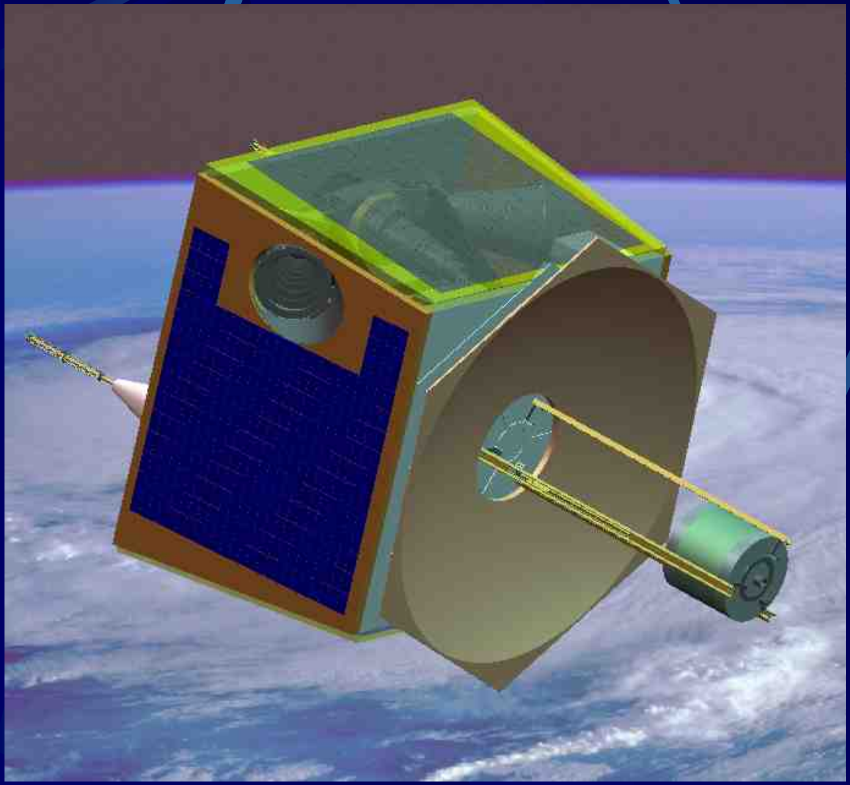
SURREY SPACE CENTRE



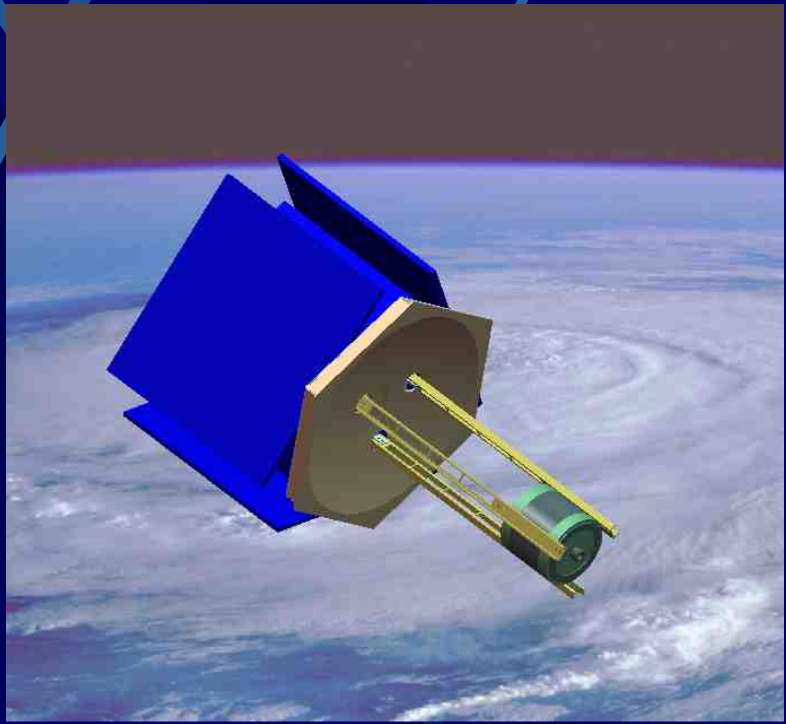
An Analysis of Preliminary Test Campaign Results for a Microscale Solar Thermal Engine

AIAA/USU Conference on Small Satellites

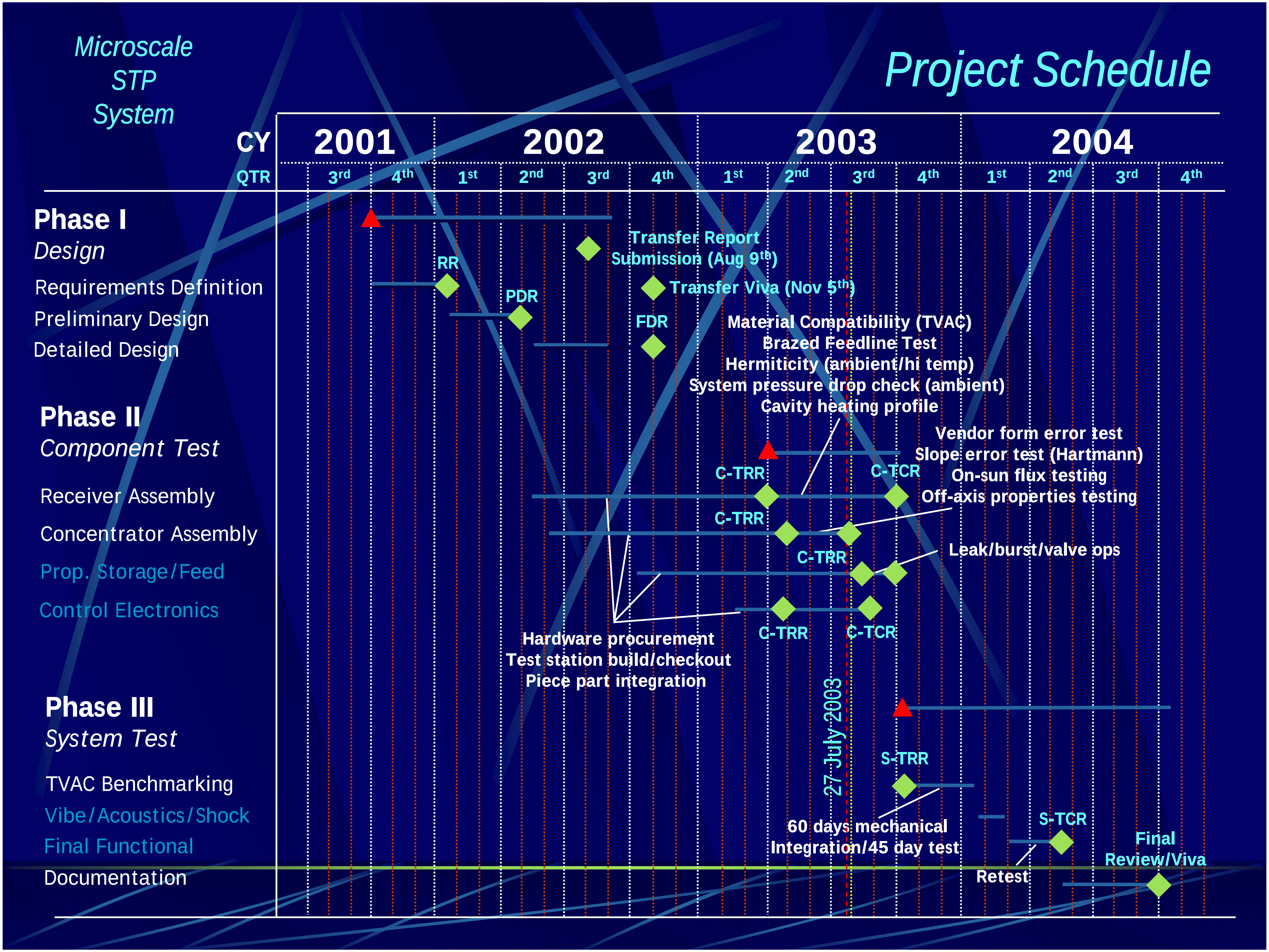
Major Fred Kennedy, US Air Force
Surrey Space Centre
13 August 2003

Agenda

- Overview
- Mission Review
    - Geosynchronous Earth Orbit (GEO)
    - Near Escape
    - Other Body Capture
- Requirements
    - System mass, volume, and transfer time
    - Derived requirements
    - Constraints and miscellaneous requirements
- Design, Test, and Performance Modeling
    - Top-level considerations and option space
    - Subsystem Design (Preliminary and Final)
    - Test Strategies, Modeling, and Results
    - What's Next?
- Q&A



1 N "Micro" Solar Thermal Demonstrator Engine Paired with a "large" SNAP-like nanosat (20 kg wet mass). Burn-average I_{sp} = 325 s., Peak I_{sp} = 360 s.

