protected pyro, propulsion & RF power buses
  - Low voltage detection & load shedding
- **Redundancy & Fault Protection**
  - Redundant electronics with the ability to record to both recorders simultaneously from either side
  - Independent system fault monitoring
  - Thermal safe mode attitude control system
- **Low Power Implementation**
- **Mission Duration**
  - 10-year lifetime
- **Radiation Environment**
  - 37KRads TID behind 100 mils Al

Derived Requirements

- **Single Board Computer**
  - 50 MIPS
    - Set by G&C control loop + downlink rate
    - Adjustable to save power
  - 16M Bytes SRAM
    - Code + File structures for 2 large recorders
    - Low power & SEU favors SRAM over SDRAM
- **Solid State Recorder**
  - NAND Flash for low power and high capacity
  - External serial interface enable recording to both SSRs
- **Spacecraft Interface Card**
  - To support instrument DPU and S/C Subsystem interfaces
  - Reduced speed cPCI for reduced power (per TIMED, RBSP)
- **Reliability**
  - Level 2 EEE parts selection in accordance with EEE-INST-002
  - Radiation TID tolerance to meet or exceed 37KRad requirement
  - SEU Tolerant (e.g. TMR logic and EDAC); SEL Immune

Reference Design:
Integrated Electronics Module (IEM)

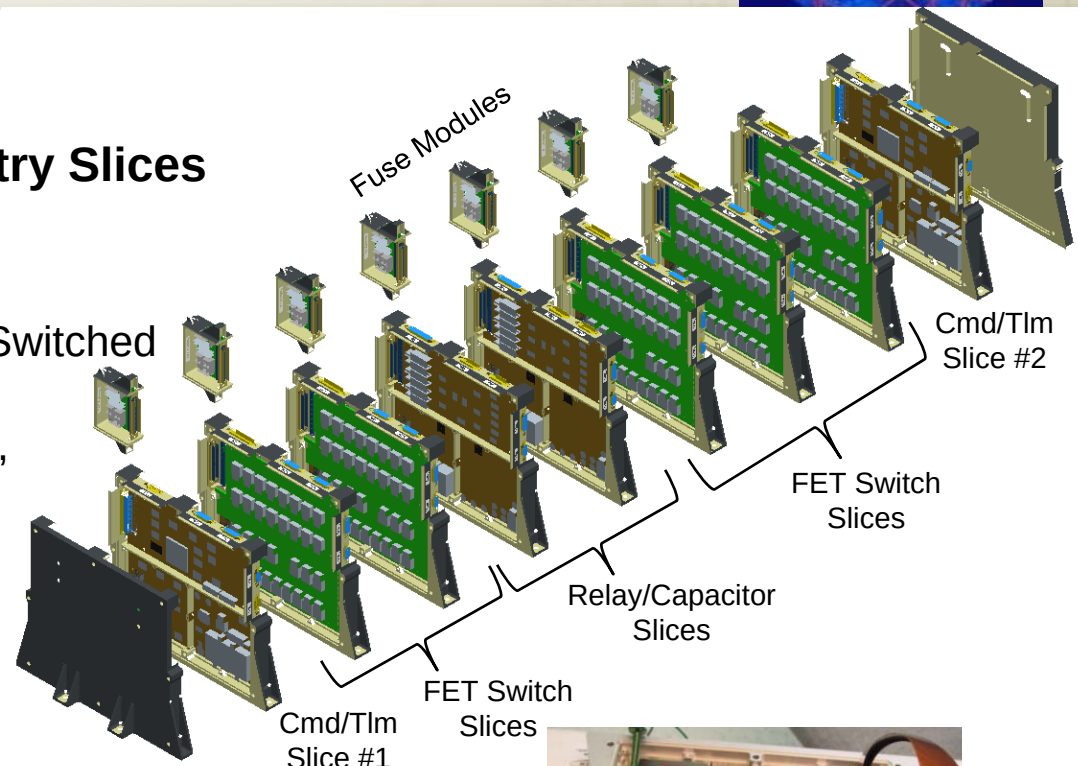
- **Five-Slot 6U cPCI STEREO/MESSENGER Architecture**
- **Single Board Computer**
  - LEON3FT @ 50MIPS
  - 16MBytes SRAM
  - 4MBytes NVRAM
  - Fuse-link Boot PROM
- **Solid-State Recorder**
  - 256GBit Flash Memory
  - EDAC to mitigate SEU and SEFI
  - Redundant External Serial Interface
- **Spacecraft Interface Card**
  - Instrument and Subsystem Interfaces:
    - Serial digital interfaces for command, data & time distribution
  - Interface to transponders
  - On-board system fault protection module
- **Thruster /Actuator Control Card**
  - High-power drivers
  - Analog control and data collection
  - Thermal safe mode attitude and solar array controller



- **DC/DC Converter Card**
  - Provide secondary voltage supplies
  - Switched and unswitched converters
  - 28V nominal S/C power bus input

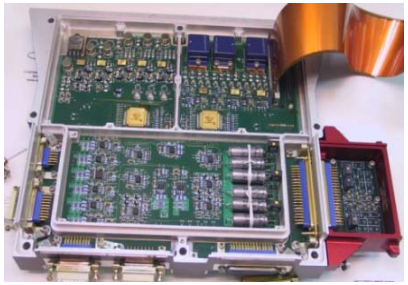
Reference Design:
Power Distribution Unit (PDU)

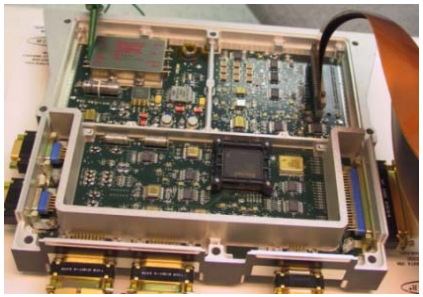
- **Scalable RBSP Slice Design**
- **Redundant Command & Telemetry Slices**
- **Switching Slices**
  - Relay and FET switch types
  - Services provided for Unswitched, Switched and Pulsed Loads
  - Configuration supports 25-switched, 60-pulsed, and 40-switched/pulsed configurable services total
  - Internally Redundant



FET Slice

Relay/Cap Slice



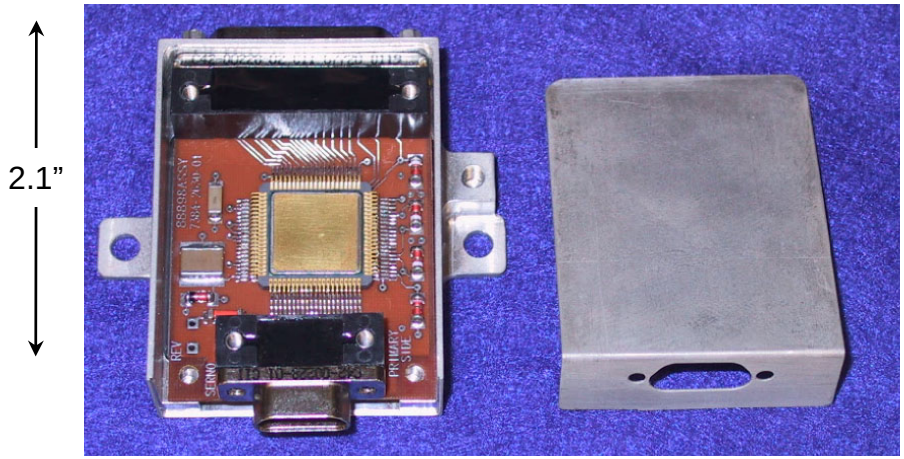


Cmd/Tlm Slice

Reference Design:
Remote Interface Unit (RIU)

- **Built upon APL's RadHard-by-Design Remote I/O (RIO) ASIC**
- **16 A/D Channels Total**
  - 10-bit conversion resolution
  - Temperature sensor monitoring
  - Analog signal monitoring (e.g. pressure transducers, telltales, etc.)
- **I^2C serial digital interface with the IEM**
- **7 units per side (14-total)**

MESSENGER TRIO

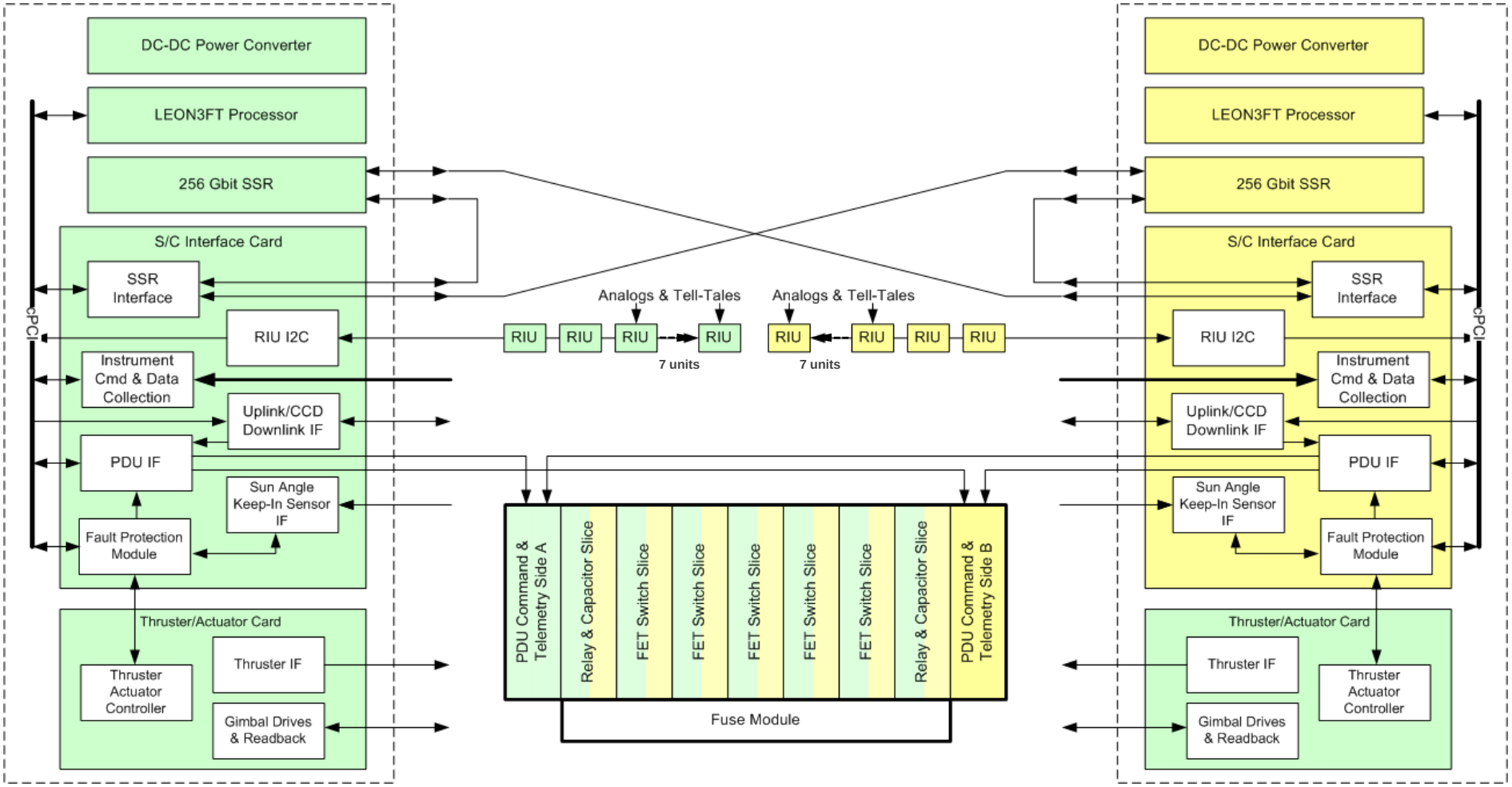


Reference Design:
Fault Protection Module & Thermal Safe Mode ACS

- **Fault Protection Module (FPM)**
  - Monitors discrete signals to detect critical faults or an unplanned processor reset and follows pre-determined hardware sequence fault tree
    - Sends commands to PDU independent from the processor to switch sides, change states, disable self; responding to on-board faults or critical commands from the ground
    - Similar to STEREO hardware reset architecture
  - Located on IEM Interface Card – Separate FPGA implementation

- **Thermal Safe Mode Attitude Control System**
  - Provides simple attitude control using solar horizon sensors and thrusters
  - Commands solar arrays to fixed position
  - Nominally inactive; Fault Protection Module activates during Thermal Safe Mode
  - Similar function performed by NEAR/TIMED Attitude Interface Unit (AIU)
  - Located on the Thruster Actuator Control Card

Redundancy Architecture

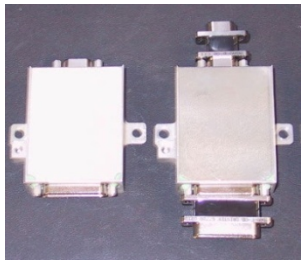


Physical Characteristics

- **Redundant Integrated Electronics Modules**
  - Chassis: STEREO Five card cPCI Housing
  - Power: 16W Active, 4W Standby, +2W SSR (for redundant recording)
  - Mass: 6kg
  - Size (cm): 14.7W x 18.5L x 23.7H

- **Internally Redundant Power Distribution Unit**
  - Chassis: RBSP slice based design
  - Power: 11W Active Side
  - Mass: 14.9kg
  - Size (cm): 25.5W x 24.0L x 21.5H

- **Redundant Remote I/O Units (MESSENGER TRIO Baseline)**
  - Chassis: MESSENGER housing design
  - Power: 43mW
  - Mass: 30g
  - Size (cm): 4.2W x 5.4L x 1.6H



Pre-Phase A Studies/Activities

- **CPU Survey & Risk Reduction**
  - Low power (5W) LEON3 selected over currently available 50MIPS options
  - Risk reduction activity initiated to demonstrate/verify LEON3FT performance

- **Solid-State Recorder Preliminary Implementation Assessment**
  - External Serial Interface selected over Internal cPCI interface as baseline
    - Supports simultaneous recording on both recorders from a single IEM
    - Lower power redundancy approach supporting requisite cross strapping

Phase-A Trades & Studies

- **Single Board Computer Trade**
  - Processor options available at the time, including results of Pre-Phase A LEON3 risk reduction activity
- **Architecture Trades & Studies**
  - S/C Bus Implementation (e.g. If LEON3 is selected, SpW may be preferred)
  - Solid State Recorder
    - Simple Serial Interface to Flash Array or Intelligent SSR with built-in File System
    - Determine appropriate EDAC approach commensurate with memory selection
  - CCSDS Framing in Hardware versus Software
    - Hardware framing could reduce required processor MIPS and power
  - DC/DC Converter 2.5V distribution versus point of load converters, or other options
    - Determine most power-efficient approach to eliminate passive regulation power dissipation
  - Fault Protection Module Implementation – interaction with redundant system, autonomy system, and ground commanding

Benefit of LEON3FT SBC Implementation

- **High-Performance, Low-Power**

STEREO / MESSENGER

PROCESSOR	Throughput	Clock	EEPROM	SRAM	Power
BAE RAD6000	25 MIPS	25 MHz	4-MBytes	8-MBytes	7.3W

RBSP

PROCESSOR	Throughput	Clock	EEPROM	SRAM	Power
BAE RAD750	50MIPS	33 MHz	4-MBytes	16-MBytes	10.8W

SPP

PROCESSOR	Throughput	Clock	EEPROM	SRAM	Power
APL LEON3FT	50 MIPS	66 MHz	4-MBytes	16-MBytes	5W Estimated

LEON3FT Heritage

- **3rd Generation Sparc Processor for space applications**
- **Fault Tolerant (FT) by Gaisler Research for ESA**
- **Extensive field checkout of IP Core in FPGAs**
- **Aeroflex version in Standard Cell ASIC**
  - Fabricated in Aeroflex's 0.25um RadHard ASIC Process
  - Standard Cell Design with No New Cells
  - Radiation and Electrical Testing Complete
- **All Other Board Components Designed or Tested for Space**
  - BAE SRAM
  - Maxwell EEPROM
  - Aeroflex PROM & Logic
  - Actel FPGA

LEON3 SBC Technology Development Plan

SPP Pre-Phase A | **SPP Phase A**

IRAD LEON3 Preliminary Design

- **Small-Sat/CubeSat Application**
- **Preliminary Board Layout Completed**

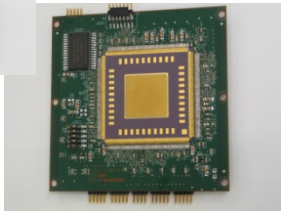
Aeroflex 6U Evaluation Board

SPP 4x4 Technology Demonstration Model

- **LEON3FT Performance Evaluation**
  - **Verify Functionality, Timing, Benchmarks**
  - **Characterize Power, speed, low-power modes**
- **EM grade parts w/ path to Flight**
- **Fabrication Completed/Debug in Process**

SPP/ILN Aeroflex LEON3 Evaluation Board

- **LEON3FT ASIC + Commercial Memories**
- **Software Development Support**
  - **Port Existing Applications**
  - **New S/W Development**
- **Benchmark with flight software & Error Injection**



4x4 Tech Demo Board

ILN 6U cPCI LEON3/SSR Prototype

- **LEON3FT ASIC + EM Grade Parts**
- **8MB EEPROM, 16MB SRAM, 1GB SDRAM**
- **Thermal Performance Testing (w/ TVAC)**
- **TRL-6 by ~ July 2010**

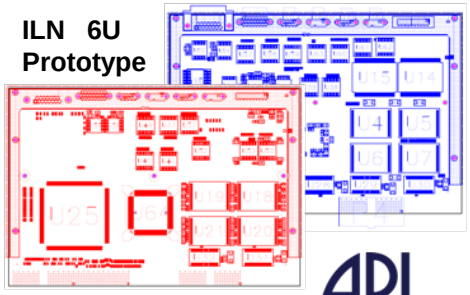
Phase-A LEON3 SBC Evaluation

- **Evaluate TRL Demonstrated by**
  - **4x4 Tech Demo Model**
  - **Aeroflex Evaluation Board**
  - **Other APL Efforts Underway**
  - **External Data**

