A bi-modal solar thermal system capable of providing propulsive and electric power to a spacecraft has been identified as a promising architecture for microsatellites requiring a substantial ΔV. The use of a high performance thermal energy storage medium is the enabling technology for such a configuration and previous solar thermal studies have suggested the use of high temperature phase change materials (PCMs) such as silicon and boron. To date, developmental constraints and a lack of knowledge have prevented the inclusion of these materials in solar thermal designs and analysis has remained at the conceptual stage. It is the focus of this ongoing research effort to experimentally investigate using both silicon and boron as high temperature PCMs and enable a bi-modal system design which can dramatically increase the operating envelope for microsatellites. This paper discusses the current progress of a continued experimental investigation into a molten silicon based thermal energy storage system. Using a newly operational solar furnace facility, silicon samples have been melted and results indicate that volumetric expansion during freezing will be the primary difficulty in using silicon as a PCM. Further experimental studies using different materials and test section fill factors have identified potentially reliable experimental conditions at the expense of energy storage density. In addition to conducting experiments, a concurrent computational effort is underway to produce representative models of the experimental system. The current models generally follow experimental results; however, difficulties still remain in determining high temperature material properties and material interactions. This paper also discusses the future direction of this research effort including modeling improvements, analysis of convective coupling with phase change energy storage and potential facility improvements.
Experimental Investigation of Latent Heat Thermal Energy Storage for Bi-Modal Solar Thermal Propulsion

Matthew R. Gilpin, USC
David B. Scharfe, ERC, Inc.
Marcus P. Young, ARFL/RQRS
Rebecca N. Webb, UCCS
Introduction

• Solar thermal propulsion (STP) has over 50 years of developmental history and offers a compromise between thrust and efficiency

<table>
<thead>
<tr>
<th></th>
<th>Monoprop. Rocket</th>
<th>Solar Thermal</th>
<th>Electric</th>
</tr>
</thead>
<tbody>
<tr>
<td>Isp</td>
<td>~230s Isp</td>
<td>300-700s Isp</td>
<td>&gt;1000s Isp</td>
</tr>
</tbody>
</table>

• No solar thermal spacecraft have been flown to date
  ➢ Novel (i.e. “awkward”) architecture
  ➢ Scale of proposed systems
  ➢ Adverse impact as a “demo” mission

• A bi-modal solar thermal microsatellite has the potential to greatly increase the operating envelope of the platform

• The development of high temperature latent heat thermal energy storage is currently an enabling technology
A review by the AFRPL Advanced Concepts Group identified STP as a promising candidate for high performance microsatellite missions (Scharfe 2009)

A bi-modal microsatellite configuration is proposed and further study is recommended

Microsatellite scaling distinguishes STP

Large \( \Delta V ( > 1 \text{ km/s} ) \) possible

Expand the Microsatellite Operating Envelope

- Expand possible “piggy-back” launch options
- GEO Insertion: \( \sim 1760 \text{ m/s} \)
- Near Escape Missions: \( \sim 770 – 1770 \text{ m/s} \)

*Possible with EP, however, STP offers a much shorter burn time and higher maneuverability*
Technological Requirements

- **Solar Concentrators**
  - 10,000 :1 Concentration Ratio
  - Low mass and deployable

- **Fiber Optic Coupling**
  - High transmission efficiency
  - High pointing accuracy

- **Thermal-Electric Conversion**
  - Operational at high temperatures
  - High specific power

- **Advanced Insulation**
  - Low Mass
  - High Temperature

- **High Temperature Storage Material**
  - Matches STP propulsion temperatures
  - High energy density (> 1000+ kJ/Kg)

- **Compatible / Effective RAC**
  - Long term compatibility
  - Effective energy coupling
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- Thin PMA (JAXA) “flight ready” concentrators achieve 200 g/m² and C. ratios > 10,000:1

- Inflatables (AFRPL, SRS) can achieve < 1 kg/ m² and have been reported as being “optical quality”

- Large rigid structures (NASA SD, ISUS) are listed at approx. 3 kg/ m² including mounting, tracking, and deployment

- Microsatellite scale system only requires < 2 m²
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  - Effective energy coupling

- Current lab systems operate at 35% $\eta_{total}$
- Estimated 70% $\eta_{total}$ for a space qualified system from better materials selection
- Pointing accuracy of approx. 0.1° required

Nakamura 2004
**Technological Requirements**

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- **Compatible / Effective RAC**
  - Long term compatibility
  - Effective energy coupling

- Thermophotovoltaics are the strongest candidate

- Operation targets properly matched to solar thermal temperatures.