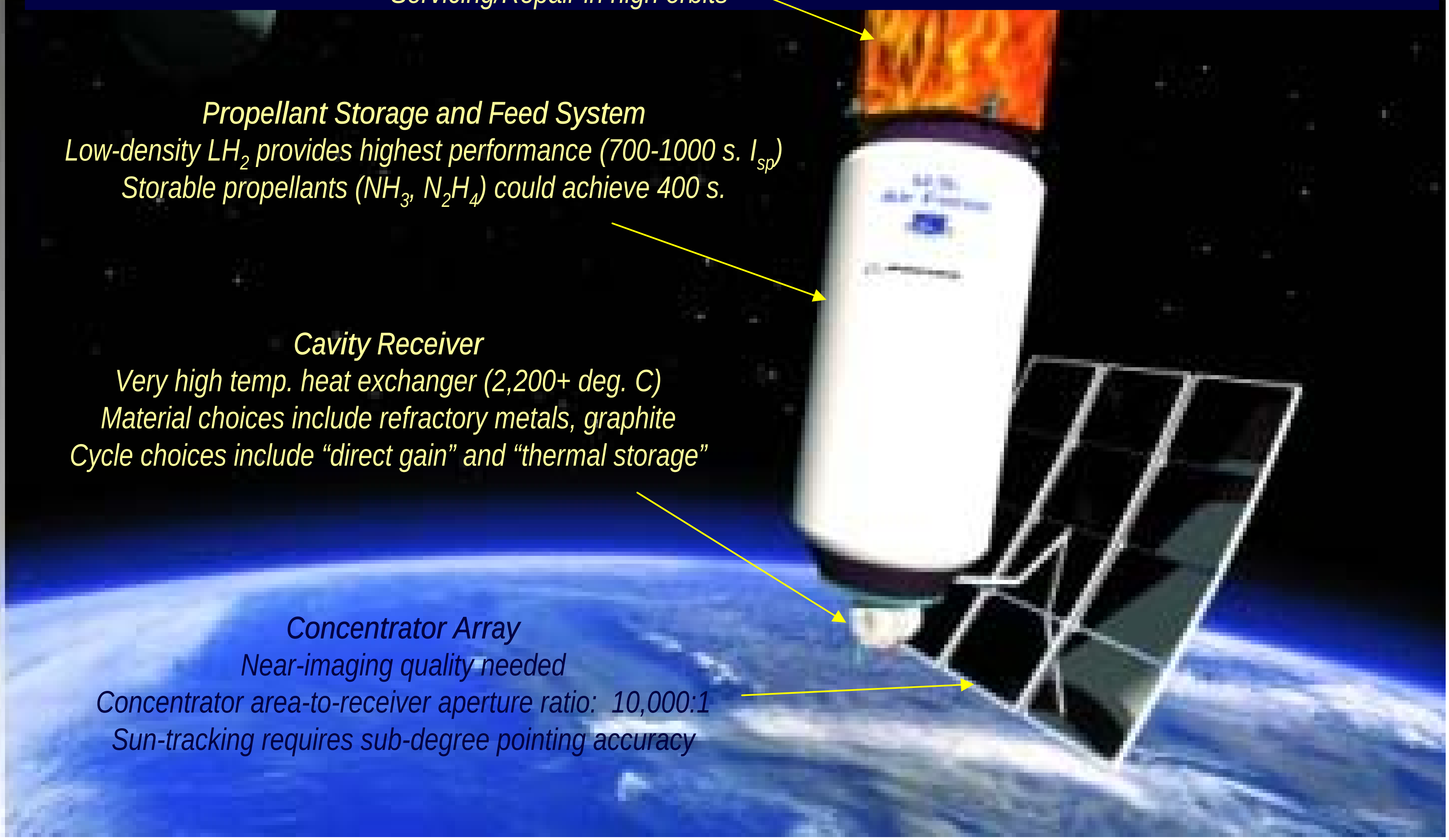


Microscale
STP
System
Project Schedule
CY
2001
2002
2003
2004
QTR
3rd
4th
1st
2nd
3rd
4th
1st
2nd
3rd
4th
1st
2nd
3rd
4th
Phase I
Design
Requirements Definition
Preliminary Design
Detailed Design
RR
PDR
FDR
Transfer Report
Submission (Aug 9th)
Transfer Viva (Nov 5th)
Material Compatibility (TVAC)
Brazed Feedline Test
Hermiticity (ambient/hi temp)
System pressure drop check (ambient)
Cavity heating profile
Phase II
Component Test
Receiver Assembly
Concentrator Assembly
Prop. Storage/Feed
Control Electronics
Vendor form error test
Slope error test (Hartmann)
On-sun flux testing
Off-axis properties testing
C-TRR
C-TCR
C-TRR
C-TRR
Leak/burst/valve ops
C-TRR
C-TCR
Hardware procurement
Test station build/checkout
Piece part integration
27 July 2003
Phase III
System Test
TVAC Benchmarking
Vibe/Acoustics/Shock
Final Functional
Documentation
S-TRR
60 days mechanical
Integration/45 day test
S-TCR
Retest
Final
Review/Viva

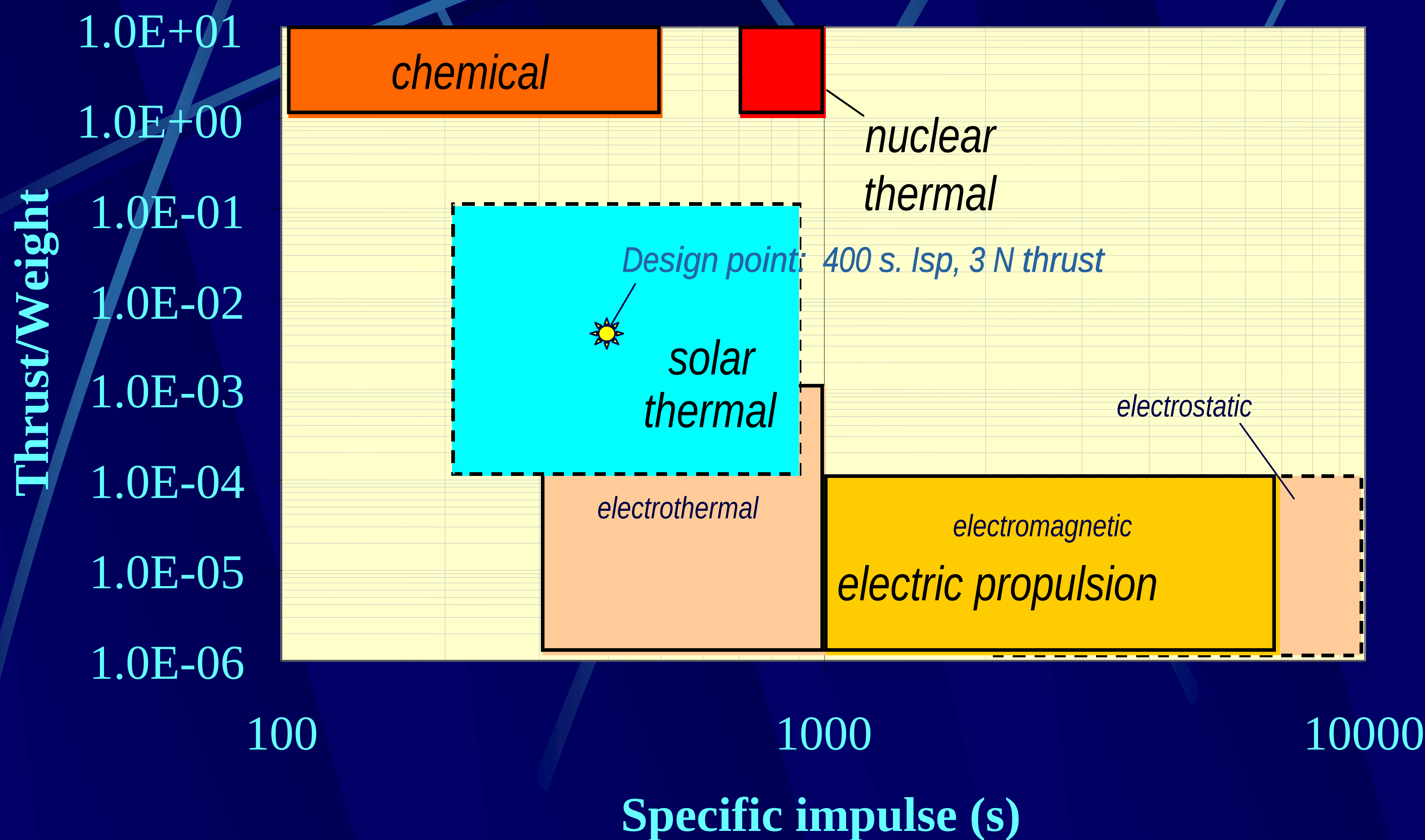
Solar Thermal Propulsion Basics (What is it?)