SPP-Specific LEON3 Prototype

- **Decision to build, based upon evaluation outcome**

ILN 6U Prototype



Solar Probe Plus
Mission Concept Review

Software

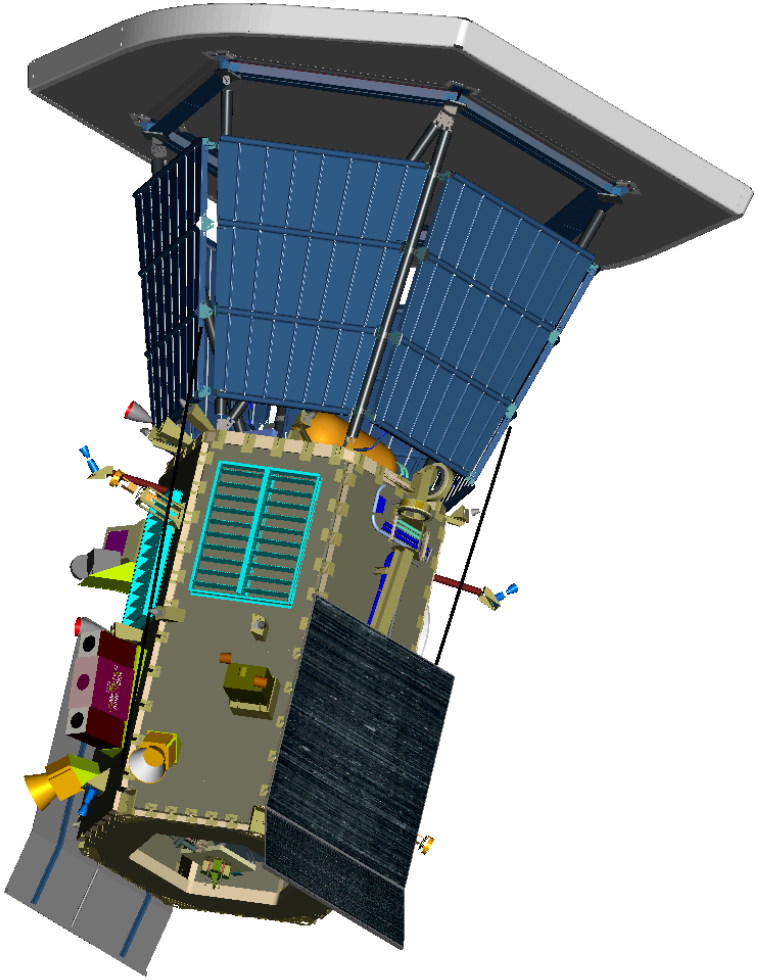
R. Michael Furrow
Mission Software Systems Engineer

The Johns Hopkins University
APPLIED PHYSICS LABORATORY

Solar Probe Plus Software

- Flight Software
- Ground Software
- Testbed Software

Flight Software



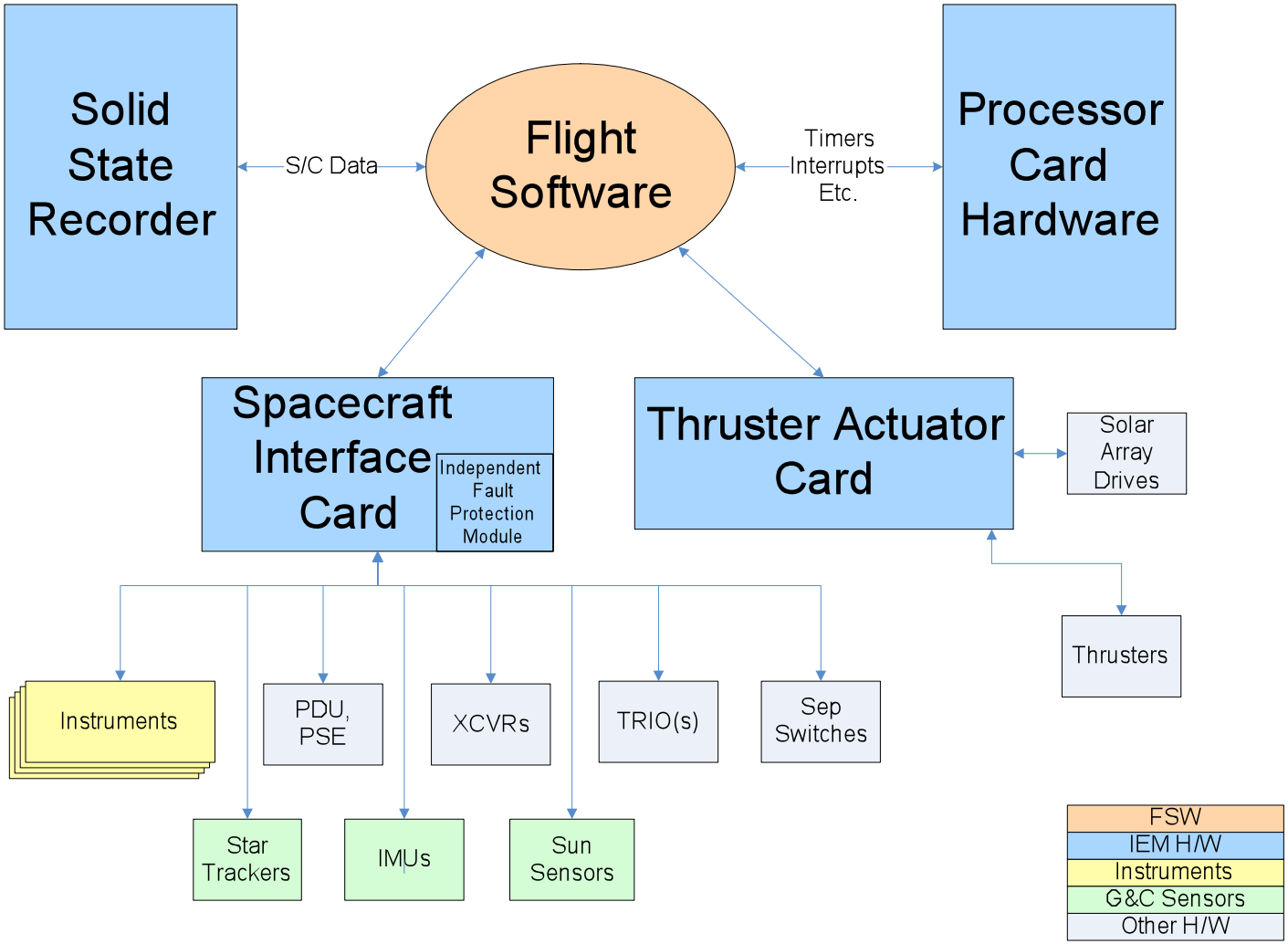
Flight Software Overview

- C&DH/G&C/Autonomy software on the same processor (similar to MESSENGER)
- Utilize Goddard's Core Flight Executive (cFE)
  - APL Space Department's baseline architecture for upcoming missions
  - Currently used on RBSP
  - Based upon software message bus with publish/subscribe architecture
  - Enables reuse of large elements of C&DH code for new missions
  - Goddard's Initial port of cFE to LEON III processer is complete
  - RBSP's scheduler process has been ported to a LEON III and is running under cFE

Flight Software Functionality

- Command management
  - Ground commands received via RF uplink hardware and processed
  - Telecommand packets extracted from transfer frames and distributed to subsystems or acted upon by FSW
  - Commands can be acted on in real-time or stored
  - Support time-tag commands (both spacecraft and instrument)
- Spacecraft subsystem housekeeping
- Science and engineering data management
- Solid State Recorder (SSR) management
- Mission Elapsed Time (MET) management and distribution
- Autonomous fault detection and safing
  - Monitors selected spacecraft and instrument telemetry
- Software management (e.g. memory load/dump/compare, memory scrub)
- Guidance and Control
  - G&C sensor interface management
  - Attitude estimation
  - Attitude control
- Instrument interface management

Flight Software Context



Flight Software Heritage

- Major Reuse from RBSP
  - Goddard's Core Flight Executive (cFE) based architecture
  - Command Ingest
  - Command Manager
  - Time Tags
  - Telemetry Output
  - Housekeeping Telemetry Monitor
  - Autonomy
  - Memory Manager
  - Memory Scrub
  - Core CFDP
- G&C development heritage from MESSENGER, STEREO, & New Horizons, but no direct code re-use
  - Cruise phase & thruster control, 50 Hz attitude control, 1 Hz attitude estimation
  - Three-axis control
  - Autocode generated from MATLAB & Simulink models by G&C analysts

Flight Software
New Development

- Instrument Manager
- CFDP Uplink
- SSR Record
- CFDP SSR Playback
- Attitude Determination
- Attitude Control

Why Use A File System?

- Simplifies Mission Operations
  - Allows data isolation and prioritization for ease of playback and SSR maintance
  - Knowledge of SSR physical layout not required
- Enables the use of CFDP
  - Auto-retransmission of dropped data
    - Allows lower link margins and stepwise downlink rate changes which increase data return without additional burden on the Mission Operations team
    - Using these techniques, MESSENGER observed a > 50% increase in downlink rates for files when compared to a conservative link margin
- Simplifies flight software development
  - Allows use of standard file I/O
- Simplifies ground processing

File System Work to Date

- Integrated filesystem with VxWorks
  - Running on MCP750
  - Using RAM emulation of flash

Flight Software Phase A Activities

- Main processor (w/LEON3 SW risk mitigation)
  - Continued prototyping with commercial LEON3 processors
  - Work with avionics team and breadboard of flight processor to develop board support package and prototype boot code
  - Continued porting of RBSP C&DH flight software prototypes
  - RAD750 fallback plan
- Continued file system prototyping
- FSW requirements definition (including G&C requirements)
- Autonomy and commanding trade
  - Virtual Machine Language (VML)
  - ExecSpec
  - Traditional (Autonomy Rules / Time Tag Commands)
  - Other

Ground Software

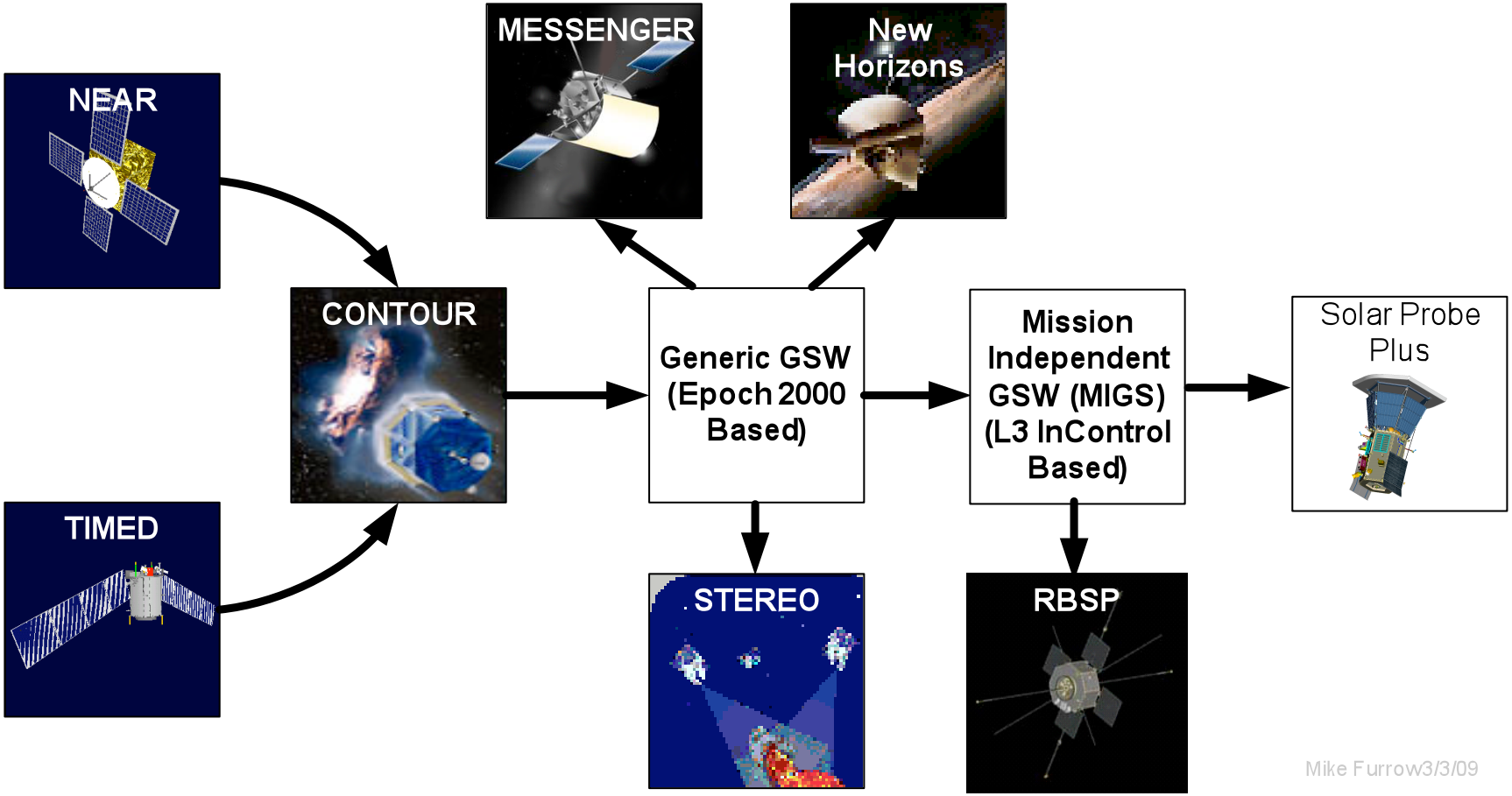
Ground Software Overview

- Ground software supports Subsystem Test, Observatory I&T, Hardware Simulator Control, & Flight Operations Life-Cycle
- Key Functionality includes:
  - Delivery of s/c and instrument commands to observatories
  - Acquisition, Routing, processing, monitoring and archive of real-time telemetry
  - Supports CFDP file downlink and uplink
  - Offline assessment processing, plotting and reporting of s/c hspk tlm
  - Support Interfaces to testbeds, POCs and DSN
  - Provides tools for on-board memory mgmt
  - Timekeeping functions
  - Navigation
- Extensive reuse of RBSP, MESSENGER, New Horizons, and STEREO Ground Software via JHU/APL's Mission Independent Ground Software (MIGS)

MOC Capabilities Supported by Ground Software

- The MOC will be able to assess the health of the spacecraft bus
- The MOC will be able to perform a high-level assessment of Instrument health by monitoring related currents and temperatures in the bus housekeeping.
- The MOC will generate after the fact data products
  - Attitude and orbit
  - Time correlation
- The MOC will support all Mission Simulations during I&T
- CCSDS File Delivery Protocol (CFDP)

Ground System - Heritage



Mike Furrow3/3/09

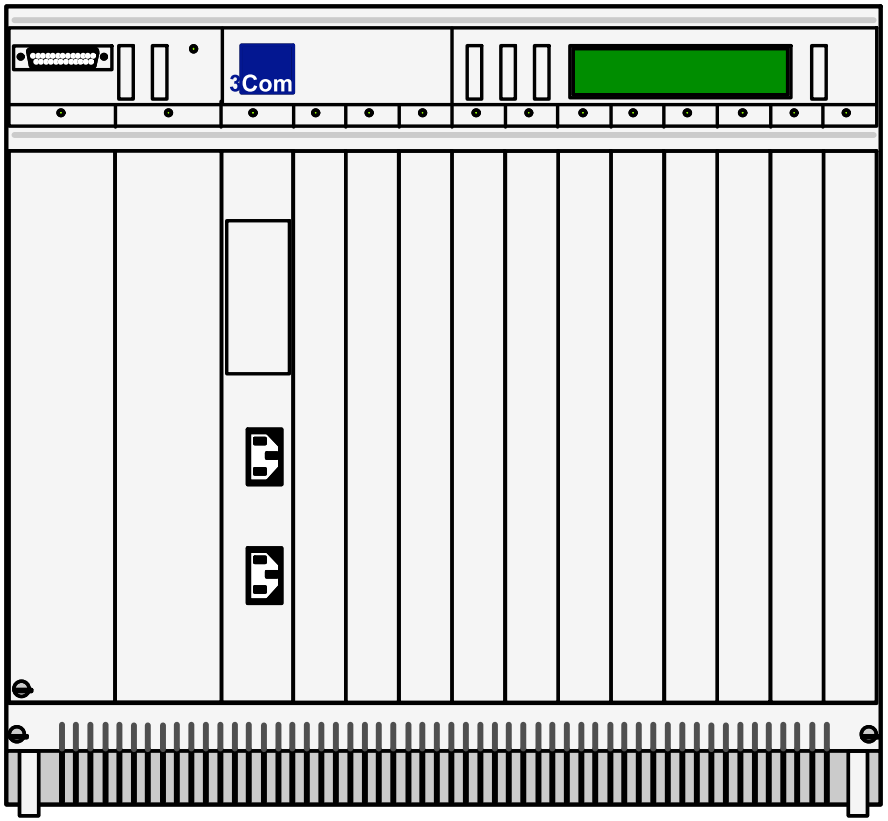
Ground Software – New Tasks

- New for every Mission
  - FSW/Instrument specific modifications for Load Dump & Compare (LDC)
  - Mission specific C/T database
- New for Solar Probe Plus
  - Develop faster than real-time software simulator
  - Develop DSN gateway
  - Develop CFDP uplink