- Closed Brayton and thermionics scale poorly for microsats

- 15 W/kg in current systems, including radiator

- Edtek

- McDonnell Douglas
Technological Requirements

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  - Matches STP propulsion temperatures
  - High energy density (> 1000+ kJ/Kg)

☐ **Compatible / Effective RAC**
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  - Effective energy coupling

---

<table>
<thead>
<tr>
<th>Material</th>
<th>(k_{th} @ 1000 \text{ C} ) (W/mK)</th>
<th>(k_{th} @ 1500 \text{ C} ) (W/mK)</th>
<th>(k_{th} @ 2000 \text{ C} ) (W/mK)</th>
<th>Density (g/cm³)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Silicon Carbide</td>
<td>45</td>
<td>30</td>
<td>25</td>
<td>3.2</td>
</tr>
<tr>
<td>Boron Nitride</td>
<td>17-33</td>
<td>22.5</td>
<td>18</td>
<td>1.8</td>
</tr>
<tr>
<td>Alumina</td>
<td>6.5</td>
<td>6.6</td>
<td>--</td>
<td>3.8</td>
</tr>
<tr>
<td>Zirconia</td>
<td>2</td>
<td>2.5</td>
<td>3</td>
<td>5.5</td>
</tr>
<tr>
<td>ONRL CBCF</td>
<td>0.17</td>
<td>0.2</td>
<td>0.26</td>
<td>0.2</td>
</tr>
<tr>
<td>Calcarb CBCF</td>
<td>0.2</td>
<td>0.35</td>
<td>0.65</td>
<td>0.18</td>
</tr>
<tr>
<td>Aerogel Filled Graphite Foams</td>
<td>0.25</td>
<td>0.4</td>
<td>0.75</td>
<td>0.07</td>
</tr>
<tr>
<td>Mo - ZrO₂ Multifoil</td>
<td>0.001</td>
<td>0.05</td>
<td>0.1</td>
<td>1.4</td>
</tr>
</tbody>
</table>

- Must operate between 1500 – 2600 K

- **Carbon Bonded Carbon Fiber**
  - Can draw from NASA RTG development
  - Carbon foams with filler to limit radiation loss currently offered by ULTRAMET

- **Low Emissivity Vacuum Gap**
  - Typically the first stage in a TPV system
  - Mo/ZrO₂ multifoil blankets also produced for RTGs

- **Ceramic Doped Aerogels**
  - Underdevelopment with JPL, RZSM, and RQRS
Technological Requirements

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- **Compatible / Effective RAC**
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  - Effective energy coupling
To date, all STP systems have used sensible heat thermal energy storage

<table>
<thead>
<tr>
<th>Material</th>
<th>$T_{\text{melt}}$ [K]</th>
<th>$C_p @ 2500 \text{ K}$ [kJ/kgK]</th>
<th>$\Delta T$ Required for 1 MJ/kg</th>
</tr>
</thead>
<tbody>
<tr>
<td>Graphite</td>
<td>3923</td>
<td>2.15</td>
<td>475</td>
</tr>
<tr>
<td>Boron Carbide</td>
<td>2700</td>
<td>2.68</td>
<td>380</td>
</tr>
<tr>
<td>Silicon Carbide</td>
<td>2818</td>
<td>1.01</td>
<td>740</td>
</tr>
<tr>
<td>Boron Nitride</td>
<td>3273</td>
<td>1.98</td>
<td>510</td>
</tr>
</tbody>
</table>

- Simplified engineering suitable for time constrained development – Low TRL level of other options
- “…moderate yet acceptable performance” - Kennedy 2002

**ISUS Data Analysis**

- Seven minute “steady” burn corresponds to an “effective” energy storage density of **0.5 MJ/kg**
- When the RAC achieves 1 MJ/kg, exit temp has dropped by > 25% and Isp has dropped by 15%
- ISUS spec for thermionic hot shoe temperature was 1900 – 2200 K. If allowed for a radiently coupled TPV system, this would correspond to a > 50% decrease in power output
# Potential High-Temp Phase Change Materials