SOTV

STP is a non-chemical, non-electric propulsion concept using concentrated solar energy to heat a propellant to high temperatures and exhaust it, providing thrust



Why STP? Balancing thrust-to-weight and specific impulse



Solar Thermal Propulsion falls in a useful gap between high-thrust/ low-performance chemical and low-thrust/high-performance electric

Mission Review

Representative Missions

- Three mission classes examined (~ 2,000 m/sec)
  - Perigee kick only (short burn time)
    - Ex.: Geosynchronous Transfer Orbit (GTO) to Earth Escape
    - Velocity change penalties associated with burn times longer than about 15 minutes become severe
  - Apogee kick only (longer burn time)
    - Ex.: GTO to Geosynchronous Earth Orbit (GEO)
    - Burn times of up to 100 minutes approximate "impulsive maneuvers"
  - Combined perigee and apogee kicks
    - Ex.: GTO to Lunar Orbit Insertion

Solar thermal propulsion appears uniquely suited to microsatellites with significant delta-V requirements (1-2 km/sec delta-V), enabling "high orbit" missions

PO2_1_Mar_05

21 Aug 2005 15:04:36.50 Time Step: 5.00 sec

All trajectory analysis performed on STK Astrogator

Representative Missions

- 2004 Asteroid Toutatis Flyby
  - 134 day mission (initial orbit: GTO), 100 kg satellite (wet)
  - Total delta-V: 1,770 m/s
    - Propellant fraction: 36.3 kg
  - 55 “burns”
    - 3 apogee kicks
    - 22 perigee kicks to stable phasing orbit (short of escape)
    - 25 escape burns
    - 5 trim maneuvers 60 days prior to encounter
  - Encounter: 13 Oct 04
    - 3,995 km miss distance
    - Closing velocity = 8.54 km/sec

Monopropellant hydrazine (I_{sp} = 220 s.) requires 56 kg of propellant
Bipropellant hydrazine/N_2O_4 (I_{sp} = 310 s.) requires 44 kg of propellant

Representative Missions

- *Micro Lunar Orbiter*
  - *177-day mission, initial orbit: GTO (7 deg. Inclination)*
    - *Solar thermal engine: 2.98 N thrust, 400 s. I_{sp}*
    - *100 kg satellite (wet)*
  - *62 "burns"*
    - *All firings = 920 s.*
    - *49 apogee/perigee raising maneuvers*
    - *7 Lunar Orbit Insertion (LOI) firings*
    - *2 Perilune kicks (lowers apolune, stabilizes orbit)*
    - *Delta-V = 2,103 m/s, propellant consumption: 41.5 kg*
  - *Final lunar orbit: 1,081 x 16,030 km altitude*
  - *Substantial perigee raising required to achieve LOI with low thrust propulsion system*

ESA's SMART-1, using ion propulsion, will require 15 to 17 months to achieve lunar orbit

Requirements Generation

Requirements Generation

- Ground Rules
  - Storable monopropellants (e.g., ammonia, methane, hydrazine, water)
    - No cryogenic storage permissible
    - Specific impulse penalty offset by greater simplicity of design
  - Focus on multi-impulse approach to minimize delta-V requirements
    - This tends to favor thermal storage and higher thrust levels
    - Concentrates much of the complexity in the solar receiver
- Key Requirements
  - Per-orbit impulse = Thrust * Burn-Time (N-s) -> drives receiver sizing
  - Specific impulse (s) -> drives choice of propellant, receiver materials
  - Charging time (s) -> drives concentrator sizing, receiver sizing

Build smallest, least-cost system with demonstrable utility

Requirements Generation

- Other Constraints
  - Developmental cost (£1M for microsat-compatible system)
    - Includes host spacecraft integration
    - Significantly less than proposed "full-up" USAF/Boeing flight experiment
    - Recurring cost should be less than competing alternatives (e.g., N_2H_4/N_2O_4)
  - Developmental schedule (< 3 yrs. to launch)
  - Space environment (radiation, contaminants, atomic oxygen degradation)
  - Host spacecraft launch environment
    - Based on Ariane 5 ASAP requirements
    - Includes shock, max. acceleration, natural frequency limits, and sound pressure levels
  - Interface requirements
    - Command and telemetry
    - Power
    - On-orbit safety
    - Pointing (accuracy and knowledge)
    - Vehicle mechanical / electrical interfaces
    - Ground station procedures, software, hardware

Design and Test

Baseline Engine Design Selection

Orbit Transfer Strategy

Spiral | **Multi-Impulse**

Minimum dV is highly desirable… and the multi-impulse approach is always more efficient than the spiral transfer.

Thermal Energy Transfer Mode

Direct Gain | **Thermal Storage**

Direct gain systems are limited in per-orbit impulse by the solar flux incident on the concentrator. Thermal storage systems do not face this limitation.

Concentrator Articulation

Articulating | **Non-Articulating**

A non-articulating reflector subsystem may rely on the spacecraft for pointing. Separate pointing and tracking will not be required.

Conc. Type

Rigid Deployable | **Rigid Fixed** | Inflatable

- Rigid Deployable: Wrap-rib, radial rib, hoop-column, tension truss, petal, etc.—expensive but potentially lightweight
- Rigid Fixed: Metal, composite, glass optics (lenses or mirrors)—highest areal density (kg/m^2) but simplest, cheapest, significant heritage
- Inflatable: Potentially very lightweight with excellent stowage characteristics, but unproven for optical applications

Rigid, fixed optics represent the least-cost, highest-quality, greatest heritage approach. The designer must accept a weight penalty, however.

Selected

Alternative

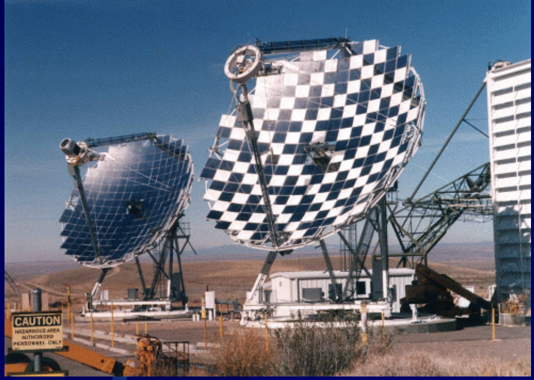
Baseline Engine Design Selection

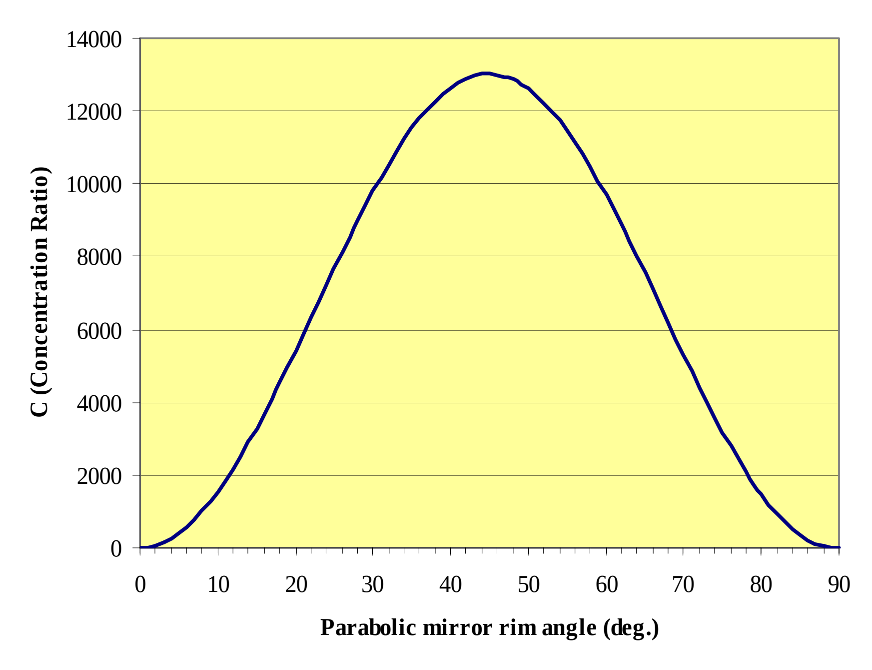
Concentrator Subsystem

What's Needed

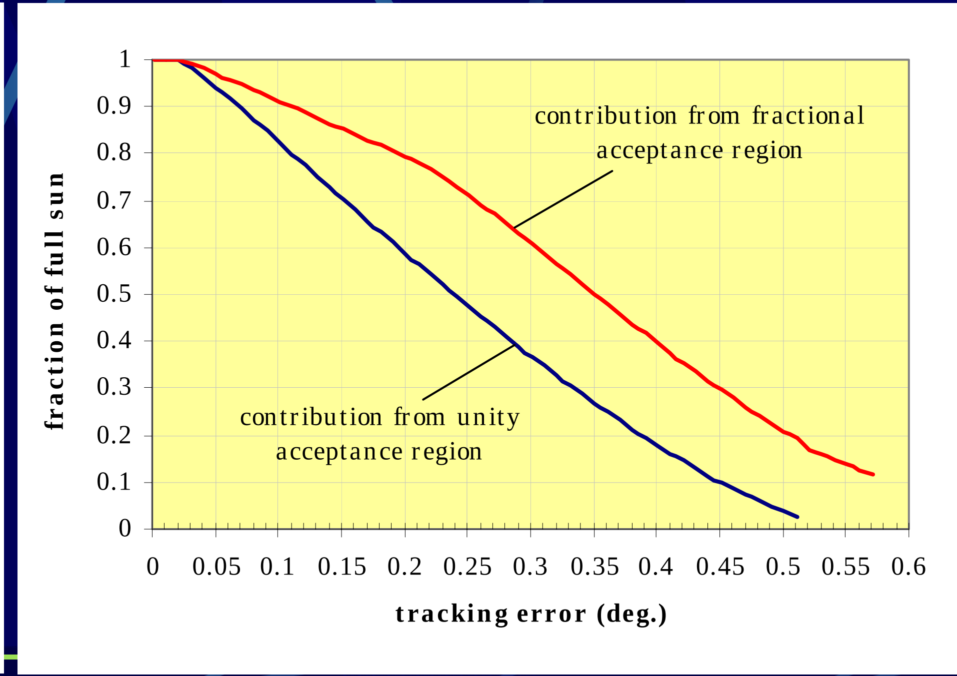
- *High flux at focal point (to achieve high temperatures), which requires:*
  - *High concentration ratios (10,000 and higher)*
  - *Steep (fractional f-number) optics*
  - *Near-optical quality surface (form errors of tens of microns or less)*
  - *Accurate sun pointing (0.1 deg typical)*
- *Options*
  - *Paraboloidal mirrors*
  - *Fresnel lenses with secondary mirrors*