Ground Software Phase A Activities

- Ground system requirements definition
- SW spacecraft simulator make/buy trade

Testbed Software



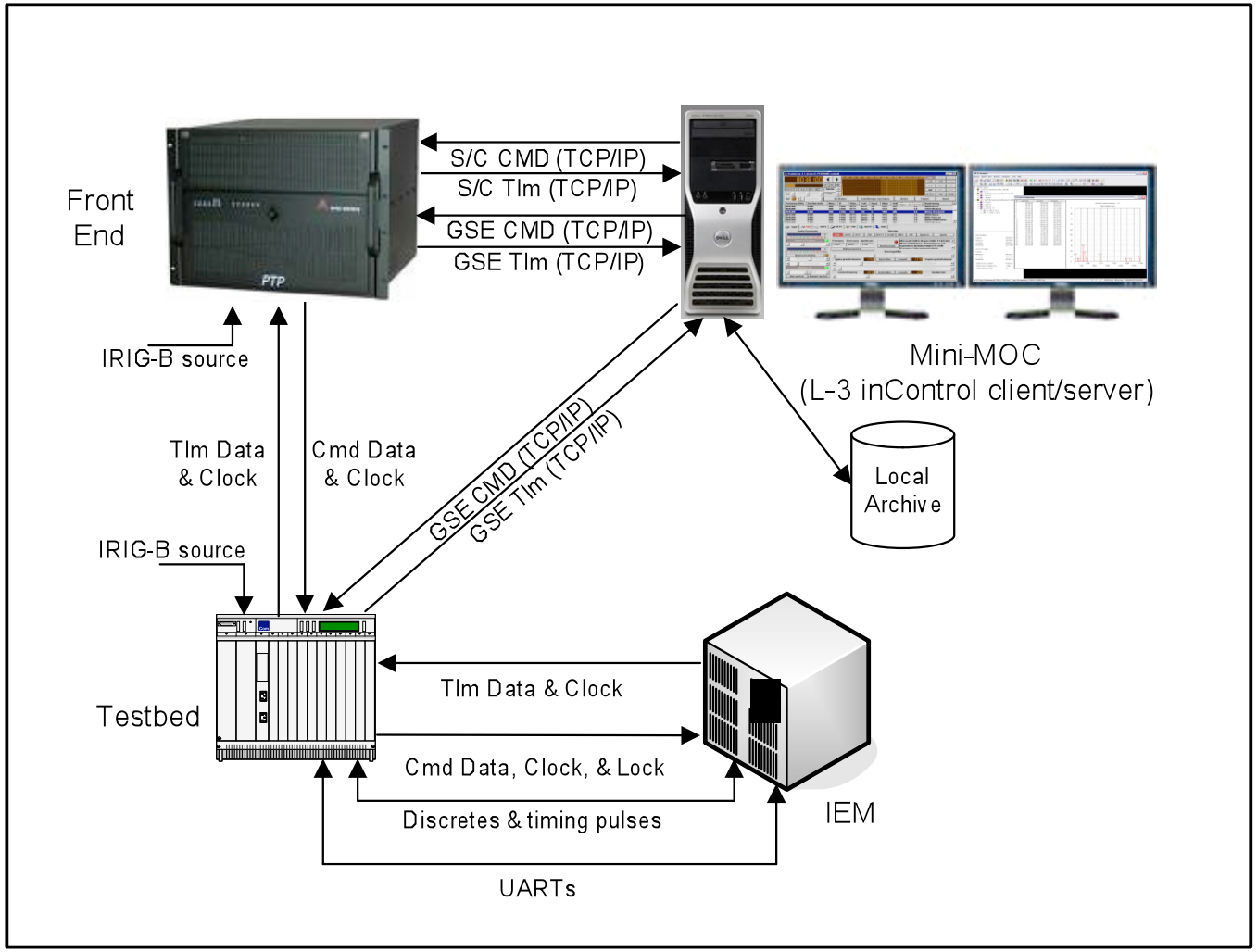
Testbed Software Overview

- Testbed hardware and software emulate S/C subsystems and interfaces connected to the IEMs
  - RF transceiver (uplink and downlink interfaces)
  - G&C sensors and actuators (IMU, star trackers, sun sensors, thrusters, propulsion system)
  - Power (PDU, PSE, battery, solar array drives)
- G&C "truth model" used for G&C testing, MOC simulations
- Power model simulates relay/load activity
- Thermal model simulates temperature monitors (static values)
- Can inject errors for FSW, Fault Management testing
- Commanded through Mini-MOC GSE interface
- Configured through testbed configuration interface

Testbed Software Capabilities

- Based on common core software (TBTK - TestBed ToolKit)
  - Significant heritage from MESSENGER, STEREO, New Horizons, RBSP
  - Upgraded by RBSP to new O/S and new user configuration interface
- Supports multiple configurations and users:
  - IEM hardware checkout
  - IEM hardware flight qualification
  - FSW development and acceptance testing
  - G&C algorithm development and testing
  - Autonomy rule development and testing
  - Spacecraft I&T
  - Mission Operations Hardware Simulator

Testbed Flow Diagram



Testbed Software Phase A Activities

- Testbed requirements definition
- Trade studies:
  - Fidelity of power and thermal models to support system Fault Management, Autonomy testing
  - SoftWare In the Loop (SWIL) concept to reduce cost, increase number of testbeds

Solar Probe Plus
Mission Concept Review

Data Systems

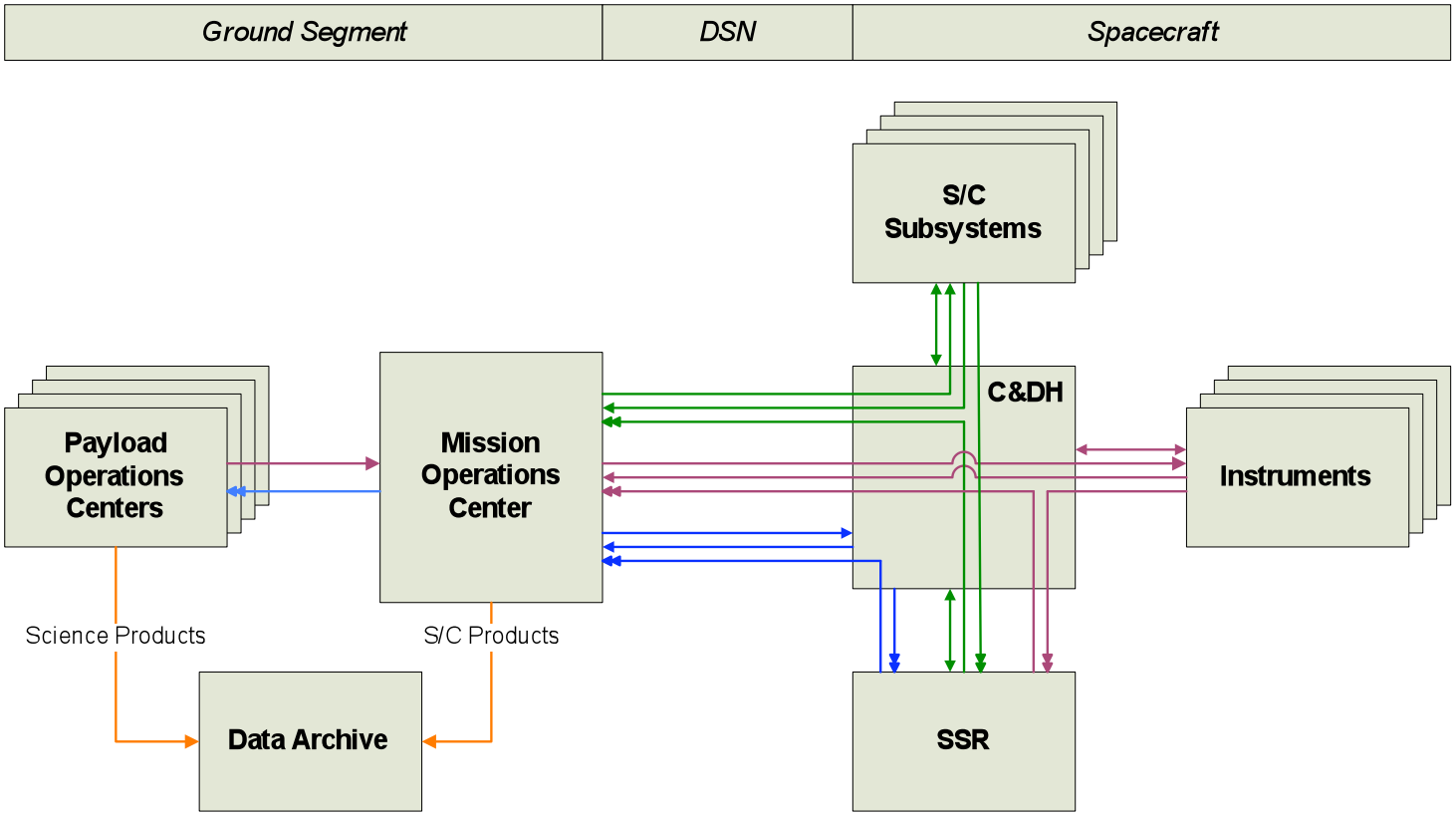
Ken Koontz
Data Systems Engineer

The Johns Hopkins University
APPLIED PHYSICS LABORATORY

What is Data Systems Engineering?

- Systems-oriented view of data analysis and management within the mission
- Crosses multiple disciplines (data avionics hardware, flight software, ground systems, mission operations, communications systems and protocols)
- End-to-end analysis of data volumes, rates, throughputs, latencies, storage throughout a system
- Flow down of functional requirements to data requirements with traceability to Data ICDs
- Establish the test approach for verification of data requirements
- Develop and promote common data handling practices across subsystems
- Ensure data will flow smoothly and system will meet its data requirements

SPP Top-Level Data Flow Diagram



Instruments
- Cmds, Cmd Seqs, Config Data, FSW Updates
- Real-time HSK Tlm (for testing)
- Raw Science, HSK Tlm (SSR record/playback)
- Pre-processed Science, Instrument HSK

C&DH
- Cmds, Cmd Seqs, Config Data, FSW Updates
- HSK, Diagnostic Tlm (real-time)
- HSK, Diagnostic Tlm (SSR record/playback)
- Instrument autonomous cmding /monitoring
- SSR, Subsystem autonomous cmding /monitoring

S/C Subsystems (e.g. G&C, Power, ...)
- Cmds, Cmd Seqs, Config Data
- HSK, Diagnostic Tlm (real-time)
- HSK, Diagnostic Tlm (SSR record/playback)

SPP Top-Level Data Requirements

- 128 Gbits on SSR after each solar encounter
  - Includes science data + S/C engineering HSK + margin

- Return science data from mission's final 3 orbits
  - "Golden" science data within 10-20 Rs
  - Goal: return science data from all solar encounters

Data Systems Engineering in Pre-Phase A

- Participate in Data Avionics weekly meetings
  - SSR flash file system study
  - Reference architecture definition

- Define activities and trade studies for Phase A
- Define scope of activities for Phases B-D
- Work overlap with subsystems

Activities for Phase A

- Define data requirements from functional requirements
- Define list of Data ICDs
- Propose C&T standards for command and telemetry packet formats, database naming conventions
- Develop preliminary models and perform analysis for end-to-end primary data flows
- Develop draft plan for system data requirements verification

Phase A Trade Studies

- Data compression: investigate alternatives to compress science and S/C HSK data on SSR and/or downlink to effectively increase volume and throughput
- Data prioritization: investigate methods to return “interesting” SSR data first (vs. oldest data first)
- CFDP uplink: investigate as alternative to separate MOC-to-S/C subsystem interfaces
- Common instrument interface: define a common instrument-S/C interface for data exchange (dependent on instrument AO selection, DPU interface)
- Common S/C interface: define a common S/C subsystem interface for data exchange
- End-to-end data modeling and analysis: select methodologies and tools to support modeling and analysis across system
- Data requirements and traceability tools: investigate DOORS capabilities in this area

Solar Probe Plus
Mission Concept Review

Ground Systems

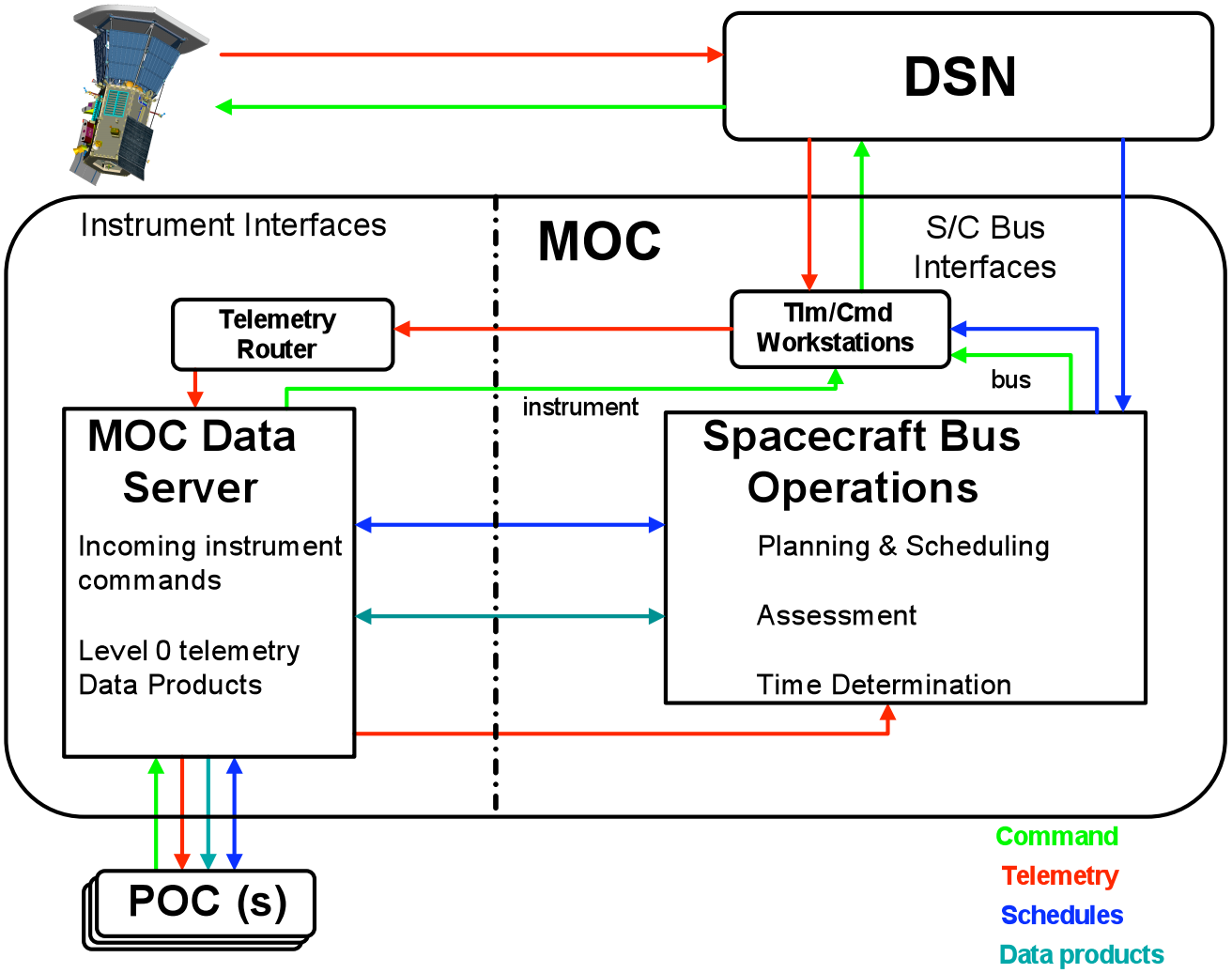
R. Michael Furrow
Mission Software Systems Engineer

The Johns Hopkins University
APPLIED PHYSICS LABORATORY

Ground System Heritage

- Major design reuse from existing missions RBSP, New Horizons, MESSENGER, and STEREO.
- MOC located in new building 21 multi-mission operations center
- Majority of infrastructure already in place
  - Diversely routed NISN communications lines
  - Voice communications

Ground System Data Flow



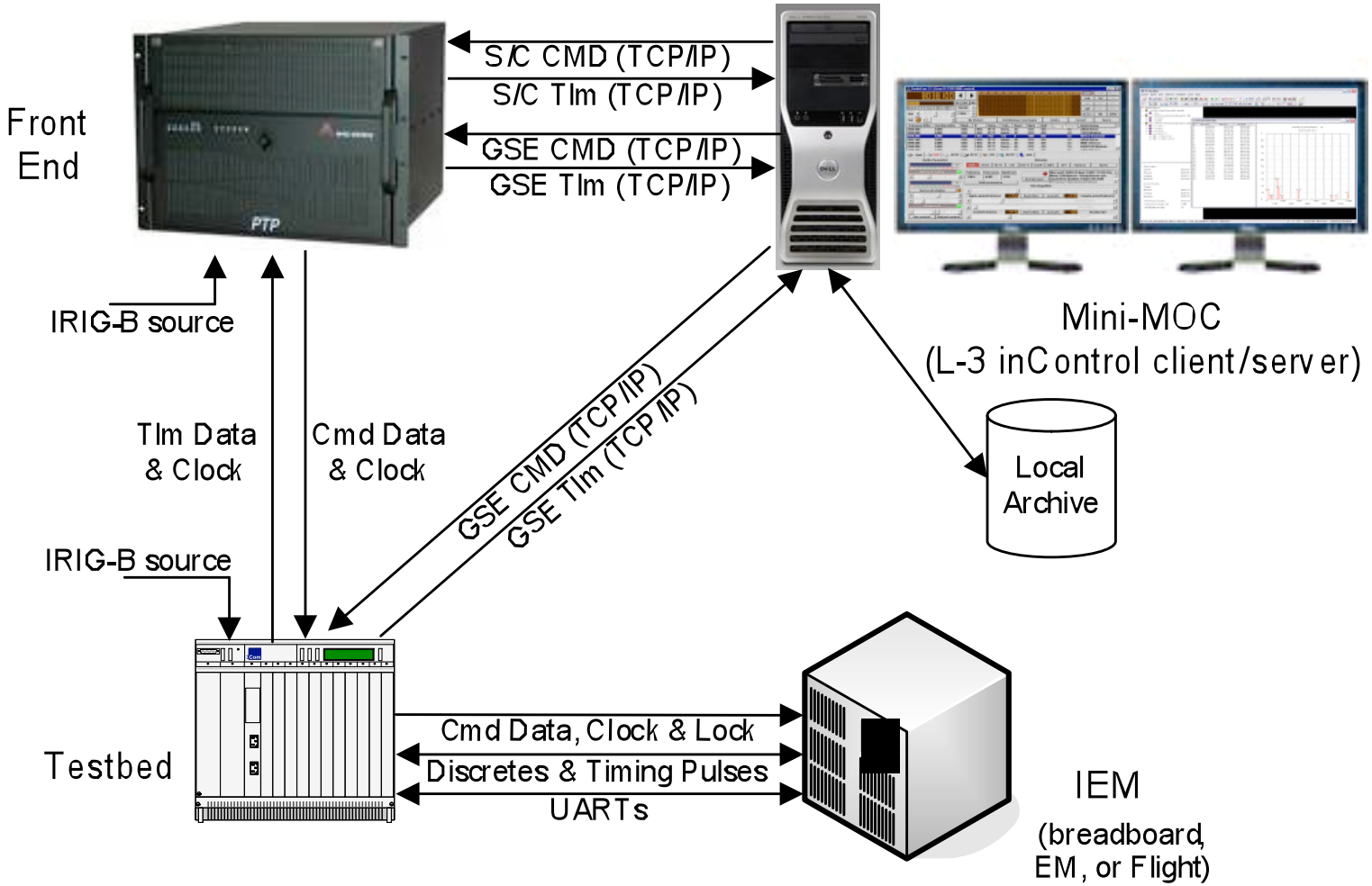
MOC/Mini-MOC Workstation

- The same workstations and software are used for all mission phases
- User Command & Telemetry Interface
  - Mission Operations Center C&T workstations
  - Hardware and Software development testbed C&T workstations
  - Hardware in the loop C&T workstations
- Linux workstations loaded with L3 and APL software.
- The only difference between MOC and Mini-MOC workstations is the configuration.
  - Same exact software
  - Same Command and Telemetry Database
  - Different configuration files and interfaces
  - Each Mini-MOC has its own archive
- Some Mini-MOC workstations will become operational MOC workstations