<table>
<thead>
<tr>
<th>Material</th>
<th>$T_{melt}$ [K]</th>
<th>$\Delta H_{fus}$ [MJ/kg]</th>
<th>$k_{th} @ T_{melt}$ [W/mK]</th>
</tr>
</thead>
<tbody>
<tr>
<td>MgF2</td>
<td>1536</td>
<td>0.94</td>
<td>3.8</td>
</tr>
<tr>
<td>Beryllium</td>
<td>1560</td>
<td>1.31</td>
<td>69</td>
</tr>
<tr>
<td>Silicon</td>
<td>1687</td>
<td>1.79</td>
<td>20</td>
</tr>
<tr>
<td>Nickel</td>
<td>1728</td>
<td>0.3</td>
<td>83</td>
</tr>
<tr>
<td>Scandium</td>
<td>1814</td>
<td>0.31</td>
<td>16</td>
</tr>
<tr>
<td>Chromium</td>
<td>2180</td>
<td>0.4</td>
<td>48</td>
</tr>
<tr>
<td>Vanadium</td>
<td>2183</td>
<td>0.45</td>
<td>51</td>
</tr>
<tr>
<td>Boron</td>
<td>2350</td>
<td>4.6</td>
<td>10</td>
</tr>
<tr>
<td>Ruthenium</td>
<td>2607</td>
<td>0.38</td>
<td>80</td>
</tr>
<tr>
<td>Niobium</td>
<td>2750</td>
<td>0.29</td>
<td>82</td>
</tr>
<tr>
<td>Molybdenum</td>
<td>2896</td>
<td>0.38</td>
<td>84</td>
</tr>
</tbody>
</table>

- **Silicon**
  - Moderate Performance
  - 330s $I_{sp}$

- **Boron**
  - High Performance
  - 390s $I_{sp}$
# System Comparison

Bi-Modal System Performance Parameters

100 kg Microsatellite - 100 W continuous power draw

<table>
<thead>
<tr>
<th>Silicon System</th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Thermal Collection</strong></td>
<td>5.3 kg</td>
</tr>
<tr>
<td><strong>Thermal Storage</strong></td>
<td>3.3 kg</td>
</tr>
<tr>
<td><strong>Power</strong></td>
<td>6.7 kg</td>
</tr>
<tr>
<td><strong>Propellant</strong></td>
<td>36.7 kg</td>
</tr>
<tr>
<td><strong>Tankage / Thruster</strong></td>
<td>6.1 kg</td>
</tr>
<tr>
<td><strong>Prop. / Power Total</strong></td>
<td>58.2 kg</td>
</tr>
<tr>
<td><strong>Payload Mass</strong></td>
<td>41.8 kg</td>
</tr>
</tbody>
</table>

\( M_{\text{Propulsion & Power}} \approx 58\% \)

1500 \(m/s\) \(\Delta V\)

- **Thermal Collection**
  - Primary concentrator
  - Support structure
  - Fiber optics

- **Thermal Storage**
  - PCM
  - Insulation

- **Power**
  - TPV cells
  - Radiator Panels

- **Propellant**
  - Liquid Ammonia

- **Tankage / Thruster**
  - Titanium Tank
  - Piping
  - Nozzle and Heat Exchanger
  - Reinforcements
# System Comparison

**Bi-Modal System Performance Parameters**

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<tr>
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<td><strong>Thermal Storage</strong></td>
<td><strong>Thermal Storage</strong></td>
</tr>
<tr>
<td>3.3 kg</td>
<td>1.9 kg</td>
</tr>
<tr>
<td><strong>Power</strong></td>
<td><strong>Power</strong></td>
</tr>
<tr>
<td>6.7 kg</td>
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  - Piping
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  - Reinforcements

\[ 1500 \text{ m/s } \Delta V \]

\[ 1850 \text{ m/s } \Delta V \]
System Comparison

The image shows a graphical comparison of system performance metrics, likely related to aerospace technology, with axes labeled "Response Time, days" and "ΔV, m/s". The graph includes various data points and trendlines, each representing different systems or technologies.