Terrestrial solar thermal concentrator





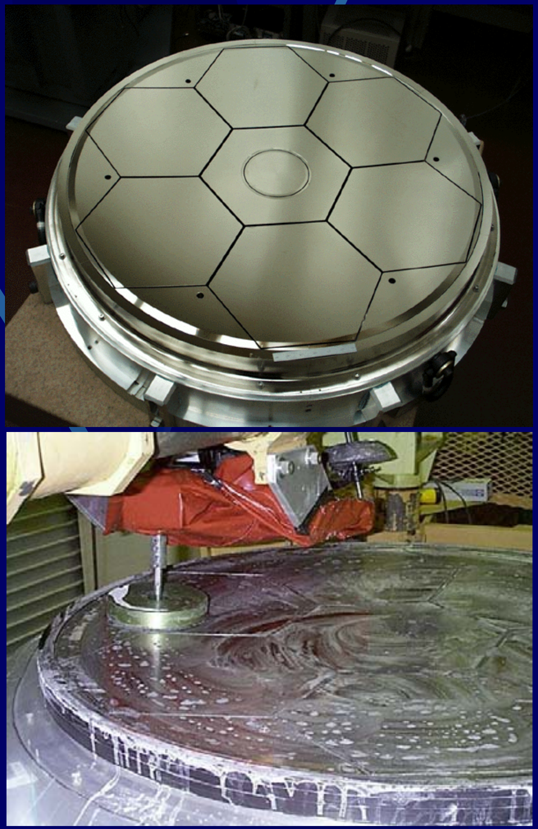
Parabolic mirror C versus rim angle Φ



Parabolic mirror performance in off-track—for a mirror of rim angle (Φ) = 45 degrees, C = 10,000

Baseline Engine Design Selection
Concentrator Subsystem

- Typical Options for Construction
  - Plastics
  - Low Expansion Glass
  - Metal
  - Ceramic
  - Carbon Fiber Composite
- Issues
  - Large plastic mirrors suffer from severe warpage
  - Glass, metal systems have high areal densities (20+ kg/m^2), fabrication techniques are well-understood
  - Ceramic systems (e.g., SiC) rely on expensive and relatively immature fabrication techniques but achieve areal densities of 10-20 kg/m^2
  - Carbon Fiber Reinforced Polymer (CFRP) has been demonstrated at 10 kg/m^2 for IR space telescope applications; also expensive, not yet optical quality



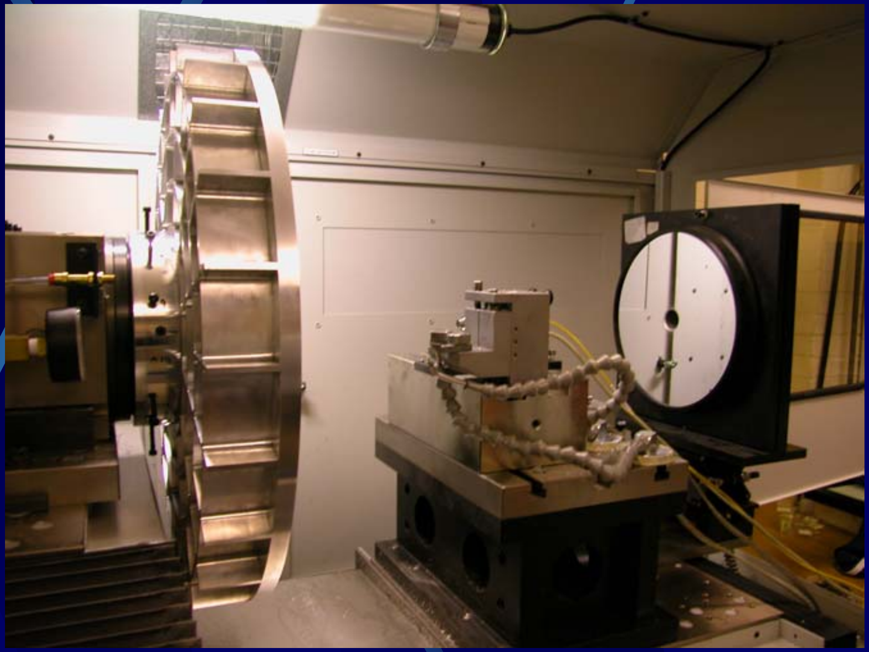
Metal Optic (NASA GSFC DCATT Mirror)
Segmented aluminium substrate with Ni overcoat; 90 cm diameter, diamond-turned surface

Baseline Engine Design Selection

Concentrator Subsystem—Final Design

Metal Test Optic (56 cm diameter f/.6 Mirror) in final fabrication, Jan 03

Single element Al diamond-turned, no overcoat



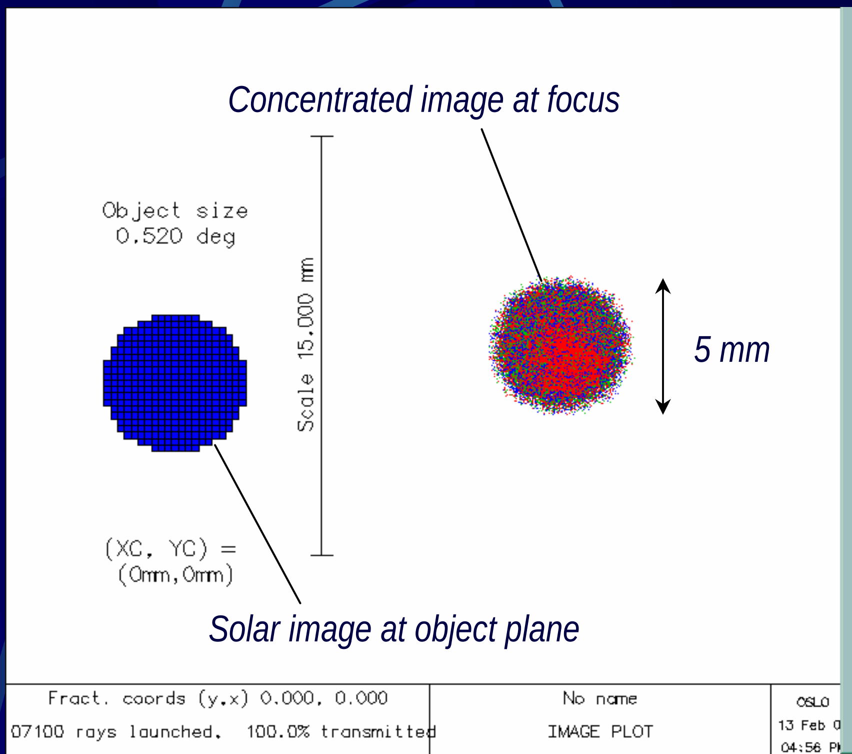
Metal Test Optic undergoing interferogrammetric testing at Precision Optical Engineering, Jan 03

Probe and Interferogram tests

- *Uncoated, lightweighted Al mirror selected over primary alternative (Mg with Ni/Al overcoat)*
- *56-cm diameter provides 250-300 W of solar flux at focal point (at 1,353 W/m^2)*
- *As fabricated, mirror masses 15 kg*