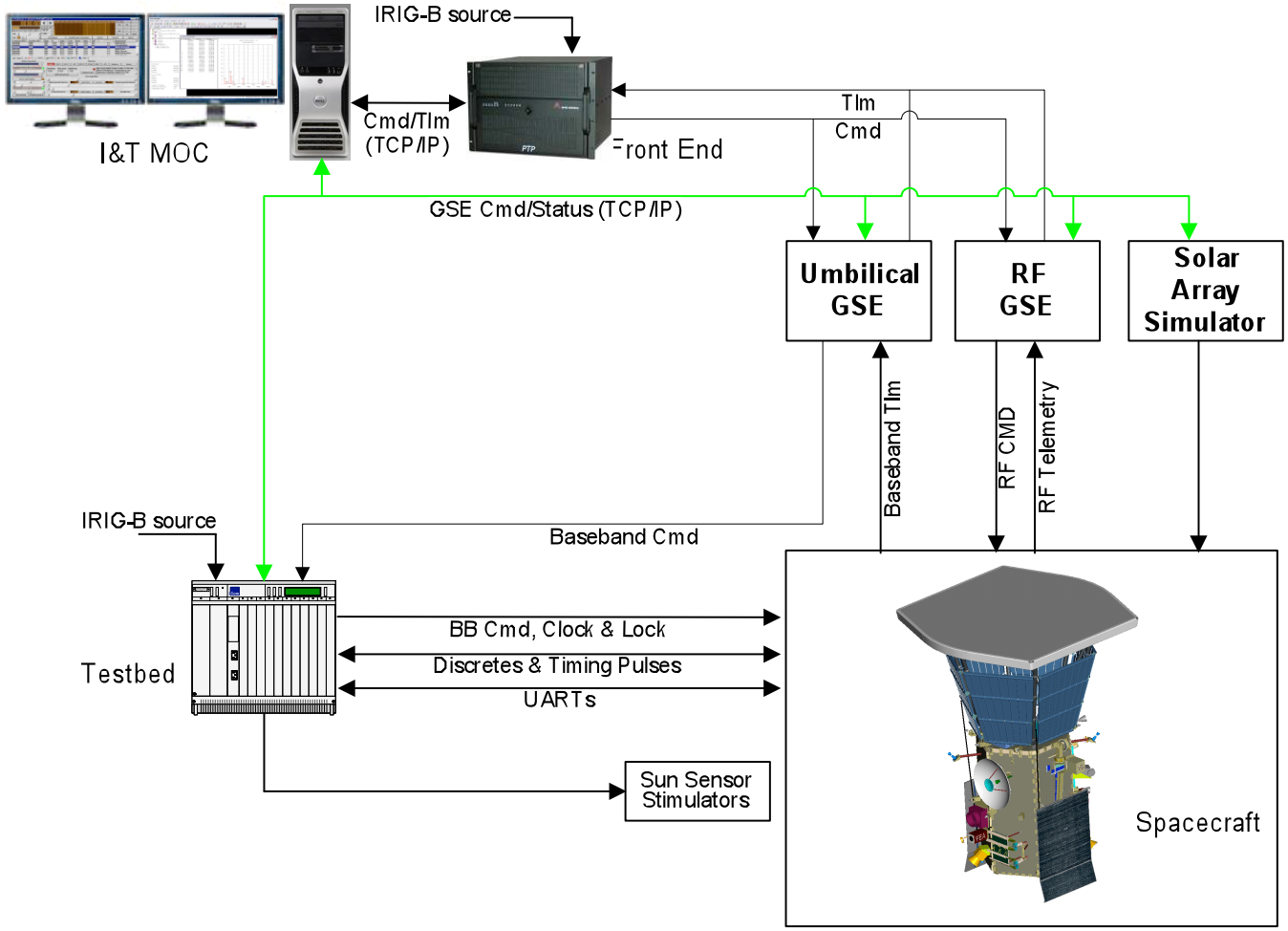
Simulators

- Three Hardware in the loop Simulator Configurations
  - Software Development Simulator (Single String)
  - I&T Simulator
  - Mission Operations Simulator (Dual String)

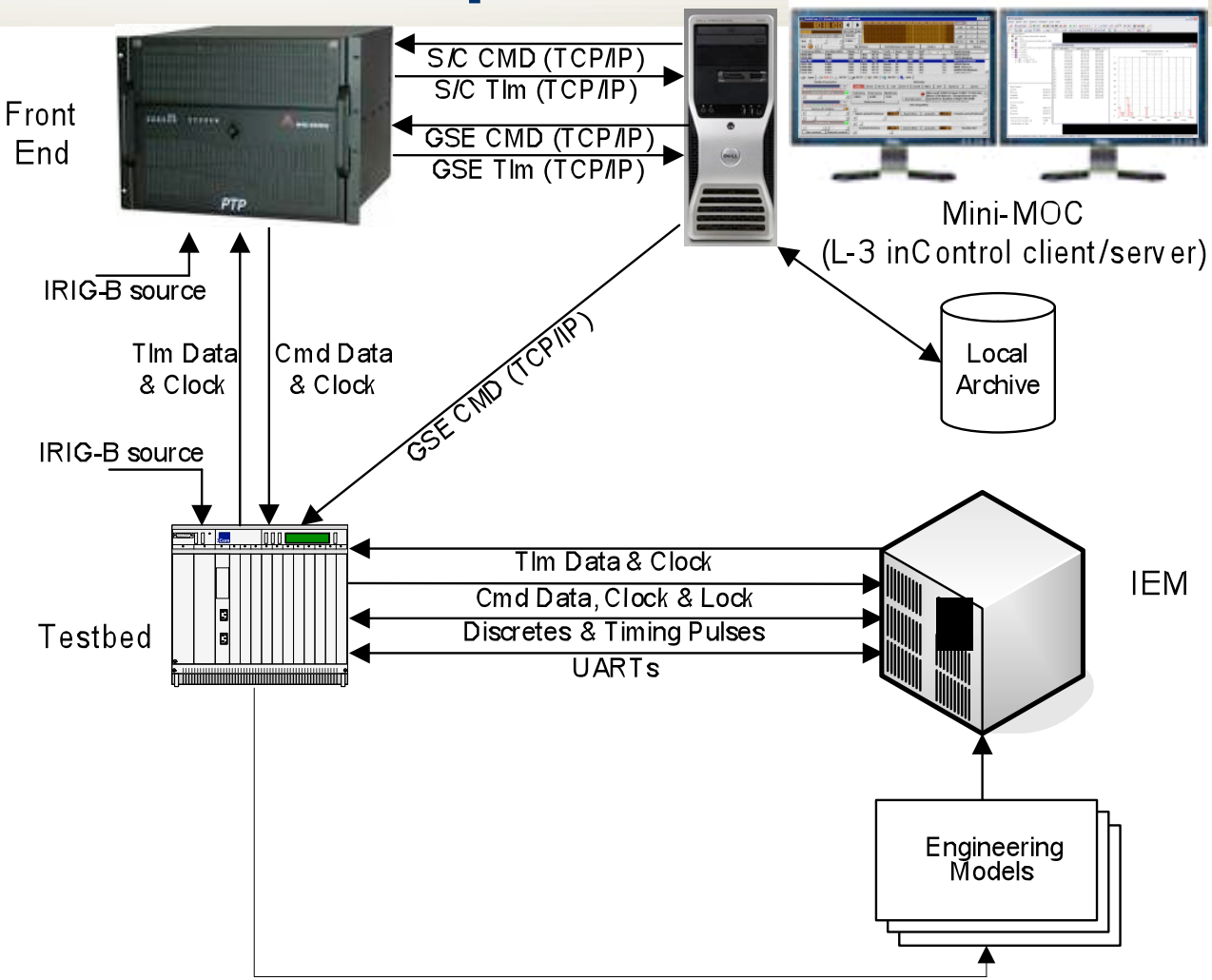
Flight Software Development Simulator



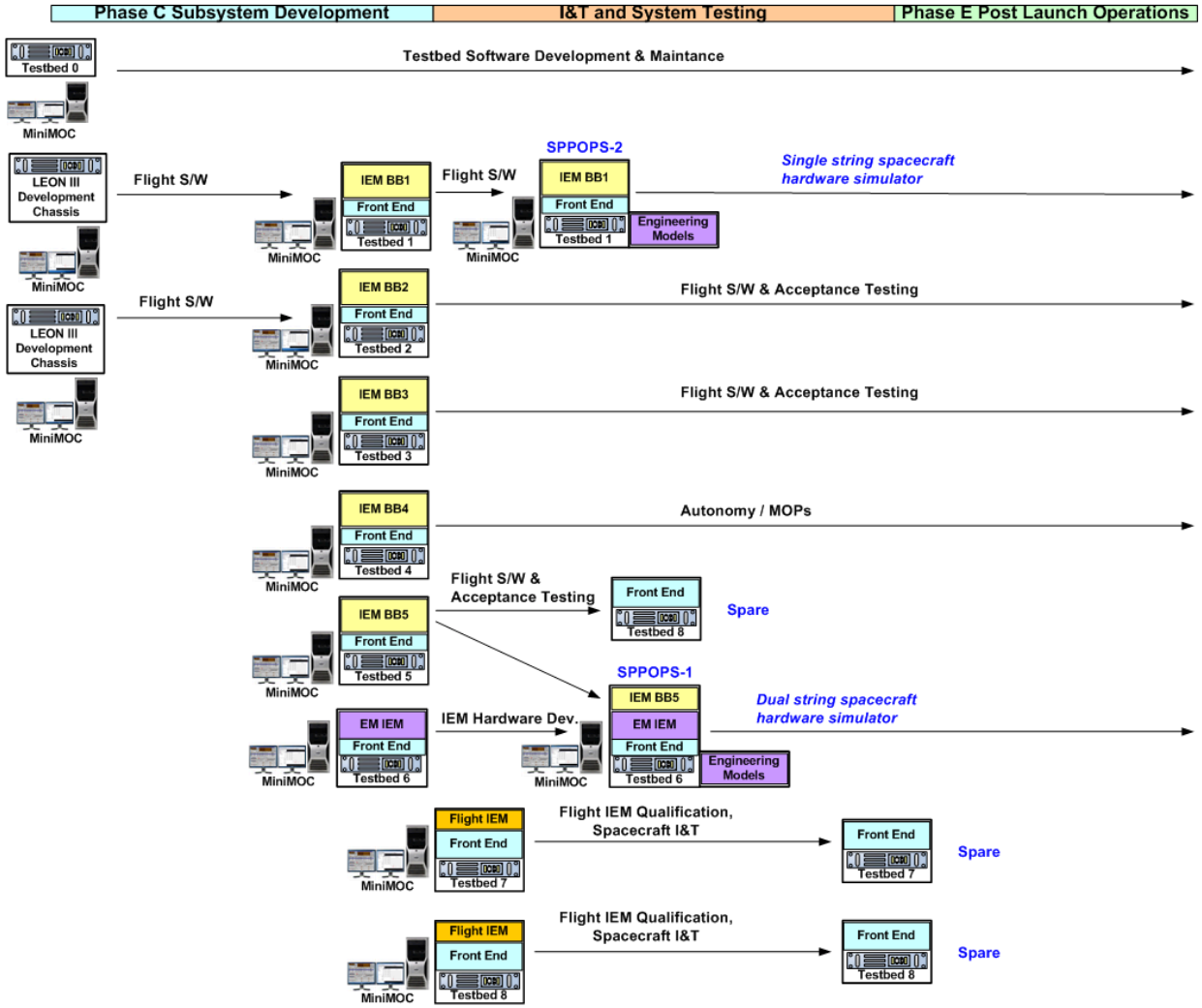
I&T Simulator



Mission Operations Simulator



Testbed Migration Plan



Phase A Activities

- Develop Ground System Requirements
- Refine Network Design

Solar Probe Plus
Mission Concept Review

Integration and Test

Michael J Colby
Lead I&T Engineer

The Johns Hopkins University
APPLIED PHYSICS LABORATORY

Integration and Test Outline

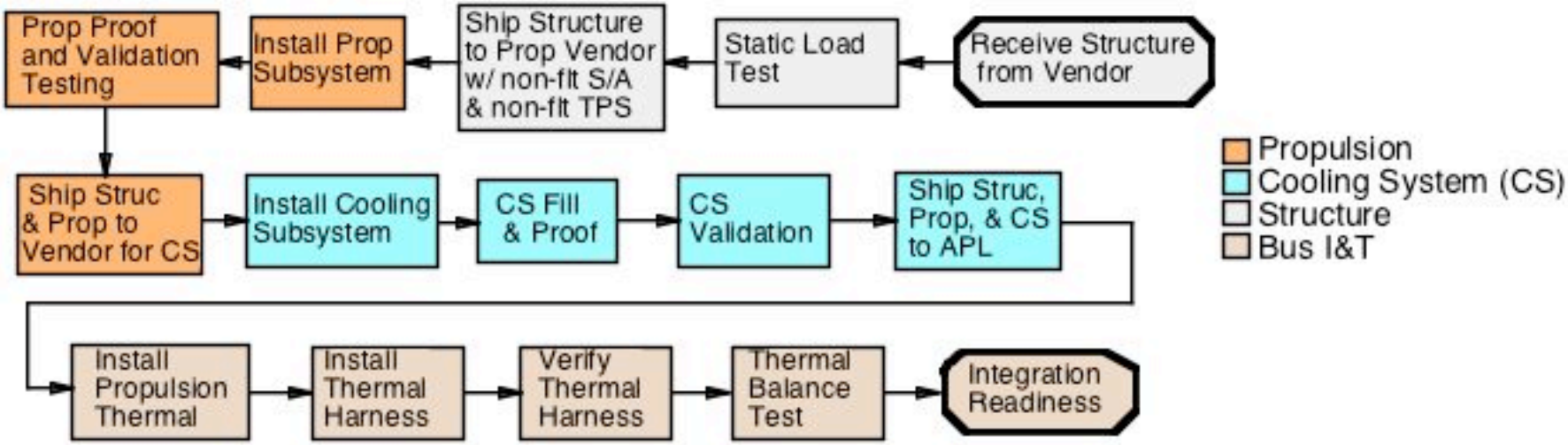
- I&T Overview
- Facility / Equipment Overview
- Phase A plans

- Backup slides
- Harness Overview
- Harness phase A

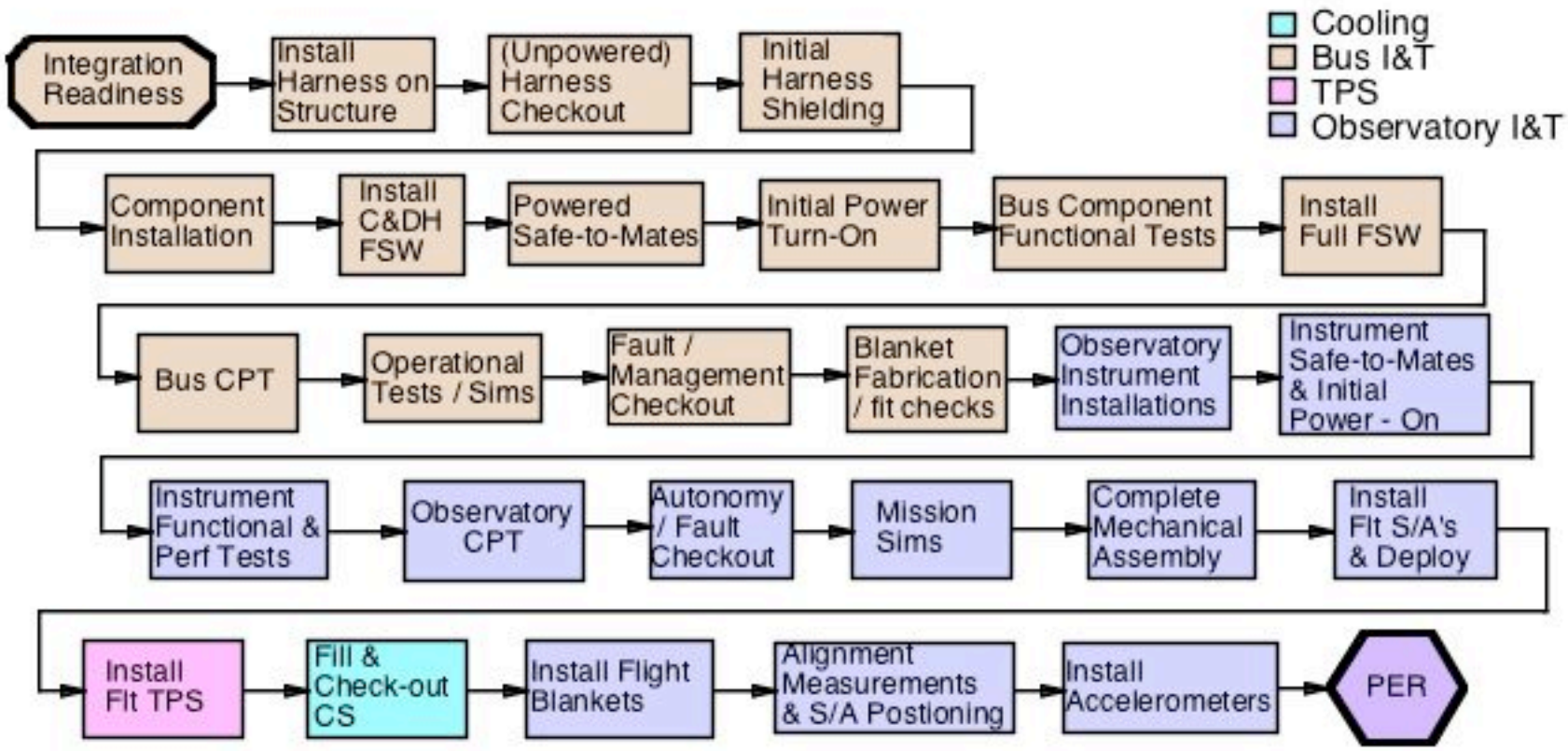
Integration and Test Overview

- Standard flow of activities with SPP additions
  - Cooling system (CS) integration and handling
  - Additional Thermal Protection System (TPS) handling
  - Extra time for combined Solar Array (SA) alignment and positioning
  - Meld Upper Stage (US) schedule similar to New Horizons experience