Detailed labels on the graph include:
- "Other Body Capture"
- "GEO to L4/L5"
- "GTO to GEO"
- "Lunar Insertion"
- "LEO to GTO"

Legend items include:
- 20 N Hydrazine
- 1 N Hydrazine
- Standard STP
- 100-Watt Hall Thruster
- Si-STP
- B-STP
- B-STP, optimistic

The image also includes a note: "DISTRIBUTION STATEMENT A: Approved for public release; distribution is unlimited. PA#XXXXXX"
System Comparison

- 10% more ΔV than chemical systems
- Delivery time: 23 days
- Uses existing technologies and weights
System Comparison

- 35% more $\Delta V$ than chemical systems
- Delivery time: 34 days
- Uses existing technologies and weights
System Comparison

- Extrapolating future tech improvements
- 60% more $\Delta V$ than chemical systems
- Delivery time: under 40 days
- Out of reach of a chemical system
Demonstrate a Proof of Concept Latent Heat Thermal Energy System Using Molten Silicon

- No experiments to-date directly targeting energy storage applications

- Mentioned as a potential buffer / storage material for TPVs
  - Woodall 1982 - IBM patent
  - Chubb et al. 1996 - white paper, “ideal storage material”

- Brief mentions in the solar thermal literature
  - Laug et al. 1995 – Initial bi-modal study
  - Kennedy 2002 – TRL level not sufficient
  - Abbot 2001 – Trade study

1) Facility Development
2) Materials Selection
3) Modeling Capability and Analysis
4) Experimental Demonstration
USC Solar Furnace

- First-surface spherical concentrator
  - $r_c = 124''$
  - SiO$_2$ coating optimized to the solar spectrum
  - Manufactured by DOTI Optics
- 3600 in$^2$ usable concentrator area
- 12 ft x 8 ft computer controlled heliostat
- COTS and surplus components
- Delivers 800-1100 W in a 1” spot
Solar Flux At Best Location

- Peak concentration ratios 4000:1
- Tailored for maximum power delivery in a 1” diameter spot
- Optimized experimental placement using CCD solar flux mapping to compensate for spherical aberrations

Geometric "Focus" : 62"
Minimum Ø : 59.8"
Best Performance : 60.5"
Materials Studies

- Possible to draw from semi-conductor industry knowledge

- **Boron nitride** has a self limiting reaction with molten silicon (*also compatible with liquid boron*)
  - Formation of $\text{Si}_3\text{N}_4$ limited at 2% boron saturation in the silicon bulk
  - Low level boron contamination expected to have little effect on silicon recrystallization

- Graphite can be used with carbon contamination on the order of 20 ppm
  - Density must be $> 1.75 \text{ g/cc}$
  - Grain size must be $< 50 \mu\text{m}$
Testing Geometry

- Cylindrical geometry for ease of manufacture and simplified modeling
- Can be manufactured in house from COTS components
- Sized for 9 g of silicon, however, this is not limited by solar furnace power
- Does not make use of radiation shielding
- Integrated Type C and Type K thermocouples
• 2D Axisymmetric about $\theta$

• Radiation and convection boundary conditions

• Neglects PCM density change and void formation

• Latent heat handled by the “enthalpy method”
Predictive Modeling

- 2D Axisymmetric about θ
- Radiation and convection boundary conditions
- Neglects PCM density change and void formation
- Latent heat handled by the "enthalpy method"

Temperature (K)

<table>
<thead>
<tr>
<th>Time (s)</th>
<th>Image</th>
</tr>
</thead>
<tbody>
<tr>
<td>20</td>
<td><img src="t=20s.png" alt="Image" /></td>
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<tr>
<td>40</td>
<td><img src="t=40s.png" alt="Image" /></td>
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<tr>
<td>60</td>
<td><img src="t=60s.png" alt="Image" /></td>
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<td>80</td>
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<tr>
<td>100</td>
<td><img src="t=100s.png" alt="Image" /></td>
</tr>
<tr>
<td>120</td>
<td><img src="t=120s.png" alt="Image" /></td>
</tr>
</tbody>
</table>
Experimental Testing
Experimental Testing

Testing Procedure

• Bake out at 300 °C and 30 mTorr to evaporate “proprietary water based binder” in the cast ZrO$_2$ ceramic

• Fill chamber with 150 Torr of Argon
  - Required to suppress ZrO$_2$ + C reaction
  - Prevents irreparable damage to quartz chamber window

• Gradually increase power until thermal equilibrium is achieved

• Use “shutter curtain” to quickly cut power and record cooling curve
Experimental Data

![Graph showing experimental data over time and temperature](image)
Experimental Data

Expansion Damage!

Graph showing temperature (K) vs. time (s) with two sets of experimental data: Experimental Data 1 (black line) and Experimental Data 2 (blue line). The graph highlights a point of expansion damage.