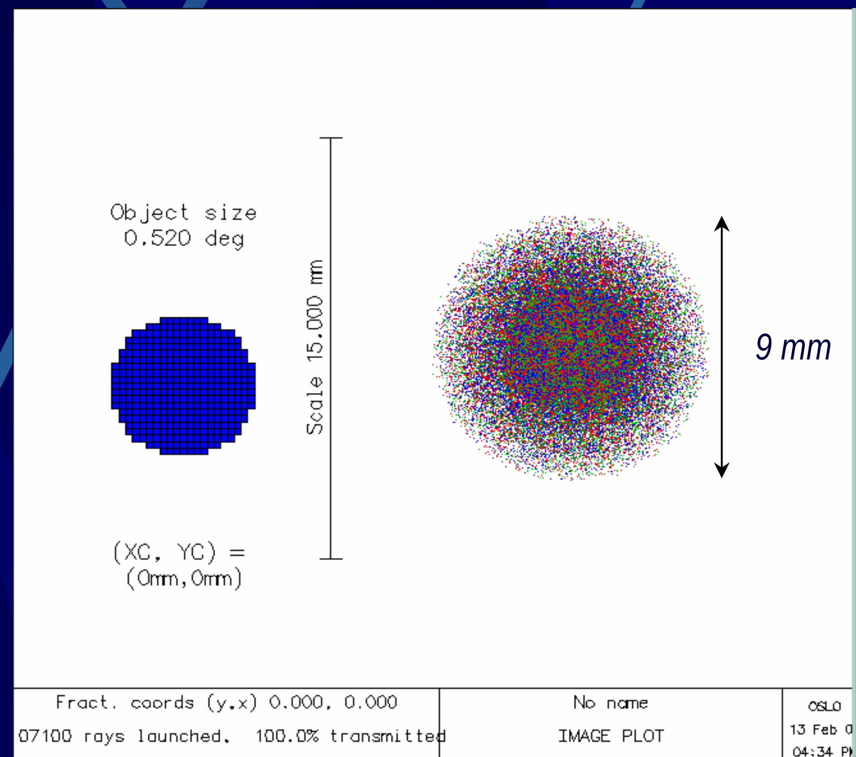
Baseline Engine Design Selection

Concentrator Subsystem—Final Design



Metal Test Optic (56 cm diameter f/.6 Mirror),

1.5 micron form error (RMS)—actual data, simulated in OSLO

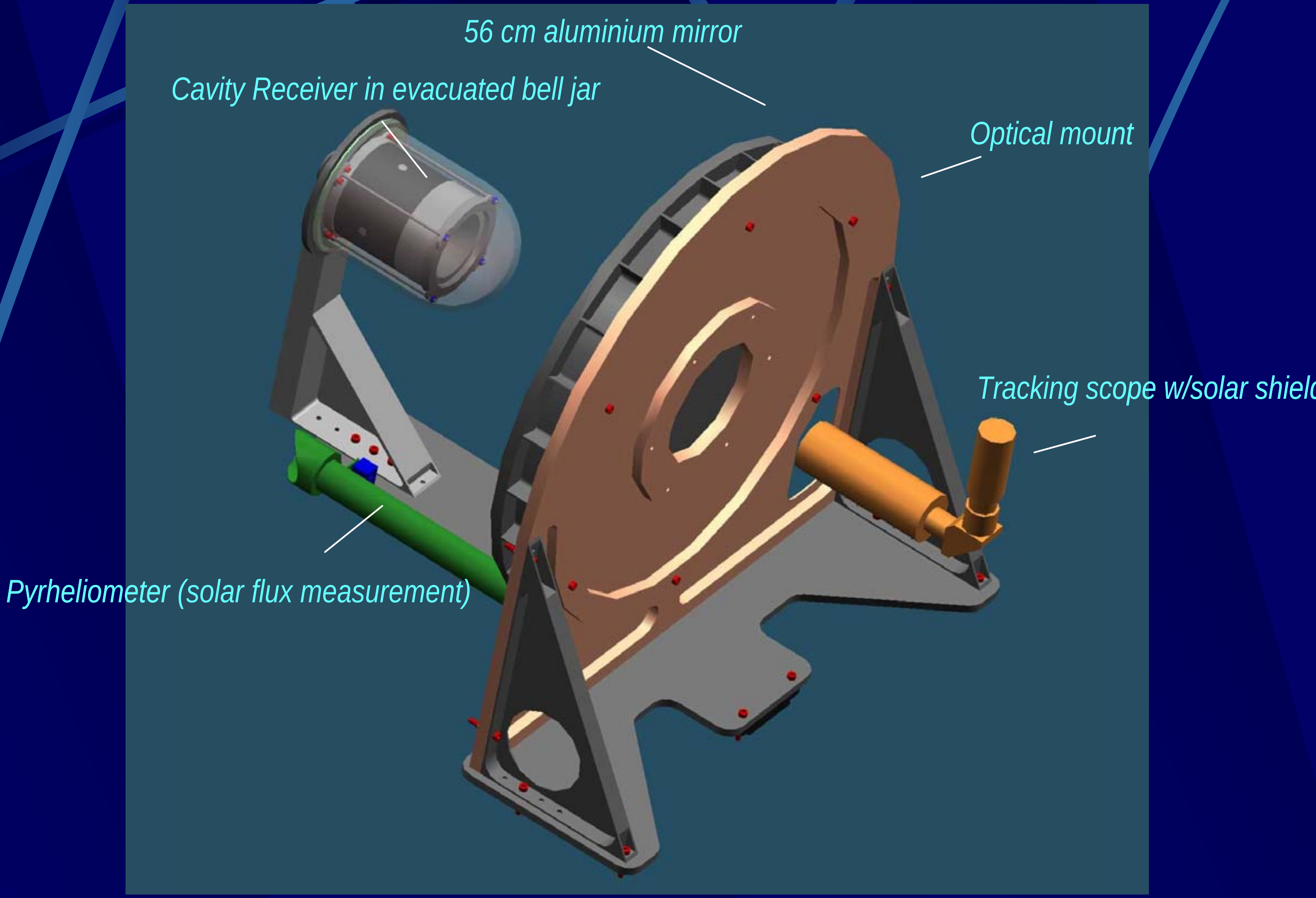


Metal Test Optic (56 cm diameter f/.6 Mirror),

58 micron form error (RMS)—profile identical to fabricated mirror, simulated in OSLO

Baseline Engine Design Selection

Concentrator Subsystem—Final Design



Metal Test Optic (56 cm diameter f/.6 Mirror) on Solar Tracking Test Stand , Two Configurations

Baseline Engine Design Selection

Concentrator Subsystem—Final Design

56-cm test optic with blackened copper target, uncooled

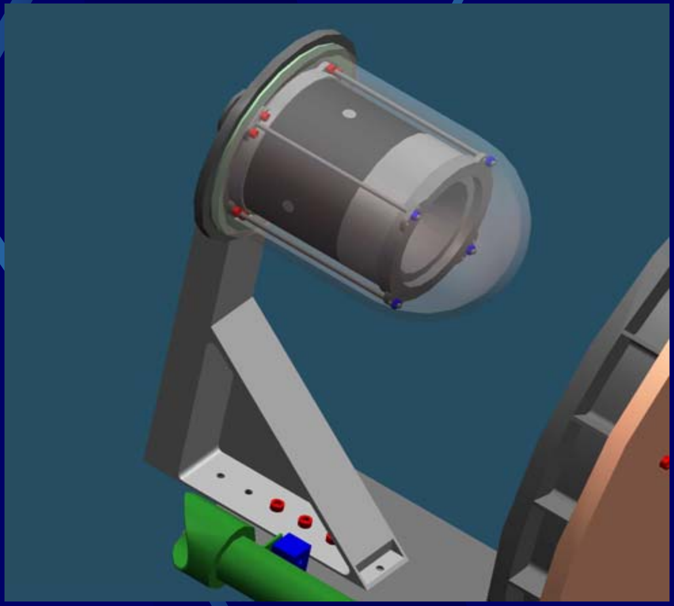
Uncooled copper target in initial test configuration

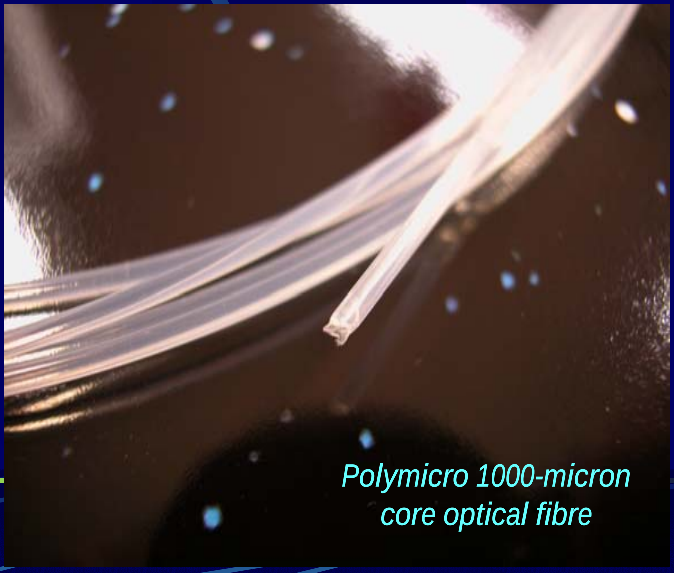
- *Results to Date*
  - *Verified spot size at focus (< 5 mm diameter) and estimated concentration ratio (10,072)*
  - *Flux levels at focus too high for conventional heat flux sensors to measure*
  - *Reflectance of aluminum mirror between .89 and .92*
  - *Absorbed power levels of 130 to 150 W at solar flux levels of up to 740 W/m^2*

Baseline Engine Design Selection

Concentrator Subsystem—Final Design

- *Future Optical Testing*
  - *Off-nominal characterization of mirror performance*
    - *focal length discrepancies*
    - *sun pointing inaccuracies*
  - *Combined mirror-cavity receiver testing with evacuated bell jar*
    - *Will provide more accurate heating profile than electrical tests*
    - *Cannot reach peak temperatures given sea level solar flux levels < 900 W/m^2*
  - *Investigation of multiple small mirrors and single cavity receiver*
    - *Use of low attenuation fibre optics decouples cavity receiver from mirror focal point*
    - *Multiple mirror input permits the designer to take advantage of the improved mass/diameter scaling at small diameters*





Polymicro 1000-micron core optical fibre

Baseline Engine Design Selection

Propellant Selection

H_2

Ultra low-density, hard cryogen, highest Isp performance (800 s.+)

H_2O

"Green," commonly available, but corrosive at high temp., storage complicated by high freezing point (I_{sp} ~ 330 s.)