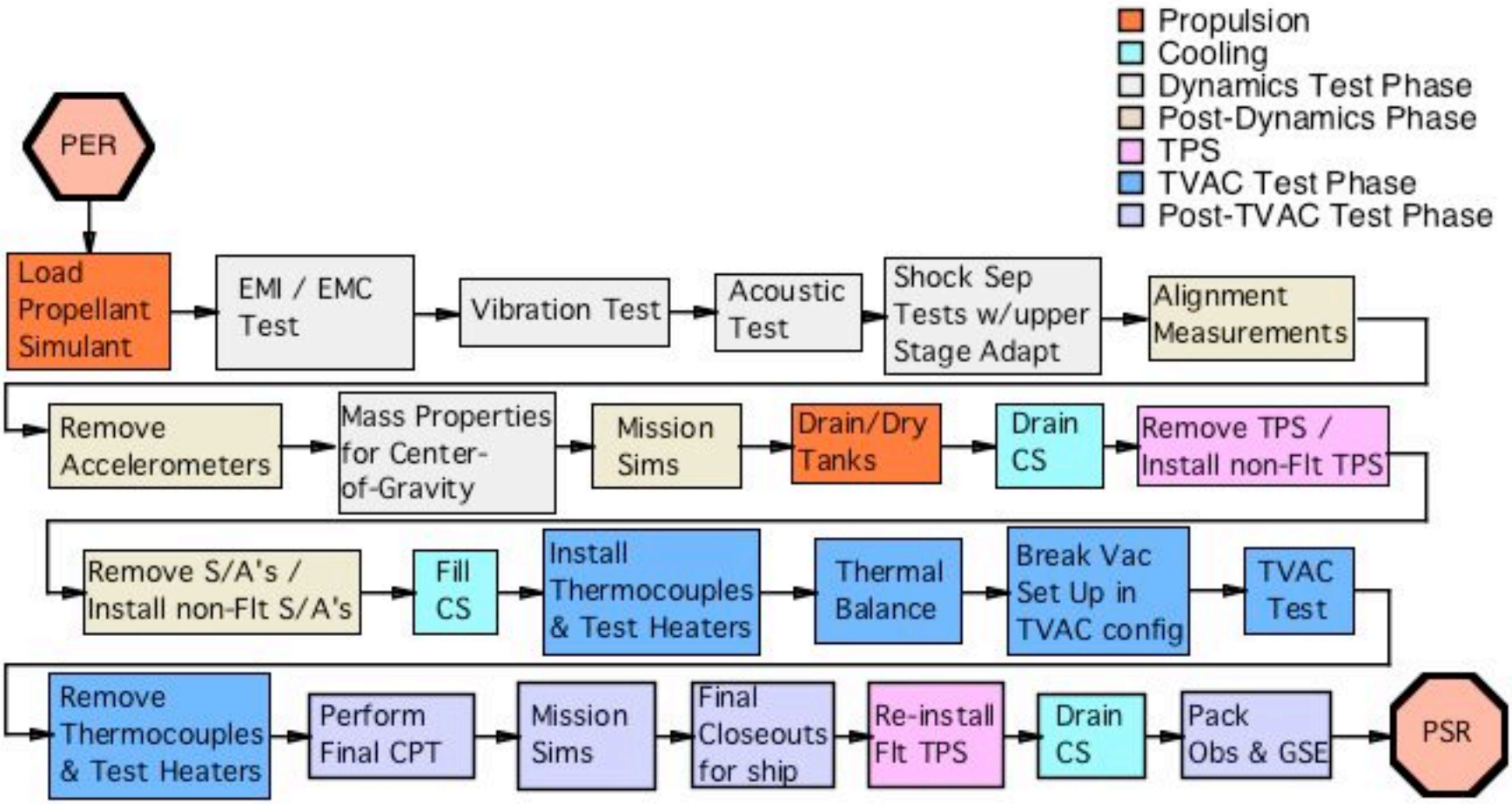
I&T Flow Pre-Integration



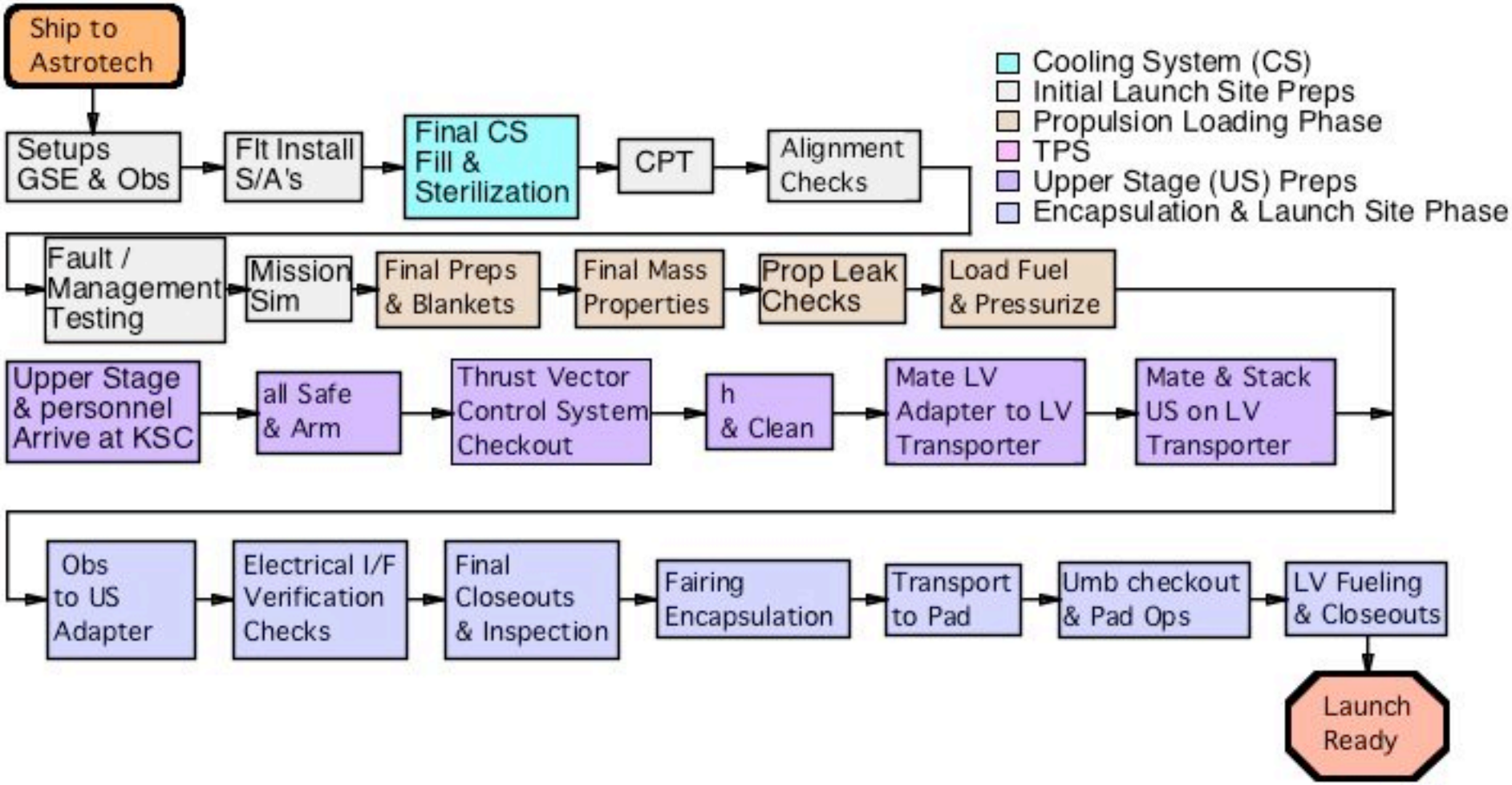
I&T Flow through PER



I&T Flow Environmental Tests



I&T Launch Flow



Facility / Equipment Overview

- L3 In-control S/W planned per RBSP
- All Special Test Equipment (STE) to be provided by subsystem
- Integration and Test through Vibration at APL per recent APL projects
- Acoustics at either APL or GSFC
- TVAC at GSFC
- EMI/EMC at either APL or GSFC
- Launch processing at Astrotech per STEREO & RBSP

I&T Phase A Plans

- Develop a preliminary detailed I&T schedule
  - Costing assumptions
  - Staffing profile
- Develop layout of a System I&T Plan w / reference to the Spacecraft system requirements & initial facility requirements
- Support System Engineering reviews of spacecraft design effort
- Develop SA alignment and full range of motion testing plans
- Develop a TPS Alignment Plan

▪Backup Slides

Harness Overview

- The harness design will follow standard practices as used for RBSP
- The current shielding requirements are similar to RBSP, therefore, SPP would use similar shielding if needed
  - NEPTAPE 1526 is being used for shield wraps on RBSP
  - RBSP is using the Converters Inc 9703 lead tape for outside harnesses with the 3M 9703 adhesive. SPP will probable use the same, if needed

- In addition, Kevlar webbing 328A, 2.5 inch or similar could be used for particle protection of the outside harnesses.

Harness Phase A Plans

- Develop a preliminary S/C harness schedule
  - Costing assumptions
  - Staffing profile
- Develop a preliminary harness development plan
- Develop a long lead parts list

Solar Probe Plus
Mission Concept Review

Fault Management

Michael Trela
Lead Fault Management Engineer

The Johns Hopkins University
APPLIED PHYSICS LABORATORY

Overview

- Fault Management background
- Fault Management development process
- Fault Management Philosophy
  - Mission Design drivers
  - Critical Scenarios
  - Redundancy Philosophy
  - Safing Strategy
- Fault Management architecture trade study summary
- Key Phase A studies

Fault Management Background

- **Fault Management (FM) is defined as the of functional requirements spread throughout the spacecraft and ground that detect, isolate, and recover from events that upset nominal operations**
  - **e.g Hardware, Flight Software, Autonomy, Mission Operations, Ground Software**

- **The goal of FM is to achieve mission reliability objectives without violating program resources.**
  - Fault management must achieve this goal by balancing project risk and the cost of developing, testing, and operating a proposed Fault Management System (FMS)

- **The SPP FMS development will follow the systems engineering FM process used for STEREO and RBSP**

Fault Management Philosophy 1 of 4: Mission Design Impacts

The following aspects of the SPP Mission Design will drive the FMS design:

- **The Solar Probe Plus solar encounter is an extreme thermal environment**
  - In the event of a serious failure which causes loss of attitude control, the FMS must respond quickly to maintain tight spacecraft pointing requirements during the Solar Encounter
- **The Solar Probe Plus space environment varies significantly due to the highly elliptical heliocentric orbit**
  - For spacecraft safing, the FMS must be able to always ascertain the spacecraft-sun distance to keep the spacecraft power positive and thermally safe

To be traded in Phase A:

- **To achieve its principle science goals the Solar Probe Plus mission must return 128 Gbits of data (science, housekeeping, margin) from each of the 3 solar encounters with perihelion distances below 10 Rs**
  - Since ground operations will not be able to resume nominal operations after a "safing" event during solar encounter, the FMS could implement a "fail->safe->operational" strategy.
  - Recovery to operational mode would only occur if on-board monitors indicate a healthy spacecraft (power positive and thermally safe) after TBD hours.*

***The time to recover can be on the time scale of hours due to margin in science collection periods. The encounter period can be defined as the 11 days when the spacecraft-sun distance is <.25 AU. It takes 9 days to generate 128 Gbits of data using the nominal constant science generation rate from the 2008 study.**

Fault Management Philosophy 2 of 4: Critical Scenarios

- Critical scenarios refer to planned mission events or unexpected fault conditions that require a timely system level response to preserve level 1 mission science or the spacecraft itself

- A majority of FMS design is driven to ensure survival in response to a critical scenario

- The SPP preliminary critical scenario list includes:
  - Launch (Launch, Separation, Solar Array Deployment, Initial Cooling System Activation)
  - Sun Keep In (SKI) Violation
  - Spacecraft Over-temperature and/or Solar Array Over-temperature
  - Low Power Condition
  - Extended Loss of Communication
  - Active IEM unhealthy for extended period of time

Fault Management Philosophy 3 of 4: Redundancy Philosophy

- **The SPP spacecraft will employ redundancy to reduce mission risk by minimizing single point failures**
  - Redundancy will not be used where it is technically and programmatically infeasible
  - Above philosophy is in accordance with Class B missions as outlined in NASA NPR 8705.4, *Risk Classification for NASA Payloads*

- **During Phase A design, numerous system reliability trade studies will be performed to:**
  - Validate and improve engineering designs via fault tree analysis
  - Identify options for functional redundancy
  - Show how cross-strapping can be used to increase reliability
  - Minimize single point failures

- **Current spacecraft reference design has near full block level redundancy**
  - Non redundant block level components include:
    - HGA Gimbal
    - Single Cooling Loop system with selective redundancy
    - Any single Solar Array Drive Actuator

Fault Management Philosophy 4 of 4: Safing Strategy – Fault Responses

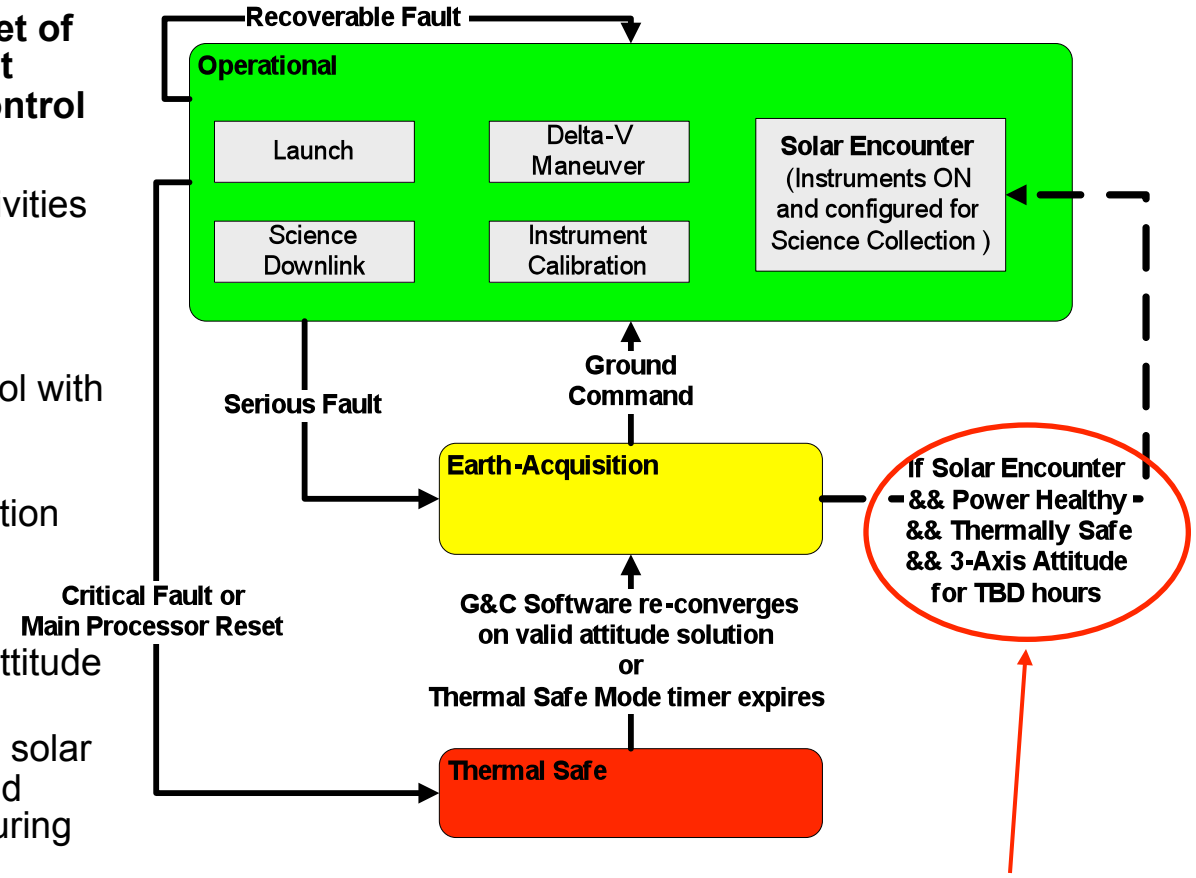
The FMS system will contain three levels of response

- Reduces risk by managing design complexity

- **Level 1 (Recoverable Failure Scenarios):**
  - Indication of failures are easily identified on-board
  - Autonomous isolation and recovery does not impact nominal operations (i.e. no spacecraft safing)
- **Level 2 (Serious Failure Scenarios):**
  - FMS can attribute the failure indications to a small set of probable causes
  - Autonomous isolation and recovery requires a spacecraft safing event
- **Level 3 (Critical Failure Scenarios):**
  - Main Processor resets (i.e. G&C algorithm no longer controlling attitude)
  - Critical safety limit has been violated (i.e. acts as a safety net as earlier attempts to correct the fault have failed)