Additional images include a close-up of a glowing object, a cracked experimental sample, and a damaged frame.
Experimental Data

![Graph showing temperature over time with MATLAB Model comparison.]

- **Experimental Data Range**
- **MATLAB Model**

**Time, s**: 0 to 250
**Temperature, K**: 1000 to 1800
Successful early tests using silicon powder and chips achieved "fill factors" of approximately 60%.

Tests were performed reducing fill factor from 100% to 80% in 5% increments.

Audible cracking provided an indication of damage during freezing.

No tests < 100% showed macro scale damage. However, all sections showed damage to the inner BN liner.
Damage Mitigation: Fill Factor
Damage Mitigation: HD Graphite

- Production of large silicon ingots precisely controls the phase front
- Graphite crucibles are used with contamination on the order of 20 ppm
- Graphite must be > 1.75 g/cc with a grain size < 50 µm
- Benefits from wetting behavior?
• Only allow silicon to partially freeze?
• Experimentally problematic due to assured failure at the end of testing
• Duty cycle must be matched to eclipse period and power available must consider thermal inertia
Convective Coupling

- The ISUS system was capable of maintaining a constant temperature with a sensible heat medium due to “extra length” being designed into the heat exchanger.
- The design spec of the ISUS RAC has the potential for 0.72 MJ/kg with a 600 K ΔT (double the original design spec).
- In reality, achieves 0.46 MJ/kg if the steady output region is considered “usable.”
- To date, no discussion of latent heat thermal energy storage discusses advantageous convective coupling.
- Will use commercial multi-physics software (Star-CCM+) to replicate a ISUS heat exchanger channel and switch storage to latent heat.
- Seek a quantification of convective coupling benefit.

H₂ at 0.0087 g/sec and 29 psia

Temperature distribution over time.
Future Work

Convective Coupling

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\( \text{H}_2 \text{ at } 0.0087 \text{ g/sec and 29 psia} \)
Conclusions

• Bi-modal solar thermal propulsion has the potential to dramatically extend the microsatellite operating envelope
  
  o  > 1 km/s ΔV  
  o  Delivery time measured in *days not years*

• Silicon and boron based thermal energy storage have been frequently mentioned in the literature but lack development due to schedule and funding constraints

• When complete, experiments will bring latent heat thermal energy storage for STP to a similar TRL level as sensible heat options
  
  o  *Volumetric expansion?*  
  o  *Convective coupling?*

• Thorough experimental investigation into high temperature latent heat thermal energy storage will provide a road map for future solar thermal system designers
Solar Thermal Propulsion Overview

- “Solar Powered Space Ship” proposed by Krafft Ehricke
- 7500 kg spacecraft with a two man crew
- AFRPL funded investigation at Electro-Optical Systems (EOS) in 1963 produced solar heated H₂ at approx. 2300 K
- Work halted due to concerns about “awkward” vehicle design and integration issues
- Funding was shifted to a competing advanced concept
Solar Thermal Propulsion Overview

- Space Shuttle – “represents a national commitment to extended operations in space” *Selph 1981*

- 1979 Rockwell report, funded through AFRPL, concludes a solar thermal rocket is possible and recommends near term production

- Vehicle integration was greatly simplified by a centrally located solar receiver and inflatable concentrators

- Compared performance of **28,100 kg**, shuttle launched spacecraft for LEO-GEO transfer

<table>
<thead>
<tr>
<th>Engine Type</th>
<th>LO₂ H₂</th>
<th>Ion</th>
<th>Solar 1</th>
<th>Solar 2</th>
</tr>
</thead>
<tbody>
<tr>
<td>ΔV (m/s)</td>
<td>4,270</td>
<td>5,850</td>
<td>5,850</td>
<td>4,800</td>
</tr>
<tr>
<td>Isp (sec)</td>
<td>475</td>
<td>2,940</td>
<td>872</td>
<td>872</td>
</tr>
<tr>
<td>Trip Time (days)</td>
<td>5</td>
<td>180</td>
<td>14</td>
<td>40</td>
</tr>
<tr>
<td>Payload to Geo (kg)</td>
<td>9,250</td>
<td>20,000</td>
<td>9,300</td>
<td>13,200</td>
</tr>
</tbody>
</table>
Solar Thermal Propulsion Overview

1956 1980s 1990s 2000s Today

- AFRPL funded effort for experimental demonstration based on findings of Rockwell report
- Rocketdyne contracted to produce a solar thermal thruster using coiled rhenium tubing with a target exit temperature of 2705 K
- Solar furnace problems limited testing temperatures to approximately 1800 K
- AFRPL declared technology “feasible” but development was slowed in 1989 due to budget cut-backs
- Note that the design does not include a means of thermal energy storage

“...time spent traversing the Earth shadow results in a trip-time increase of approximately 10% at no increase in propellant expended.”