CH_4

Soft cryogen, decomposition > 1100-1500 C results in carbon deposition ("coking") in receiver

NH_3

Low-density (600 kg/m^3), but relatively high I_{sp} performance (approaching 400 s.)

N_2H_4

High-density, thermally decomposes to ammonia and hydrogen with substantial energy release, toxic, I_{sp} ~ 400 s.

Low-molecular weight exhaust gases are preferred. The engine should retain traceability to H_2-based systems. Oxygen-containing or precipitate-forming compounds are to be avoided. Hydrogenous compounds which store as liquids at likely operating temperature, decompose at high temperatures, with high hydrogen gas yields, are also favored. Ammonia and hydrazine are primary candidates, with water as an option.

Subsystem Trades

Thermal Storage Mechanism

Phase Change

Greater energy storage per kilogram, but two phase (solid-liquid) storage media is hard to model (e.g., void formation, size and location)

Sensible Heat

Permits monolithic structures, does not take advantage of heat of fusion of material

Sensible heat thermal storage systems lack the high energy density of phase change materials but are simpler to build and model. They will be slightly heavier for a given thermal storage requirement. Thermochemical storage represents an additional option (not examined).

Baseline Engine Design Selection

Thermal Storage Media

C

High heat capacity, highly reactive with candidate propellants at high temperature

BN

High heat capacity (1900 J/kg K at 2500 K), inert in hydrogen, ammonia environments to high temperature

B_4C

Very high heat capacity (2700 J/kg K at 2500 K), poor thermal shock characteristics

BeO

High heat capacity, high thermal conductivity, potentially toxic, expensive

TiB_2

Moderate heat capacity, high density, high melting point

B

High heat capacity, high melting point (2348 K), very reactive in hydrogenous environments

W

Low heat capacity (177 J/kg K at 2200 K), high thermal conductivity, high density

Mo

Higher heat capacity than W (390 J/kg K), high thermal conductivity

Re

Similar to W, compatible with C at high temperatures

Al_2O_3

Moderate heat capacity, significant heritage, moderate melting point (2340 K)

Si

Inexpensive, common, low melting point (1683 K) permits use as a phase change thermal storage medium

SiC

Moderate heat capacity, high thermal conductivity, gaining in industrial usage

Desirable properties include high heat capacity, very high melting points (> 2500 K), high thermal conductivity, and low thermal expansion/thermal shock.

Ceramics and refractory metals are nominal candidates Selected materials should be non-reactive with candidate propellants in the temperature regime of interest.

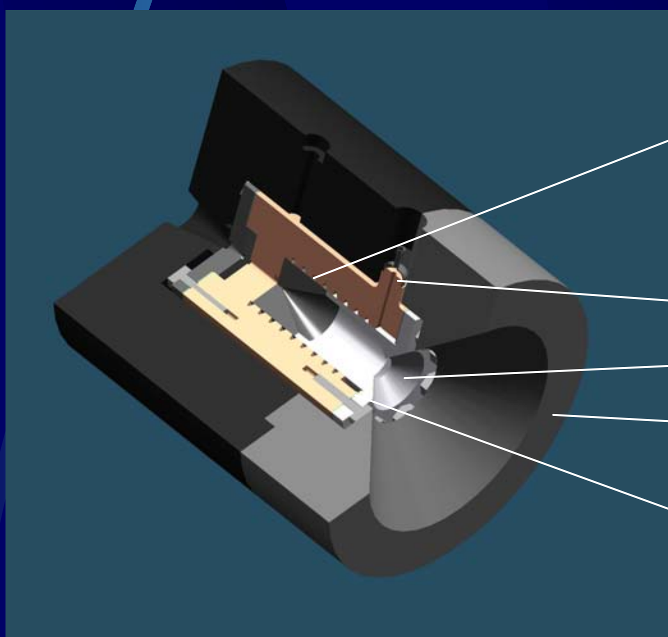
Metals typically exhibit very low heat capacities (< 400 J/kg K), even at high temperatures. Boron carbide, graphite, beryllia, and boron nitride have the highest heat capacities of the materials under investigation.

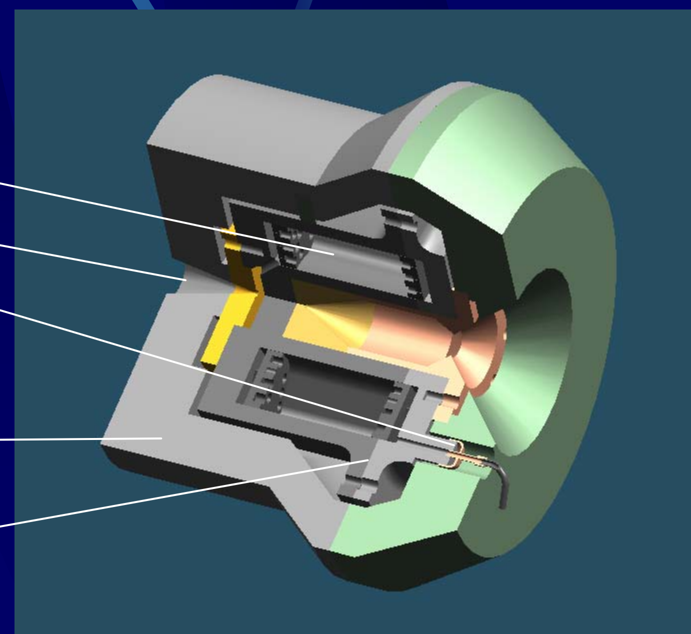
BN and B_4C are selected on the basis of their low reactivities at temperature, coupled with high heat capacities. Graphite (C) and elemental boron (B) represent alternative storage media. They will require non-reactive coatings to prevent erosion at temperature.

Baseline Engine Design Selection

Receiver Subsystem

- Receiver Issues
  - Very high temperatures (2,000-2,500 K)
  - Significant, deep thermal cycling (ambient-2,500 K, ~50 occurrences)
  - High temperature compatible, ultra-low thermal conductivity insulation required
  - High-efficiency thermal storage media-to-propellant heat transfer is needed
  - Sensible heat thermal storage requires high heat capacity, high melting point materials
    - Metal-to-ceramic and ceramic-to-ceramic joining questions
    - Ceramic strength at operational temperature and pressure





Mk. II Micro Cavity (<500 N-s, 1,000 mN)
TiB_2/BN containment,
30 mm graphite foam insulation, 1.3 kg

Mk. I Micro Cavity (>750 N-s, 1,000 mN)
BN particle bed (500 μm) in a TiB_2/BN containment,
15 mm graphite foam insulation, 1.3 kg

Baseline Engine Design Selection

Receiver Subsystem

Mission	Thrust (N)	Per-Orbit Impulse (N-s)	Firing Duration (s.)	Receiver Mass (g)	Insolation (W)/ *Mirror Diameter* (cm.)	Charging Time (hrs.)/*Peak Temp* (K)
GTO-to-GEO	0.15	750	5,000	1,002	333 / *63*	3.1 / *2,501*
GTO-to-Near Escape	3.0	2,808	936	2,745	533 / *79*	5.4 / *2,408*
LEO Orbit Raising	1.0	428	428	712	100 / *34*	2.7 / *1,957*

Mk. II Micro Cavity (<500 N-s, 1,000 mN)
TiB_2/BN containment, 30 mm graphite foam insulation, 1.0 kg

Mk. I Micro Cavity (>750 N-s, 1,000 mN)
BN particle bed (500 μm) in a TiB_2/BN containment,
15 mm graphite foam insulation, 1.5 kg

Baseline Engine Design Selection

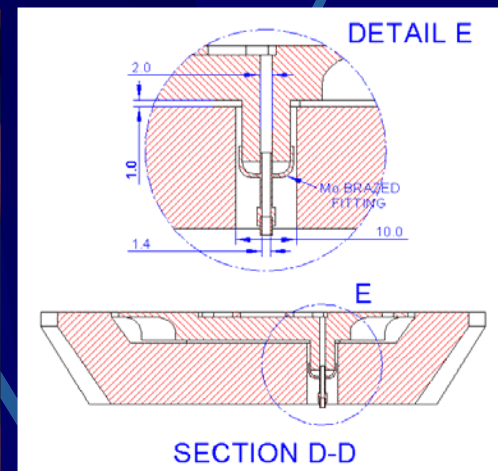
Receiver Subsystem

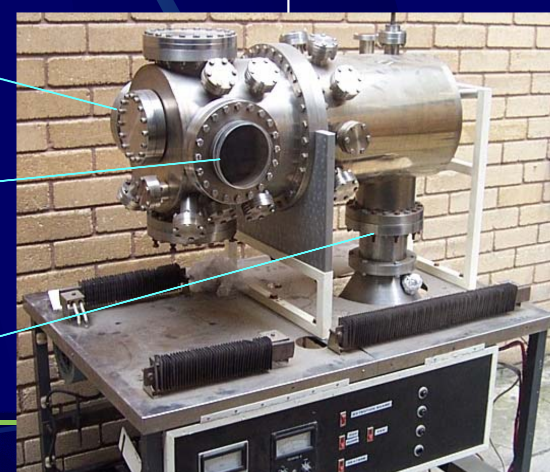
- *Receiver Testing*
  - *Coupon Oxidation*
    - *To determine acceptable levels of vacuum for receiver testing*
  - *Hermetic Seal Verification*
    - *Graphite foil, Mo bolts*
  - *Ceramic to Mo feed line*
    - *Verify mechanical and brazing approaches (Mo-Ru, Mo/$MoSi_2$/Si)*