Fault Management Philosophy 4 of 4: Safing Strategy - Operating Modes

- **The Safing Strategy define the set of objectives the Fault Management System uses to guide system control**
- **Operational Mode:**
  - Support time-tagged MOPS activities
- **Earth-Acquisition:**
  - Maintain a thermally safe 2-axis attitude
  - Active solar array pointing control with positive power generation
  - Configure RF to maximize the potential for ground communication
- **Thermal Safe:**
  - Temporary mode (1-2 minutes)
  - Maintain thermally safe 2-axis attitude during
  - Regardless of orbit location, the solar arrays are commanded to a fixed position that is thermally safe during solar encounter
  - Non-critical loads are powered off

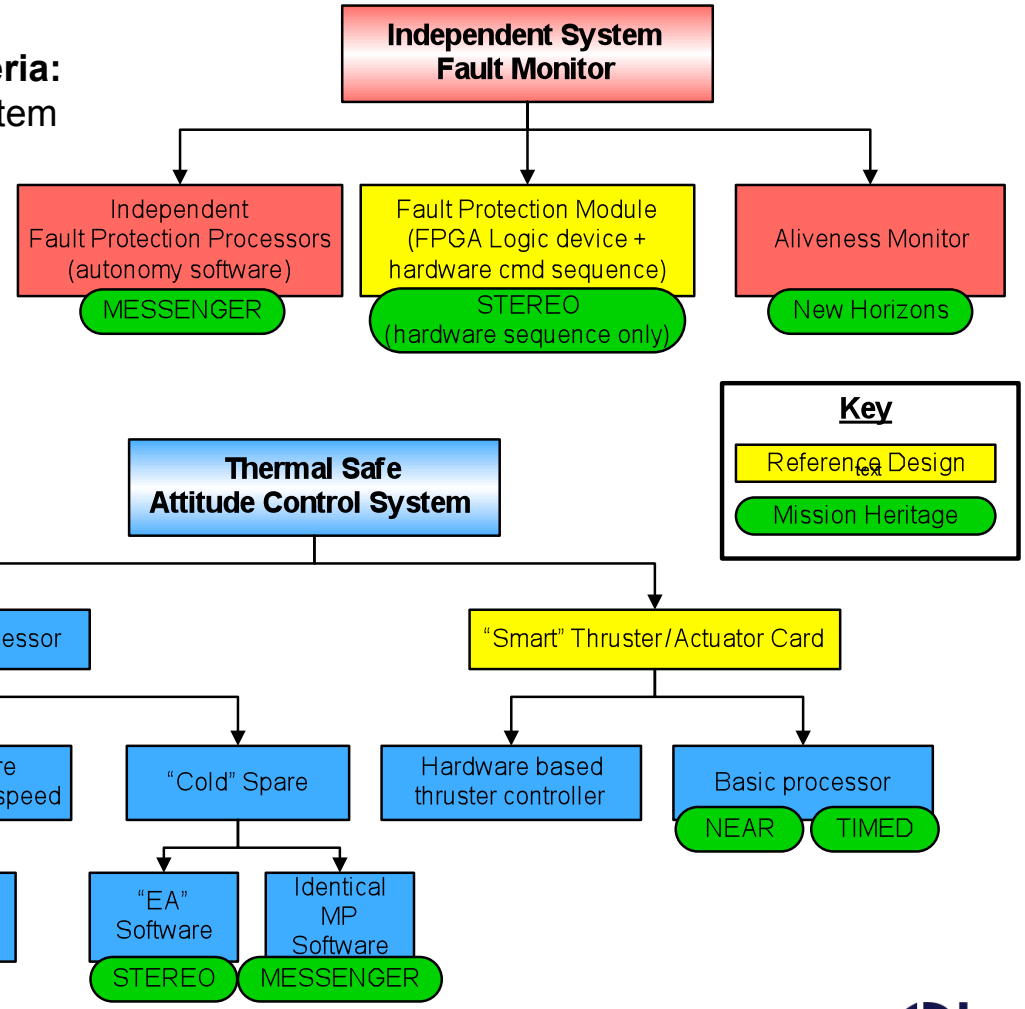


Recovery to be traded in Phase A

Fault Management Architecture Trade Study

Architectures evaluated on the following criteria:

- Time without an active attitude control system
- Independent system level fault monitoring
- Required power
- Interface complexity
- Ease of testability
- Required software/hardware development efforts
- APL heritage



Key Phase A Analysis

During Phase A Fault Management will participate in the following key analysis:

- Launch sequence
- Better define necessary response times and pointing accuracy needed for SKI violations
- Work with Autonomy Lead in identifying required Autonomy Architecture
- Detailed Fault Protection Module definition
  - Detailed Thermal Safe Mode definition
  - Detailed definition of architecture to maintain tight pointing requirements
- Reliability trade studies (FMEA, PRA)
- Solar Array control
- Detailed cooling loop control requirements
- Detailed Solar Horizon Sensor definition
- Fault Management System testing requirements

Solar Probe Plus
Mission Concept Review

Concept of Operations

Nick Pinkine
Mission Operations Manager

The Johns Hopkins University
APPLIED PHYSICS LABORATORY

Outline

- Operations Concept Key Features
- Spacecraft & Mission Design Drivers/Operational Impacts
- Mission Operations Interfaces
- Orbit Planning & Operations Architecture
- Science Data Downlink Strategy
- Early Operations Timeline
- Phase-A Plans

Operations Concept Key Features

- **Decoupled Spacecraft and Instrument operations**
  - Extensive heritage from TIMED & STEREO operations
- **24 orbits with repeatable events from orbit to orbit**
  - Facilitates mission planning & allows reusable command constructs
  - High priority science data downlink periods each orbit
- **7 Venus fly-bys and multiple TCMs**
  - Requires detailed maneuver planning & execution process
- **Primary science data taken during solar encounter phases**
  - Condensed, contiguous time period for instrument operations
  - Simplifies mission planning & payload operations
- **Strong heritage in operations architecture and processes**
  - Planning, Control & Assessment architecture & software tools inherited from TIMED & STEREO

Operations Concept Drivers

- **No requirement to change spacecraft attitude during solar encounter science operations**
  - Enables decoupled spacecraft & payload operations
- **Sufficient power for all instruments during solar encounter phase**
  - No duty cycle requirements on instruments
  - Enables decoupled spacecraft & payload operations
- **7 Venus fly-bys and multiple TCMs**
  - Requires mission design & navigation team interfaces
  - Requires detailed maneuver planning & execution process
  - No payload operations during Fly-Bys

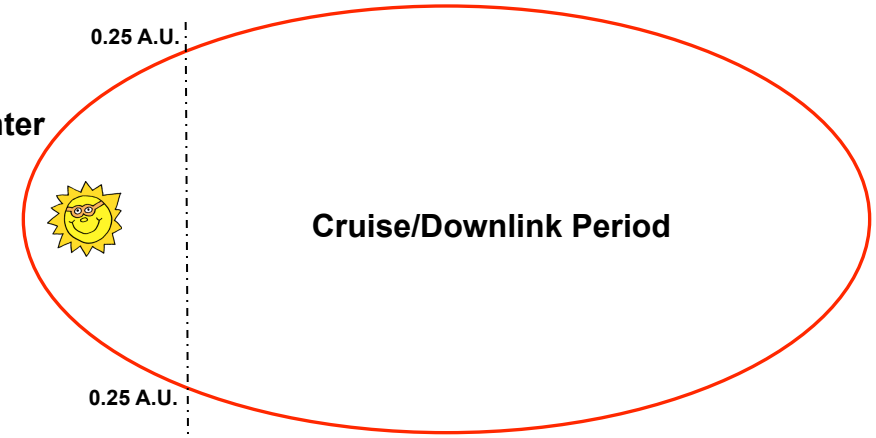
Operations Concept Drivers (cont'd)

- **HGA Usage Constraints**
  - Power/Thermal and SPE angle constraints limit HGA use
  - Results in high priority DSN utilization periods each orbit to support science data downlink
- **128 Gbits of data collected each orbit**
  - Need to model data volume collected each orbit and impose maximum data volume allocation for instruments & housekeeping
  - Develop data collection and downlink strategy to maximize data return
- **SSR file system and CFDP downlink capability**
  - Simplifies SSR management. Provides robust method for retrieving missing data.
  - Implemented successfully on MESSENGER

Operations Concept Drivers (cont'd)

- **Dual array design changed to combined array**
  - Simplifies solar array operations
  - Requires array calibration and array angle management strategy
- **No reaction wheels**
  - No spacecraft momentum management required
- **No optical navigation requirements**
  - Simplifies instrument, navigation and ground system operations

Orbital Operations Planning Concept



24 Solar Encounter Orbits

Orbital Periods Vary (169 days to 88 days)

Solar Encounter Period

Encounter Phase

- Primary science data collection phase – All instruments can be powered on
- LGA periodically available for communications & Nav
- Real-time commanding supported but not nominally planned

Cruise/Downlink Period

Cruise Phase

- All instruments nominally powered off
- LGA for communications – H/K data only
- Commanding as needed to support spacecraft maintenance

Science Downlink Phase

- All instruments nominally powered off
- HGA for communications – SSR playbacks
- Commanding as needed to support spacecraft maintenance

Orbit Planning Activities

- Detailed operations planning performed for each orbit
  - 24 total orbits (Orbital period varies between 169 - 88 days)

- Key operations planning required for each orbit include:
  - Venus Fly-By Events
  - TCMs
  - Spacecraft Slews
  - HGA downlink opportunities
    - High priority downlink periods each orbit
    - Dictates SSR management scheme
    - Slews will be required for some downlinks
  - Spacecraft housekeeping & maintenance activities
    - Flight software loads Command Sequence Uplinks
    - Autonomy loads Parameter maintenance
    - Special sub-system and/or payload tests
  - Eclipse & solar conjunction periods
  - Doppler range & Delta-DOR tracking requirements
  - Solar encounter operations (POCs)

- Orbital operations template created to capture and schedule these activities

Orbital Operations Template

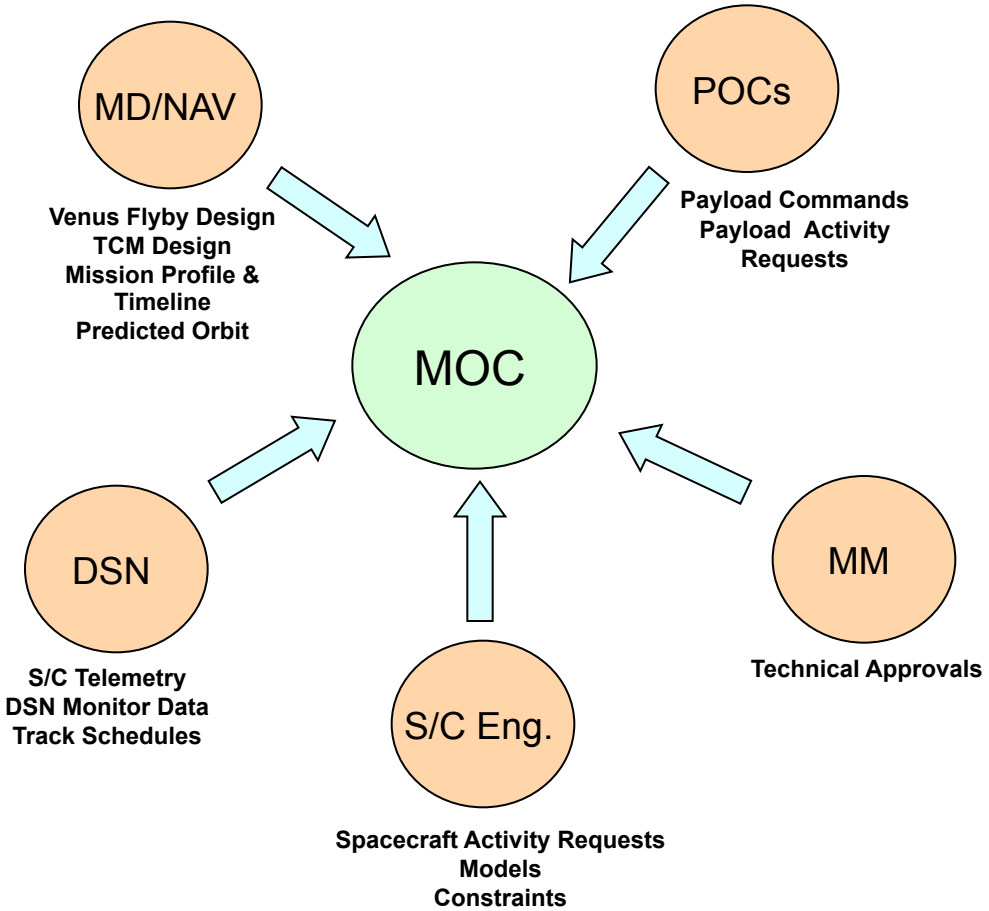
Activity	Week (P-9 to P-7)	Week (P-7 to P-5)	Week (P-5 to P-3)	Week (P-3 to P-1)	Week (P-1 to P+1)	Week (P+1 to P+3)	Week (P+3 to P+5)
S/C Phase	Cruise		Science D/L	Cruise	Encounter	Cruise	
Spacecraft Housekeeping	Latch Valve Test (99)		Instr. S/W Load (136)		G&C Ephem Load (152)		Load Tlm Tables (178)
SSR Playbacks	TBD frequency & duration		SSR Playbacks (10 hrs/day) (125 – 139)				
Delta- DORs							
TCMs	TCM # 5 (101)						
Venus Fly-bys		Venus Flyby # 3 (115)					
Solar Encounter					Science Data Collection (156 – 167)		
Eclipse Period						(169 1.5 hrs)	
Solar Conjunction					SEP < 1.0 deg for 12 hrs (155)		
DSN Contacts	TCM & Flyby DSN supports (100 – 120) LGA		SSR Playbacks (10 hrs/day) (125 – 139) HGA			Contacts for H/K data & NAV (156 – 157) LGA	

P = Perihelion

Note – Numbers in parentheses denote DOY for illustrative purposes only

Mission Operations Interfaces

Detailed Mission Ops Interfaces still need to be developed



Orbit Planning & Operations Architecture

SPP Orbit Planning

Orbit Planning Team (Designee from each team)

- MD/NAV
- POCs
- MOC
- DSN
- S/C Eng
- MM

Orbital Template Development

- Venus Fly-Bys
- Solar Encounter Periods
- TCMs
- HGA Downlink Periods
- S/C Mode Changes
- Eclipse Periods
- Solar Conjunctions
- Prelim DSN Contacts
- Special S/C or Payload Activities
- Flight S/W Load
- G&C Test
- Instrument Calibration
- Autonomy Load

(Distributed by MOM)

Orbit Templates

- MD/NAV
- POCs
- DSN
- S/C Eng
- MM

Orbit Templates

(Distributed by MOM)

MOPS Inputs

SPP Operations Architecture *

Mission Operations Team

Planning	Control	Assessment
Track Scheduling	R/T Telemetry Processing	Trending
Maneuver Planning	SSR Playbacks	Anomaly Investigation
SSR Management	R/T Anomaly Investigation	MOPS Databases
Maintenance Events	Command Uplinks	Simulation Validation
Cmd Sequence Builds	R/T Telemetry Displays	
Special Events	Unattended Ops	
Procedure Development		
Instrument Queue Management		

* Strong heritage from TIMED/STEREO/RBSP operations architecture

APL

Orbit Planning & Operations Architecture

- Unique orbital operations template is developed for each orbit
  - Orbit planning team provides inputs for scheduling key events
  - Mission operations manager distributes finalized orbit templates to various teams
  - Template serves as master reference for generating & delivering products used by mission operations planning

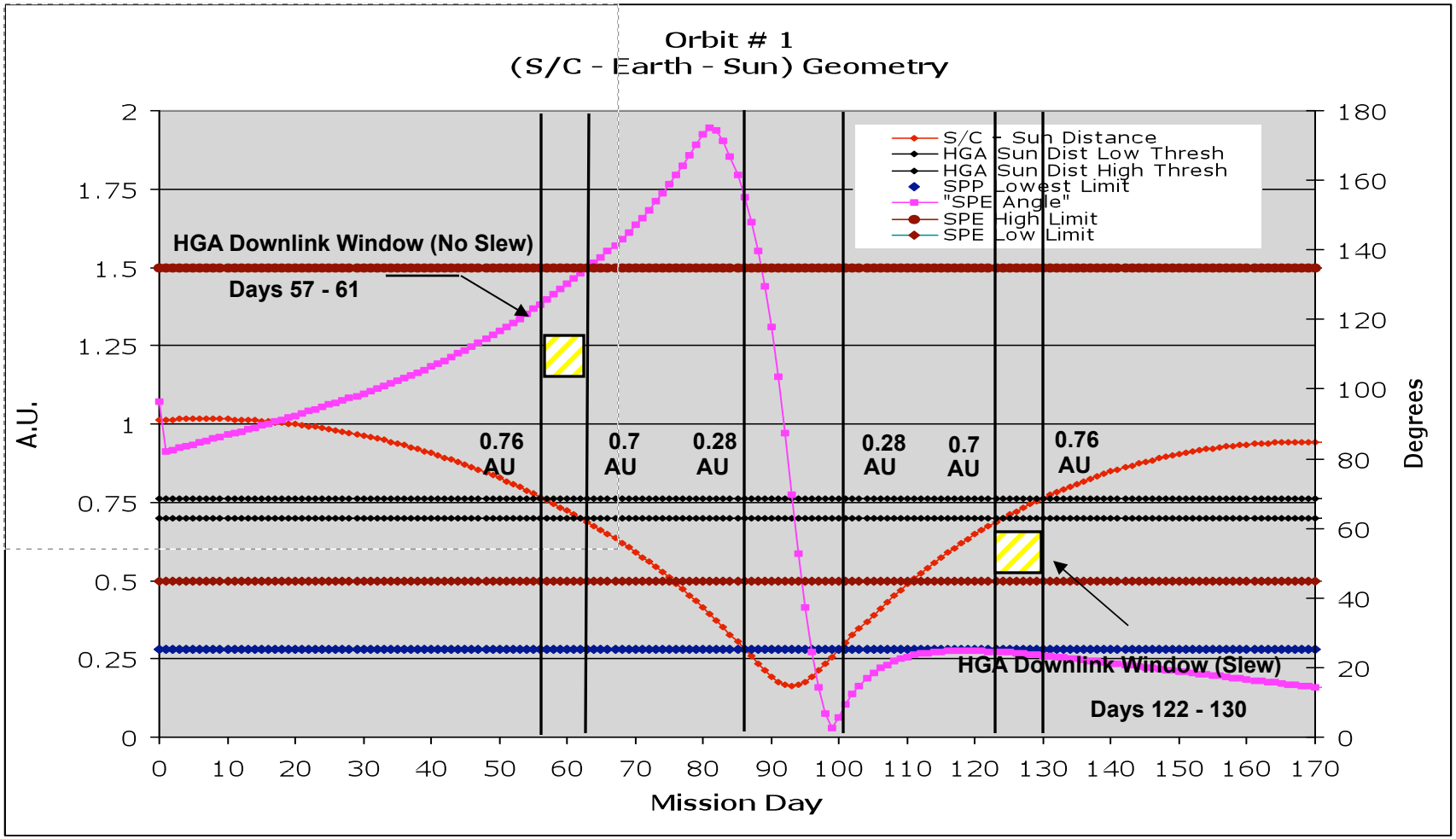
- SPP will use heritage Planning, Control & Assessment architecture and software tools from TIMED/STEREO/RBSP operations
  - Planning
    - Track Scheduling — Maintenance Events — Procedure Development
    - Maneuver Planning — Command Sequencing — Instrument Queue Mgmt.
    - SSR management — Special Events
  - Control
    - R/T telemetry processing — Command Uplinks — R/T Anomaly Investigation
    - Unattended Operations
  - Assessment
    - Trending — Simulation validation — MOPS database