Ethridge 1979
A bi-modal nuclear thermal system capable of providing propulsive and electrical power was proposed in the early 1990s.

Integrated upper stage design supplies electrical power to the payload after orbit transfer.

Reduced mass: potential for launch vehicle “step down”

**Delta II 7925**
- $50M in 1995
- 1800 kg to GTO

**Titan IIIG**
- $18-30M in 1995
- 1000 kg to GTO

Due to waning interest in nuclear thermal research, AFRPL considered the concept with a solar thermal architecture.

Sought to quickly reduce the cost of Air Force space operations using existing technology.
Solar Thermal Propulsion Overview

- Integrated Solar Upper Stage Program (ISUS) initiated in 1994
- ISUS program targeted a “militarily” useful payload on orbit by 1998 – very optimistic
- Performed a ground test of a prototype Receiver-Absorber-Converter (RAC)
- RAC incorporated sensible heat thermal energy storage – necessitated by the bi-modal design
- Succeeded in recording data for hot flow hydrogen testing
- Program closed in 1998 – followed by Boeing Solar Orbit Transfer Vehicle (SOTV) and the STP Critical Flight Experiment at NASA Marshall
Solar Thermal Propulsion Overview

Solar Collection
Primary concentrated sunlight enters through a Secondary Connector

Solar energy is absorbed at the RAC cavity walls

Graphite thermal storage heats as the sunlight is absorbed

Propulsion
Propellant enters a preheater through an inlet line

Power
Thermionic converters produce electricity as heat from the RAC passes through them. Waste heat is radiated to space by heat pipes.

A layer of insulation reduces heat leakage from the RAC

The propellant heating is completed in the RAC body

Hot propellant gathers in the outlet plenum and then passes to the thruster where it expands and exhausts.
Solar Thermal Propulsion Overview

- Concept shifted to microsatellites (10-100 kg) in an effort to finally mount a space demonstration
- Project headed by Kennedy, a veteran of the ISUS program, at Surrey Space Center
- Proposed the use of non-cryogenic propellants such as N₂H₄ and NH₃ and “packed bed” sensible heat thermal energy storage
- Achieved experimental NH₃ temperatures approaching 2000 K
- Other small scale research efforts
  - Thin film concentrator and Mo receiver work at JAXA
  - Fiber optic coupling work at Physical Sciences Inc
Drawing from Kennedy’s microsatellite study, a review by the AFRPL advanced concepts group identified STP as a promising candidate for high performance microsatellite missions.

A **bi-modal microsatellite** configuration is proposed and further study is recommended.

Microsatellite scaling distinguishes STP.

Large $\Delta V (> 1 \text{ km/s})$ possible

**Expand the Microsatellite Operating Envelope**

- Expand possible “piggy-back” launch options
- GEO Insertion: $\sim 1760 \text{ m/s}$
- Near Escape Missions: $\sim 770 – 1770 \text{ m/s}$

*Possible with EP, however, STP offers a much shorter burn time and higher maneuverability*
• 3700 Nodes

• \( dt = 0.00025 \) seconds

• Approx. 14 hours runtime for a 300 second simulation

\[
\Delta z
\]

\[
\Delta r
\]

\[
\begin{align*}
(m, n+1) & \quad (m-1, n) \\
(m, n) & \quad (m, n-1) \\
(m+1, n) & \quad (m-1, n)
\end{align*}
\]

\[
V = 1.3 \times 10^{-5} \text{ cc} \\
m = 2.65 \times 10^{-5} \text{ g}
\]
MATLAB Model

Conduction in the “r” Direction

\[ q = \frac{\left( T_\beta - T_\alpha \right)}{\ln(r_2/r_1) + \ln(r_3/r_2)} \cdot \frac{L}{2\pi Lk_\alpha} + \frac{L}{2\pi Lk_\beta} \]

Conduction in the “z” Direction

\[ q = A\left( T_\beta - T_\alpha \right) \cdot \frac{1}{L\left(1/k_\alpha + 1/k_\beta\right)} \]

\[ A = \pi\left(r_2^2 - r_1^2\right) \]