- *Further Testing*
  - *No-Flow Heating, Ambient and High-Temperature Flow Testing (10-20 bar N_2, N_2/H_2)*
    - *Cavity electrical heating with no installed plumbing*
    - *Electrical and solar heating with Mo feedlines*


Seal verification

Feed line braze validation





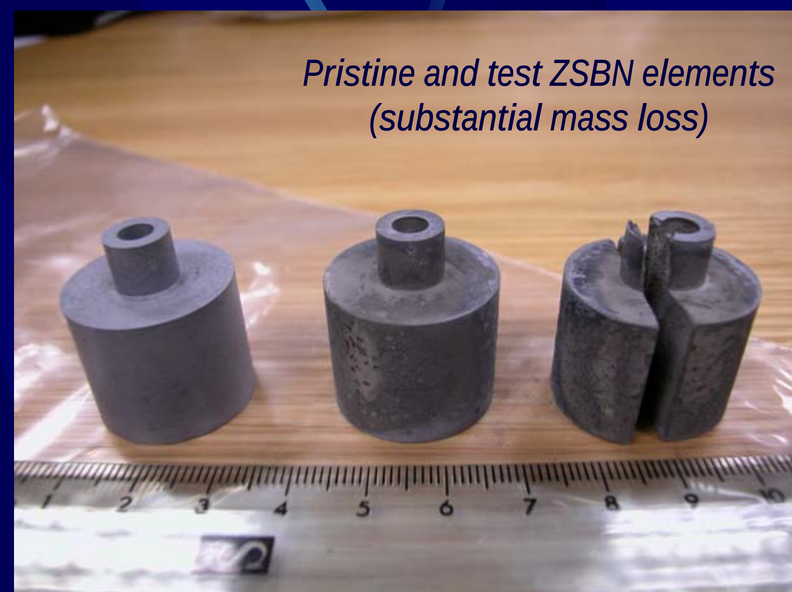
Vacuum chamber acquired for use at SSTL's Westcott E-Site

Optical pyrometry and W/Re thermocouples provide temperature data

Forepump and oil diffusion pump permits hard vacuum ($\sim 10^{-5}$ mbar)

Andy Blackburn with MAST Carbon graphite furnace

Test TiB_2/BN elements



Pristine and test ZSBN elements (substantial mass loss)

ZSBN and TiB_2/BN element testing, MAST Carbon, near Guildford, England, Jan 03

Elements exposed to 2,300 K, low vacuum (20 mbar or less He purge), 40 minutes

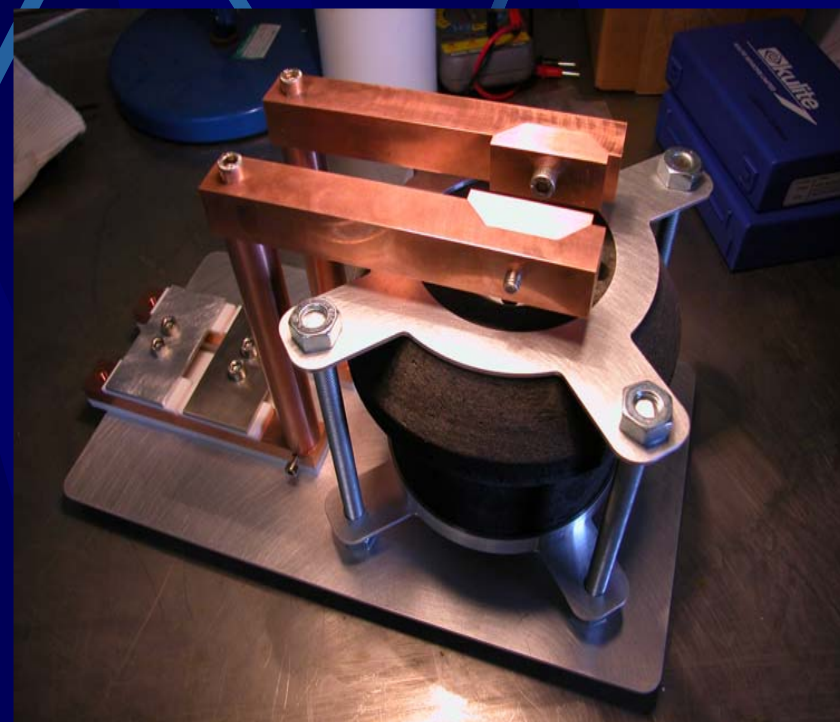
Baseline Engine Design Selection
Receiver Subsystem

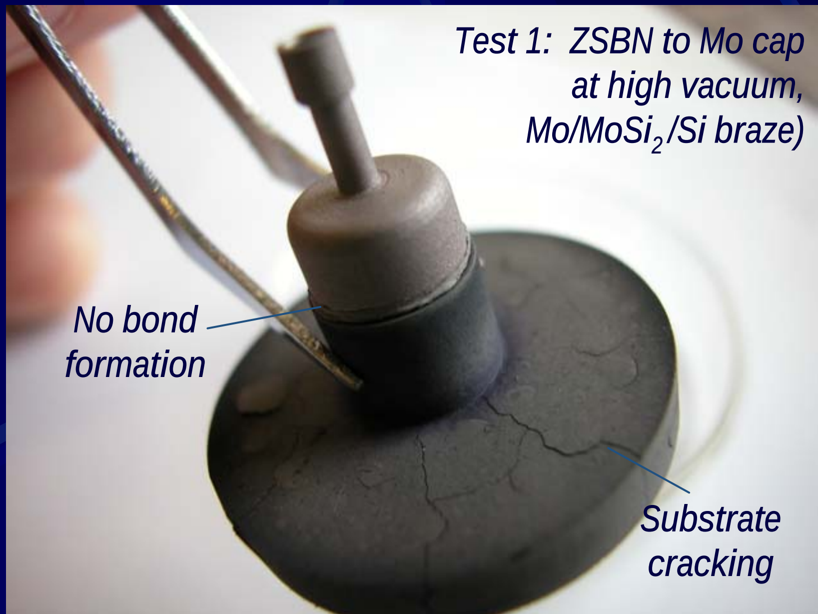
Brazing tests, UMIST (Manchester), Mar 03;
MAST Carbon, May 03

ZSBN and TiB_2 /BN posts brazed to molybdenum caps, using (1) molybdenum/ruthenium and (2) molybdenum/silicon

Cavity heating, Westcott, England, ongoing

Testing to validate thermal models using electrical heating to simulate concentrated sunlight input





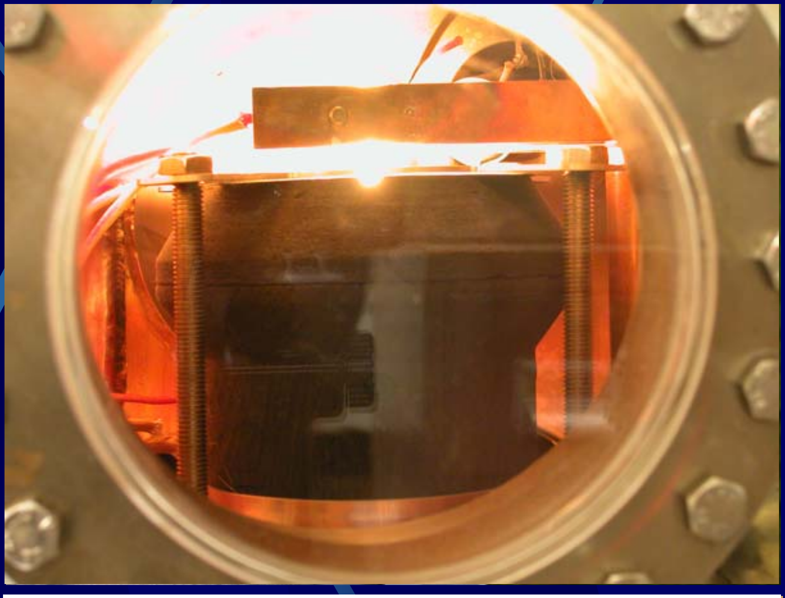
ZSBN and TiB_2/BN to Mo braze testing, UMIST, Manchester, 12-14 Mar 03