- Need detailed assessment of STEREO/RBSP tools to determine applicability to SPP
  - Identify modifications needed to support SPP

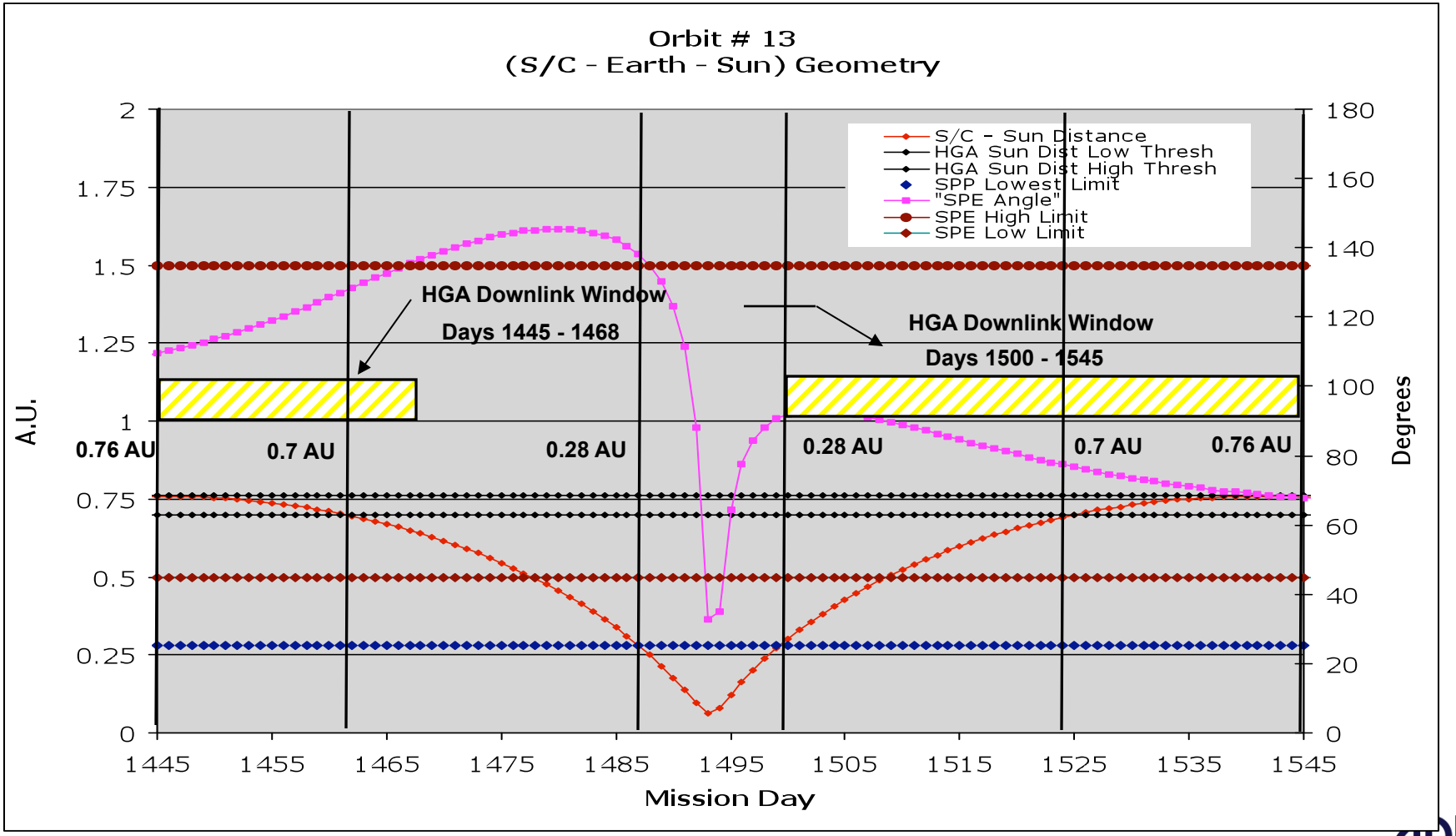
Science Data Downlink Strategy

- Downlink of science data requires use of HGA
  - Spacecraft power/thermal and SPE angle constraints limit HGA use
    - 0.28 AU < S/C-Sun Distance < 0.76 AU
      - 45 deg < SPE Angle < 135 deg (No slew required)
    - 0.7 AU < S/C-Sun Distance < 0.76 AU
      - 0 deg < SPE Angle < 45 deg (Slew required)
- Results in high priority DSN utilization periods each orbit
  - These periods can span from a few days up to several weeks per orbit leg
  - Baseline analysis assumes 10 hour daily DSN contacts (8 hour downlink) but longer contact times should be investigated
  - 34m antenna baselined for all contacts
- Detailed HGA availability analysis required for each orbit
  - HGA downlink periods will be incorporated into each orbital operations template
- For some orbits, science data collected during a solar pass may need to be downlinked on the subsequent orbit

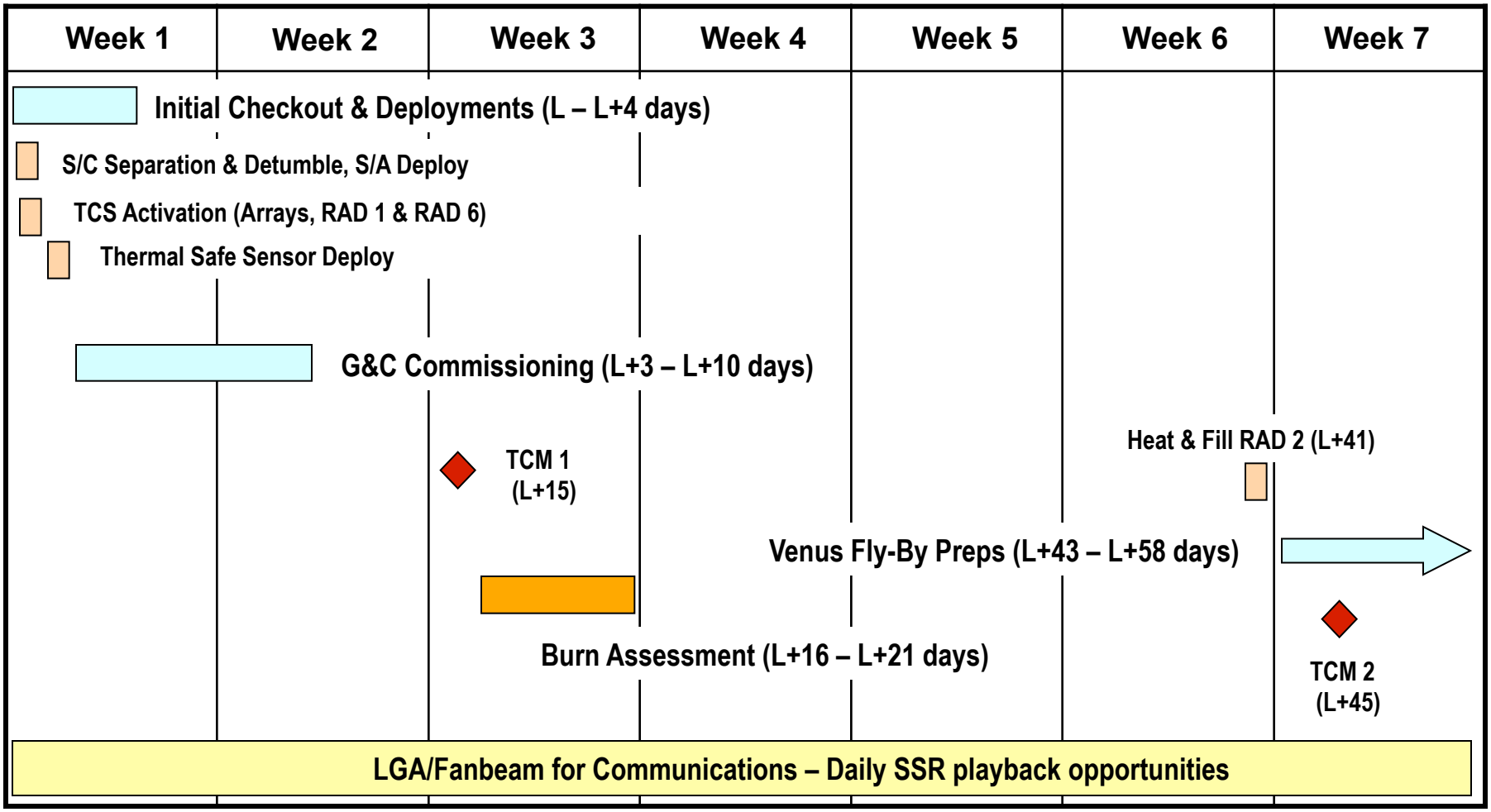
Orbit 1 – HGA Availability Analysis (Low Data Return Scenario)



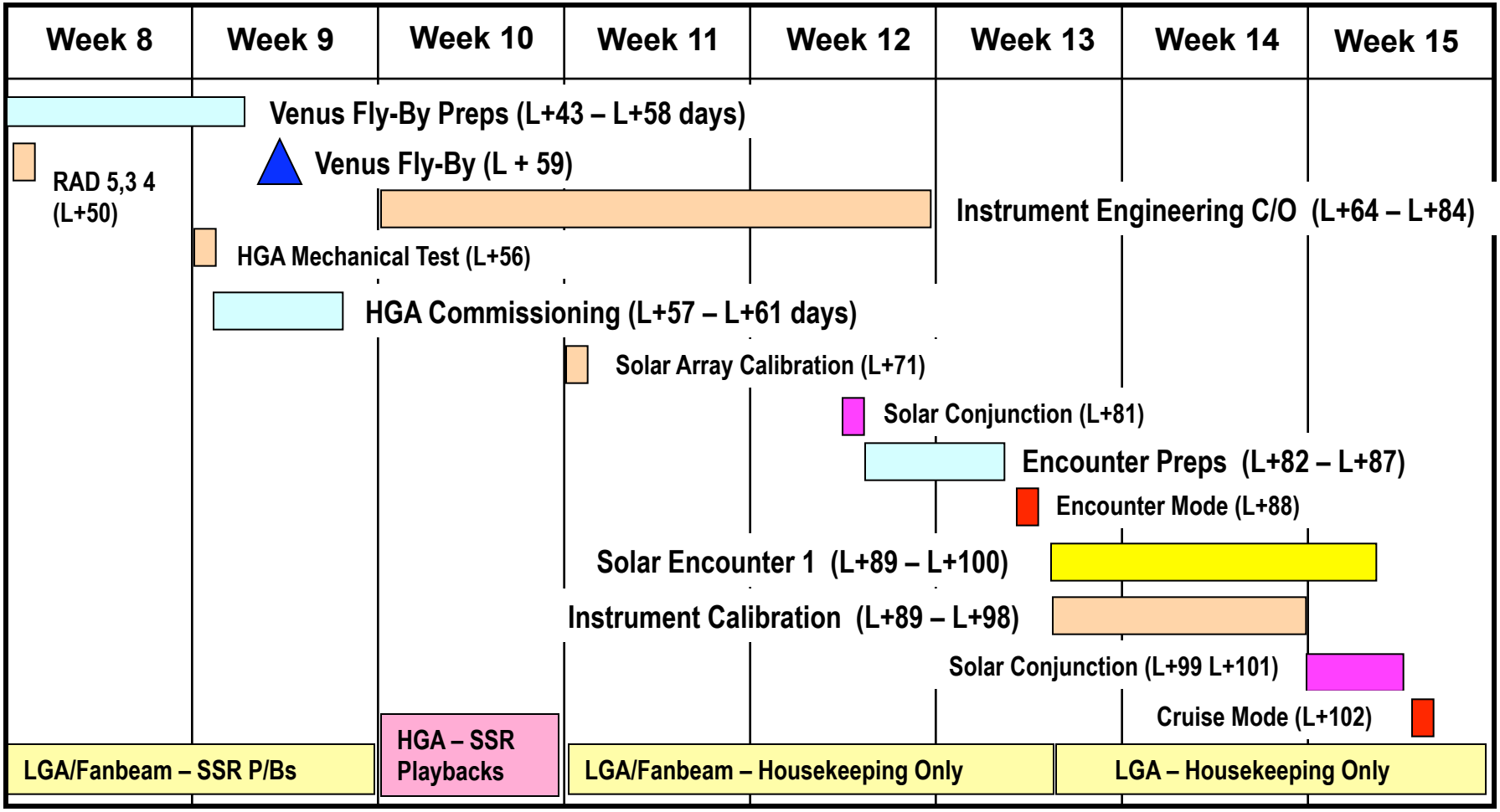
Orbit 13 – HGA Availability Analysis (High Data Return Scenario)



Early Operations Timeline



Early Operations Timeline



Early Operations Highlights

- Autonomous spacecraft de-tumble and solar array deployment at separation (L +0)
- Thermal Control System Activation – Solar Arrays & Radiator Panels 1 & 6 (L +0)
- G&C commissioning completed prior to TCMs (L+3 – L+10)
- TCM-1 at (L+15), TCM-2 at (L+45)
- Solar Array Calibration (L+41)
- Venus Fly-By (L+59)
  - Payload powered off
- HGA Commissioning during Venus Fly-By (L+57 – L+61)
- Instrument Engineering/Functional Checkout (L+64 – L+84)
- First Solar Encounter (L+89 – L+100) - Limited instrument calibration

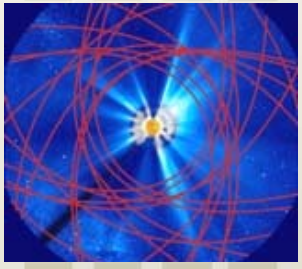
Phase A Plans

- Assess STEREO/RBSP software tools for applicability to SPP
  - Identify areas that need modifications
- Analyze science data downlink strategy using longer contacts and possible rate-stepping
- Continue to work early operations timeline and define orbital operations templates
- Define maneuver implementation process
- Define detailed mission operations interfaces
- Refine mission operations staffing and DSN profiles as needed
- Support system engineering and project management as needed

Backup Slides

DSN Support Profile

Mission Phase	Contact Frequency	Duration
Launch & Initial C/O	Continuous	2 weeks
Early Commissioning	5 – 10 hr contacts per week	4 weeks
Cruise Operations	3 – 8 hr contacts per week	Weekly
Science Downlink	1 – 10 hr contact per day	Entire science downlink period (Varies each orbit leg, ~ 4 - 21 days)
1st Solar Encounter	Continuous	2 weeks
Solar Encounter Phase	1 – 8 hr contact per day	Entire solar encounter period (~ 2 weeks)
Venus Fly-Bys	5 – 10 hr contacts per week 1 – 10 hr contact per day	V-5 to V-1 weeks V-1 to V+1 weeks



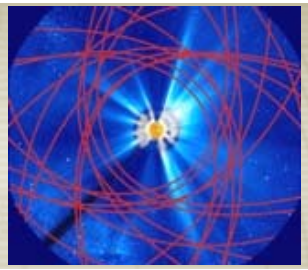
Solar Probe Plus
Mission Concept Review

Safety and Mission Assurance

Rick Pfisterer
Systems Assurance Manager

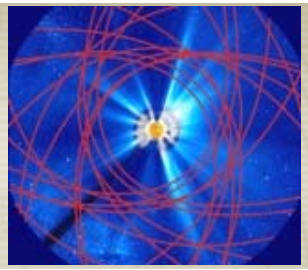
APL
The Johns Hopkins University
APPLIED PHYSICS LABORATORY

Overview Topics



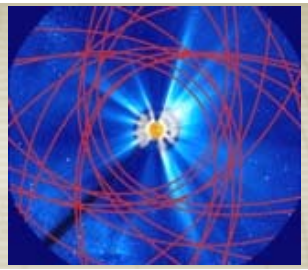
- Mission Assurance Team
- SQA Organization
- Quality Assurance
- Quality Assurance Documents
- ESD Control
- Procurement Product Assurance
- Software Quality Assurance
- Reliability Engineering
- Parts and Materials Engineering
- Problem and Failure Reporting
- Waiver / Deviation Process
- Configuration Management
- Limited Life Items List
- System Safety
- Conclusions