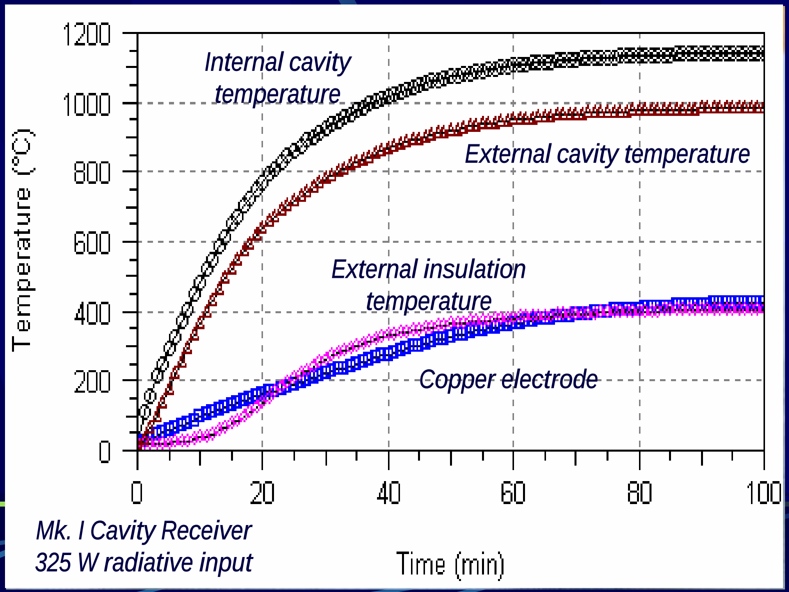
Elements exposed to temperatures of up to 2,100 K, high (< 10^{-4} mbar) and low (1 mbar) vacuum

Baseline Engine Design Selection

Receiver Subsystem

- Results to Date
  - Electrical heating
    - Peak external/internal cavity temperature on Mk. I cavity receiver: 1,424 K/1,650 K at 1,300 W input power, 73 minute charge
    - Peak external/internal cavity temperature on Mk. II cavity receiver: 1,788 K/1,930 K at 1,400 W input power, 71 minute charge
    - Mk. II has experienced three thermal cycles (ambient to 1,930 K)
    - Model suggests that roughly 50% of applied power is lost to conduction or heater lead line radiation
  - Observed Effects on Cavity Receivers
    - No deformation
    - Discoloration, darkening, and graphite precipitate formation on receiver surface
    - “Vacuum welding” of molybdenum bolts to ceramic substrate after temperature cycling





Baseline Engine Design Selection

Receiver Subsystem

- *Future Receiver Testing/Modeling*
  - *Continued effort to reliably heat receiver cavities to 2,000 K and above*
    - *critical for flow testing*
    - *to date, difficult to achieve due to small heater size required, high heat fluxes*
    - *correlation of thermal model to test results not yet complete*
  - *Ambient and moderate temperature flow testing*
    - *will validate mechanical seals and feedline design*
    - *can be conducted on sun tracking mount*
  - *High temperature flow testing (>2,000 K)*
    - *inert gas (He, Ar), nitrogen, and ammonia*
    - *hydrazine testing possible, perhaps in Spring 2004*
    - *will incorporate multiple thermal cycles (up to 60), to provide data for relevant mission profiles*
  - *Coupling to optical fibers*

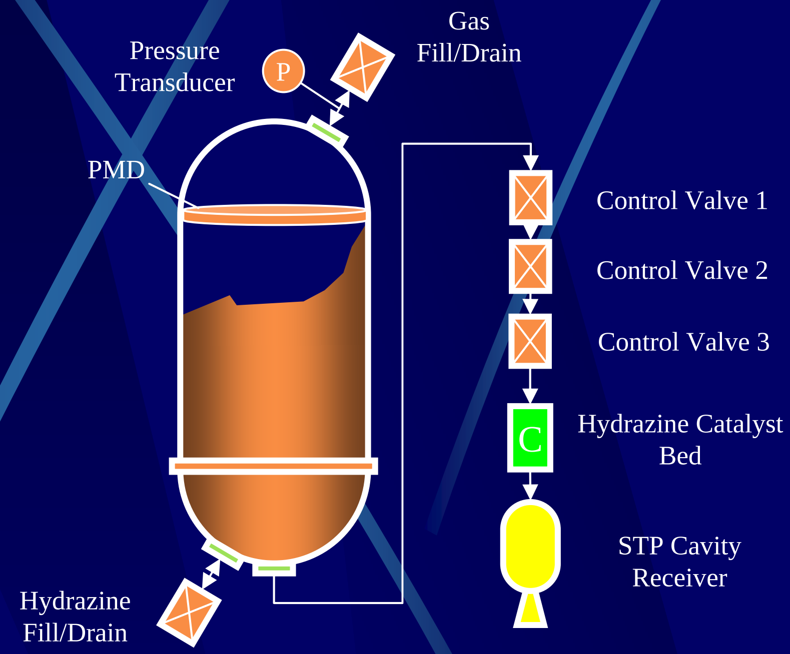
Failed graphite heating element with tantalum leads

Mk. II receiver with molybenum feedline attached

Baseline Engine Design Selection

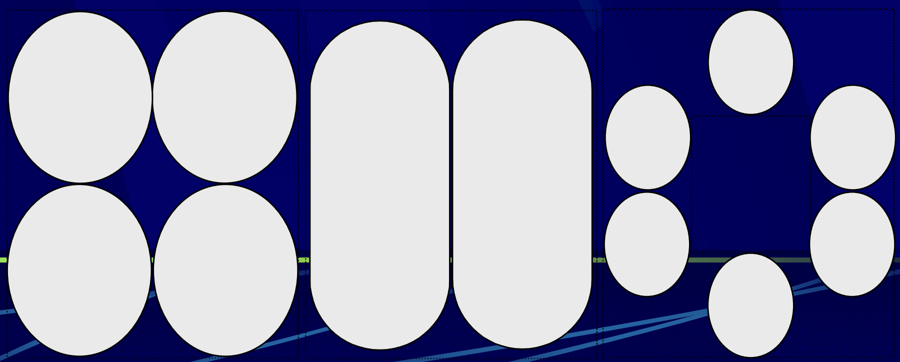
Propellant Storage and Feed Subsystem

- *Issues*
  - *Select a system with maximum space heritage*
  - *Propellant selection: Ammonia (NH_3) or hydrazine (N_2H_4) are primary candidates*
  - *Ammonia can be stored under its own vapor pressure at 120 psi (8 bar) and 600 kg/m^3*
    - *Blowdown feed system, no additional pressurant*
    - *Not a common propellant choice*
  - *Hydrazine possesses high density (1,005 kg/m^3) at slightly decreased performance due to higher decomposition product molecular weight*
    - *Blowdown or regulated system with pressurant*
    - *Toxicity and reactivity impose handling restrictions*
    - *Significant flight heritage*
  - *Ariane 5 ASAP volume constraints, traditional SSTL microsatellite designs constrain location, size of tanks*



Hydrazine 3.5:1 blowdown system

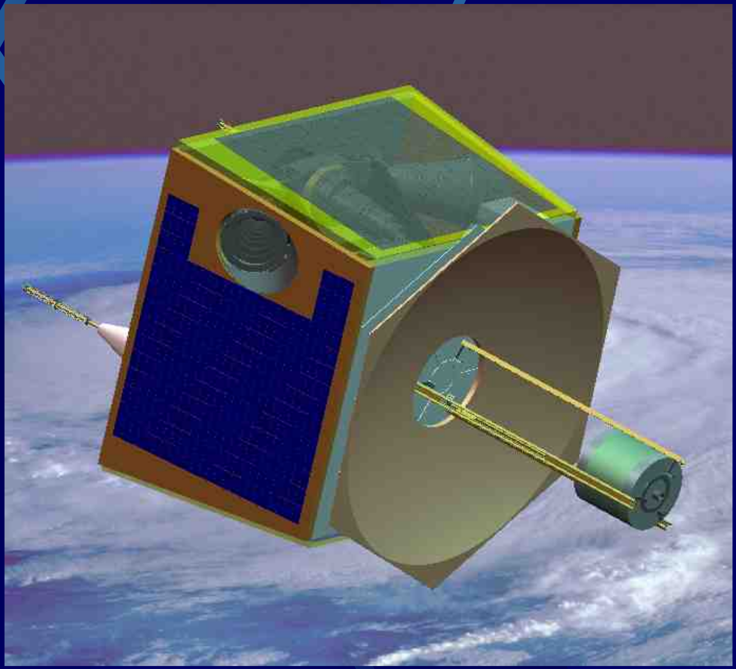
He pressurant (initial pressure = 350 psi), series redundant control valves, three possible tank configurations (below)



Summary

- Summary
  - *Test engine configuration based on detailed mission and performance models*
  - *Two "Micro" designs emerge from effort*
    - *Receiver design appears robust, but issues remain (hermetic sealing, feed line bonding)*
    - *Metal concentrator acceptable for diameters of ~60 cm or less, lighter approaches required for larger engines*
    - *Propellant storage and feed subsystem relies on heritage hydrazine monopropellant efforts*
    - *Detailed optical and thermal models are permitting accurate performance prediction*

- Current Status
  - *Optical and thermal modelling progressing*
  - *Concentrator and receiver tests underway*
  - *Full optical path testing to commence in Aug 03*
  - *Flow testing to begin in Aug 03*



Notional Microsatellite Host
100-kg microsatellite augmented with the small STP engine (150 mN, 355 s burn average Isp)