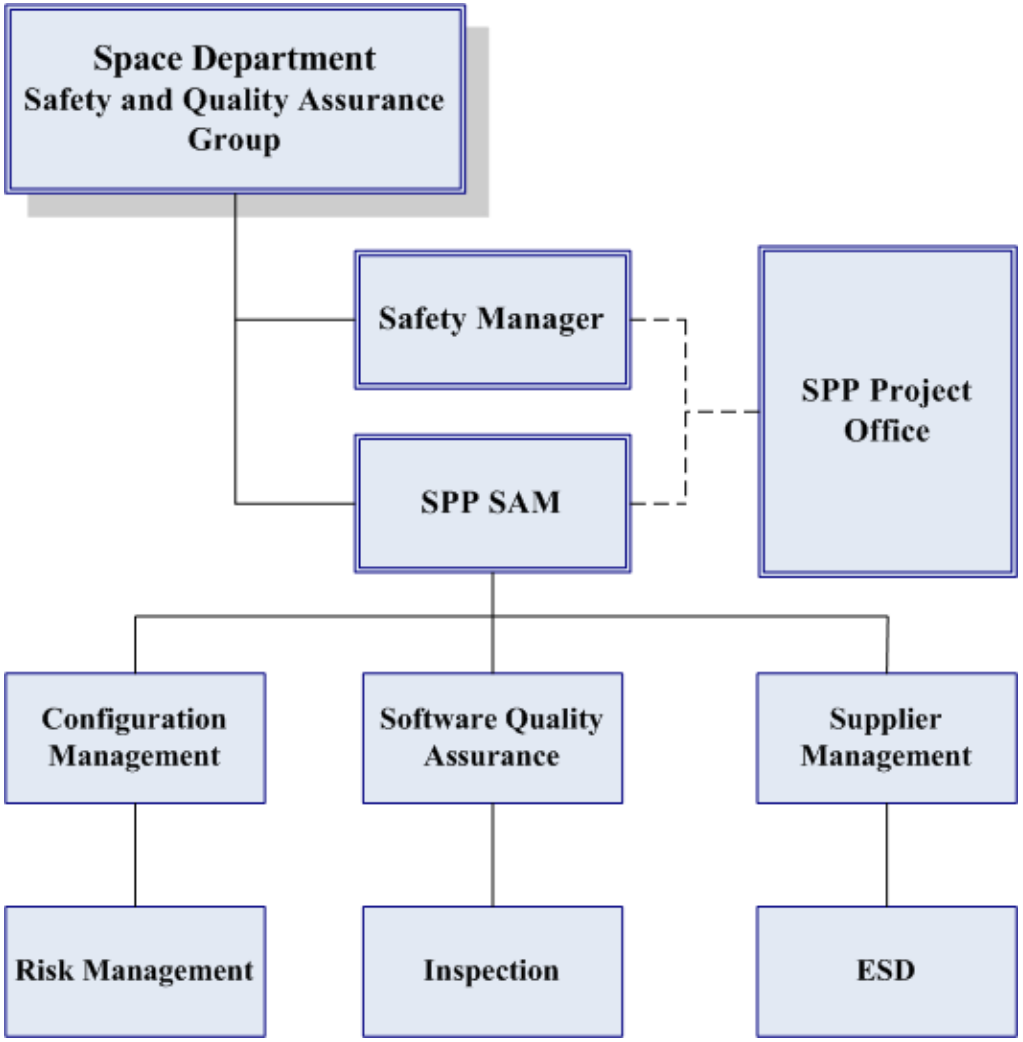
Mission Assurance Team



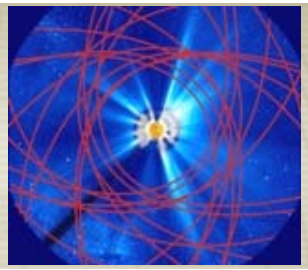
- **Systems Assurance Manager**
  - Rick Pfisterer
- **System Safety Manager**
  - Obi Ndu
- **Reliability (Parts)**
- **Software Quality Assurance**
- **Risk Management**
  - Rick Pfisterer
- **Parts Control**
- **Configuration Management**
  - W. Turgeon
- **Supplier Quality Management**
  - Greg Ellers
- **Problem Failure Reporting**
- **Materials and Process Control**
- **Contamination Control**

SQA Organization



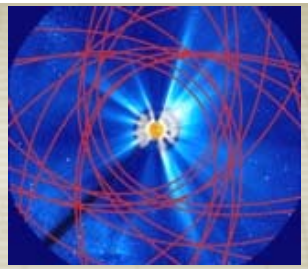


Quality Assurance



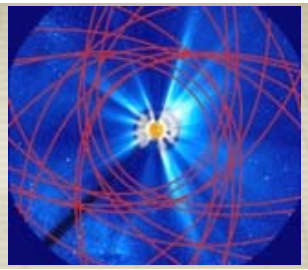
- Quality Assurance provisions are in accordance to and compliant with the APL Quality Management System (QMS)
- The APL Space Department QMS is AS9100 / ISO9001 Certified
- Instrument (s) will be designed, fabricated and tested in accordance with practices that produce hardware / software that meets or exceeds mission requirements
- Quality Assurance staffing has been planned through the end of the mission
- Documentation and critical GSE will be maintained for the life of the program
- Workmanship of Flight Product will be to NASA 8739.x Standards

Quality Assurance Documents



- Configuration Management Plan
- Contamination Control Plan
- Electromagnetic Compatibility Plan
- System Safety Program Plan
- Risk Management Plan
- Parts Control Plan
- Software Quality Assurance Plan
- Instrument Performance Assurance Matrix
- Product Assurance Implementation Plan / Matrix (PAIP)
- Software Development Plan
- Planetary Protection Control Plan
- Mission System Pre-Launch Safety Plan
- Limited Life Items List
- Reliability Program Plan
- Materials and Processes Control Plan
- EMC Test Plan
- Mishap Reporting Plan
- Functional, Development & Assembly Testing Req. Document

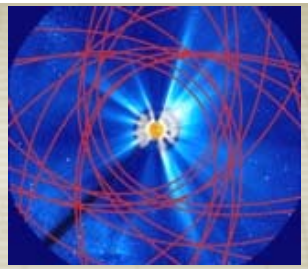
ESD Control



- JHU/APL Space Department ESD control requirements are defined in QY3-540, “Electrostatic Discharge Control”
  - Defines the requirements for all personnel, facilities, equipment, etc.
  - Sets forth requirements for ESD sensitive components
  - Developed in accordance with ANSI/ESD S20.20

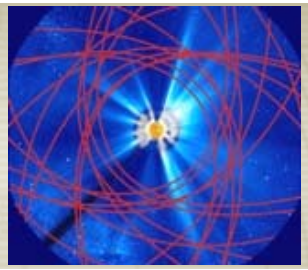
- ANSI/ESD S20.20 ESD control requirements flowed down to all Instrument providers and subcontractors

Procurement Product Assurance



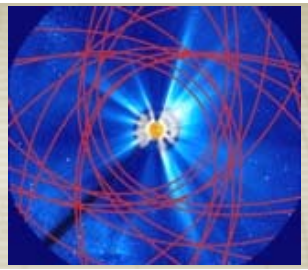
- Release a Statement Of Work (SOW) tailored to each subcontract to ensure requirements flow down for all subcontracted items
- Review and approve critical and sole-source suppliers
- All suppliers must be on the ASL (Approved Suppliers List) prior to release of a PO.
- Participate in technical reviews, identify any area(s) of risk and develop plan(s) to for risk mitigation
- Review and approve materials, parts and processes
- Coordinate PWB coupon testing when required
- Establish inspection point(s) when / where required
- Review and approve end item deliverable package(s)

Software Quality Assurance



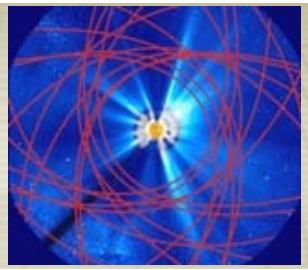
- Verify the level of software processes used for the type of software being produced
- Verify that the content of developed documents comply with the software development process(s)
- Identify and track software QA metrics
- Verify adherence to applicable CM standards
- Verify that planned peer reviews address all applicable software functions
- Verify that unit and integration tests are performed and documented
- Verify that software problem defects are recorded and tracked to resolution
- Review and approve documentation before software is delivered

Reliability Engineering



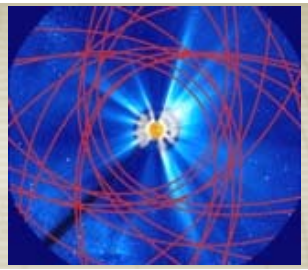
- Executed via the SPP Reliability Engineering Plan which is compliant to the QY3-602, Reliability Engineering.
- Reliability analyses including FMEA, FTA, and worst case analysis are limited to that presented in the Phase B Inheritance and Preliminary Design Review deliverable packages.
- Electrical Parts Lists (EPLs) configured for the flight level assembly shall be provided in Phase C/D for review.
- Electrical / Electronic Parts Stress Analysis (EPSA) shall be provided in Phase C/D per the requirements of EEE-INST-002, which include:
  - Instructions for EEE Parts Selection, Screening, Qualification, and Derating and QY3-777, EEE Parts Derating Process.

Parts and Materials Engineering



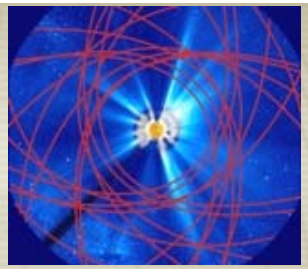
- Parts selection shall be consistent with EEE-INST-002, Level 2
- Per QY3-772, Parts , Materials and Processes Control Board, JHU/APL shall implement an internal Parts, Materials, Process Control Board (PMPCB) for parts and materials selection and approval.
- At a minimum, the following shall be part of the PMPCB:
  - SEC Component Engineer - PCB Chair
  - SPP SAM
  - SPP Mission System Engineer
  - Materials Engineer
- Screening and qualification shall be in accordance with QY3-529, EEE Component Screening and Qualification Requirements for a Level 2 parts program.

Problem Failure and Anomaly Reporting



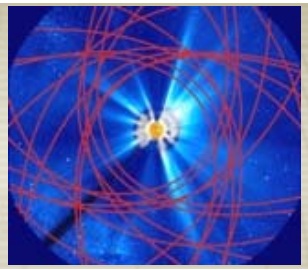
- Problem/Failure Reporting and Corrective Action
  - SPP shall follow the QMS requirements as defined in the SPP PAIP and in QY3-105, Anomaly, Problem, and Failure Reporting Process
  - APL uses an electronically based software tool, the Anomaly Problem Failure Report (APFR)
  - The APFR is managed by the APL SQA Group
  - SAM approval is required for closure of all PFRs
    - The SAM reviews all APFRs to ensure that the failure has been adequately documented, the disposition is appropriate, and all remedies and corrective actions are implemented and verified before closure.

Material and Failure Review



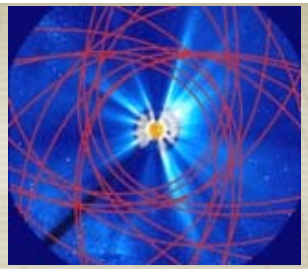
- The MRB process is governed by QMS QY3-106, Material Review
- The SPP Project Manager reserves the right to disapprove a supplier's MRB decisions
- Nonconforming material is designated, controlled, and segregated from normal production flow
- The FRB process is governed by QMS QY3-105, Anomaly, Problem and Failure Reporting Process
- The MRB / FRB shall consist of the following, at a minimum:
  - SPP Mission System Engineer - MRP / FRB Chair
  - SPP SAM
  - Specific Lead Subsystem Engineer (s)
  - At least one technical expert representative(s) related to fabrication, materials, process, part and/or test engineering

Waiver / Deviation Process



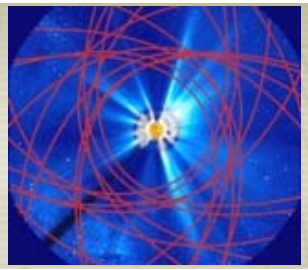
- The Waiver/Deviation Process is governed by the SPP PAIP and is compliant with the APL Quality Management System (QMS)
- APL generated waivers to the QMS are processed in accordance with the QMS QY3-480, Configuration Management Requirements and Implementation.
- Waivers to subsystems and Instruments are processed in accordance with the SOW / PO / Contract. As a minimum, approval of the SAM, Project Manager, and the Mission Systems Engineer is required.

Configuration Management



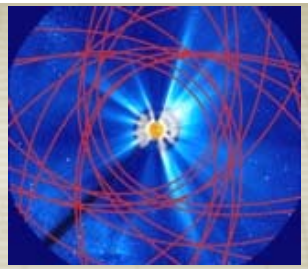
- Executed via the SPP Configuration Management (CM) Plan which is compliant to the requirements of QMS QY3-480, Configuration Management Requirements and Implementation
- Configuration Management is performed by the SQA organization
- CM activities will be performed at all levels: Top, subassembly, board, and module level
- SPP documents and drawings will be maintained in Product Lifecycle Management (PLM) software tool
  - Provides Product Documentation Tree
  - Provides Product Breakdown Structure
  - Provides active control and management of change process documentation
  - Ensures required approvals are obtained prior to release of documents

Limited Life Items List



- A Limited Life Item (LLI) is defined as “any hardware which is subject to degradation because of age, operating time, or cycles”.
- JHU/APL shall identify, document and track the pre-launch cumulative operating times or cycles on LLI’s.
- LLIL requirements will be flowed to all instrument providers.
- A Draft Limited Life Items List will be generated with inputs from instrument providers and the EEE Parts list.
- The Mission System Engineer is responsible for the creation and maintenance of this LLIL.

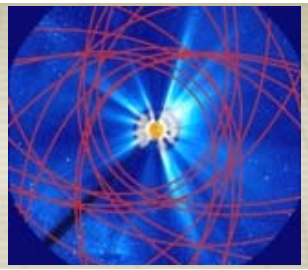
System Safety



- Hazard Analysis and Hazard Mitigation
  - A detailed hazard analysis will be performed on all project hardware, test facilities, ground support equipment, related software, and ground and test operations in order to identify hazardous subsystems and functions.
  - Appropriate controls for mitigating the hazards will be identified together with the associated risk buy-down.

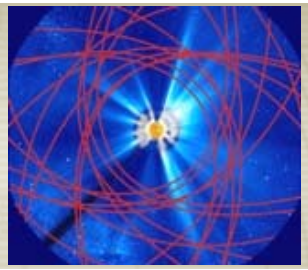
- Hazard Reports
  - Hazard analysis results documentation will be provided in a hazard report which will be part of the projects Safety Data package.
  - Hazard analysis results will be presented to the appropriate level of management for risk acceptance per QMS QY3-301, Chapter 5.5

System Safety



- Tracking, Verification and Closure
  - Safety compliance will be tracked and verified through analysis, test, and inspection
  - The safety review process will be a forum for providing formal feedback on compliance status for all imposed safety requirements and hazard controls

Conclusion



- APL has an well established and fully implemented QMS that encompasses all flight and non-flight elements, interfacing GSE and all software critical for mission success
- Independent NCAS and AS9100 Assessments have confirmed that the APL Space Department is following the best practices required to maintain AS9100 Certification
- The Project is well positioned to be successfully executed

…thank you.

Solar Probe Plus
Mission Concept Review

Preliminary Risk Assessment

Jim Kinnison
Mission System Engineer

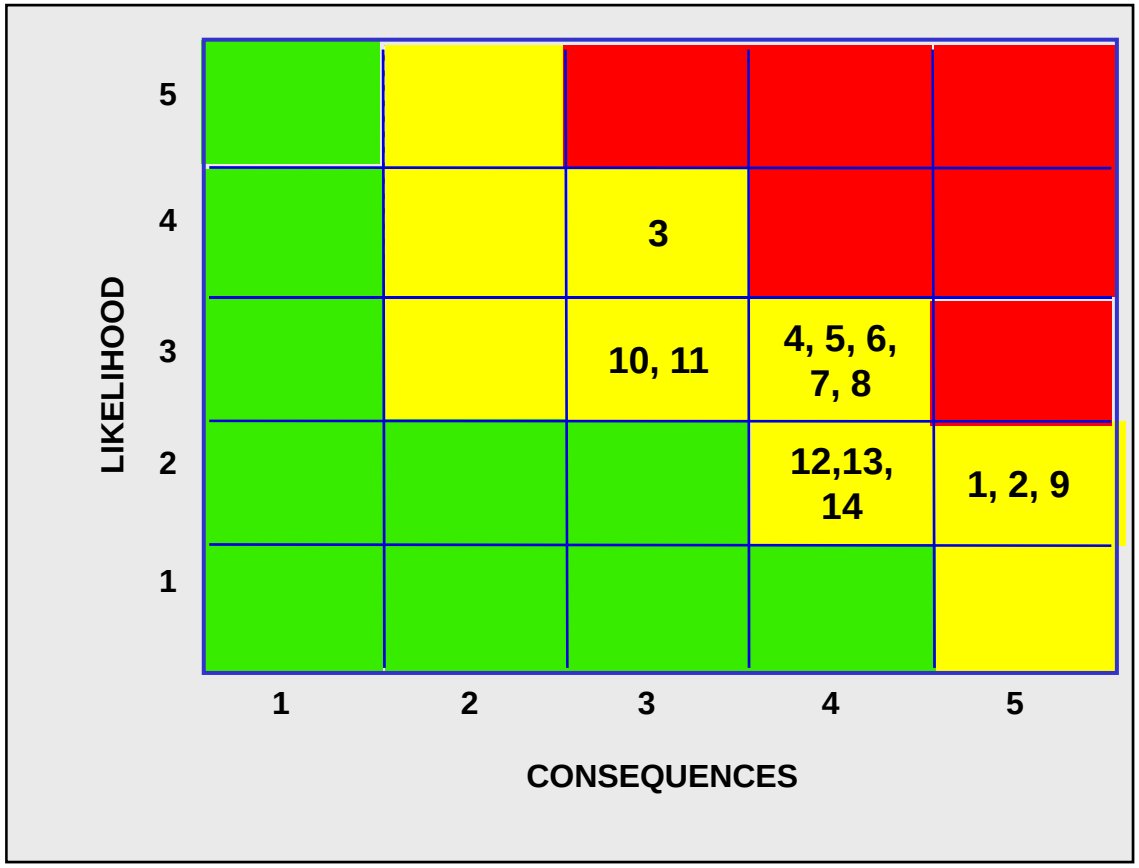
The Johns Hopkins University
APPLIED PHYSICS LABORATORY

Risk position

During Pre-Phase A, we have

- **Identified our technology development risks**
- **Developed risk mitigation plans**
- **Defined adequate system margins for future phases of the program**

We see no showstoppers

Summary of preliminary risk assessment

1- Payload Accommodations
2- Lift Mass
3- TPS Thermal Performance
4- Solar Cell/Array Performance
5- Cooling System Development
6- End-to-end Testing
7- In-flight TPS Alignment
8- Solar Cell/Array Development
9- Safe Mode
10- Cooling System Performance
11- TPS Launch Loads
12- TPS/Array Skew
13- Solar Array Control
14- TPS Development