At Each Time Step

\[ q_{net} = q_{left} + q_{right} + q_{down} + q_{up} \]

\[ \Delta T = \frac{q_{net}dt}{mc_p} \]

Node With PCM

\[ \Delta T = \frac{q_{net}dt}{mc_p} + \text{LatentHeat}(rr, zz) \]
MATLAB Model

Radiation Boundary Condition

\[ q_{rad} = -\sigma \varepsilon A \left( T_{node}^4 - T_{amb}^4 \right) \]

Convective Boundary Condition

\[ q_{conv} = -hA \left( T_{node} - T_{amb} \right) \]

\[ q_{net} = q_{left} + q_{right} + q_{down} + q_{up} + q_{rad} + q_{conv} \]

\[ \Delta T = \frac{q_{net} dt}{mc_p} \]

Vertical Plate \( \Rightarrow h \approx 6.7 \text{ (W/m}^2\text{K)} \)

\[ h = \frac{k}{L} \left( 0.825 + \frac{0.387Ra_L^{1/6}}{(1 + (0.492/Pr)^{9/16})^{8/27}} \right)^2 \]  

(Churchill and Chu)

Assume laminar flow with Argon at 500 K and 150 Torr \( (Ra_L = 10^6) \)

Horizontal Cylinder \( \Rightarrow h \approx 5.7 \text{ (W/m}^2\text{K)} \)

\[ h = \frac{k}{D} \left( 0.6 + \frac{0.387Ra_D^{1/6}}{(1 + (0.559/Pr)^{9/16})^{8/27}} \right)^2 \]  

(Churchill and Chu)

Assume laminar flow with Argon at 500 K and 150 Torr \( (Ra_L = 10^6) \)
MATLAB Model

Node-Node Reflected VF, total rays = 30000000

Node Number where Ray Strikes
Node Number where Ray Launches

0
0.005
0.01
0.015
0.02

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Accuracy Requirements in the Literature

- ± 0.3° - Hukuo 1957
- ± 0.1° - Ethridge 1979
- ± 0.5° - Holmes 1995
- ± 0.1° - Kennedy 2004

Note idea image for a parabolic concentrator

\[ d = 2f \sin(\frac{\theta_{\text{sun}}}{2}) \]
Pointing Error Calculations

Hukuo and Mii 1957

Figure 1 — Section of a parabolic mirror.

Figure 2 — Solar image produced by a ray which is reflected by the paraboloid with the angle $\beta$. 
Heliostat Coverage

January 15th

Concentrator
Mirror Projection - 1:30
Mirror Projection - 2:30
Mirror Projection - 3:30
Mirror Projection - 4:30
Heliostat Coverage

June 15th

- Concentrator
- Mirror Projection - 1:30
- Mirror Projection - 2:30
- Mirror Projection - 3:30
- Mirror Projection - 4:30
CCD Diagnostic Method

• Read pixel intensities from CCD after subtracting representative “dark frame”

• Convert using black body calibration from counts/µs to $W/m^2nm$ at 980 nm

• Account for reflectivity of pseudo-Lambertian surface

• Convert from $W/m^2nm$ to # of Suns
  ➢ Weight against filter band pass
  ➢ Use ASTM G173 data to scale 980 nm values with the full spectrum
  ➢ Multiply by ASTM G173 standard insolation to get pixel reading in W/m²

• Compare to locally measured insolation to get map of concentration ratios

Sony ICX445 CCD - 1.2 MP
16-bit mono output format
Images captured at 640 x 480
<table>
<thead>
<tr>
<th>Solar Thermal w/o Energy Storage</th>
<th>Chemical Thrusters</th>
<th>Electric Propulsion</th>
</tr>
</thead>
<tbody>
<tr>
<td>• Eliminated PCM and TPV</td>
<td>• Astrium Hydrazine Monoprop</td>
<td>• XHT-100 Hall Effect Thruster</td>
</tr>
<tr>
<td>• Reduced solar collector size</td>
<td>• Commercially available 1 N and 20 N models</td>
<td>• 95 W power draw</td>
</tr>
<tr>
<td>• Added photovoltaic panels and batteries</td>
<td>• Isp: 220-230 s</td>
<td>• Isp: 750-1000 s</td>
</tr>
<tr>
<td>• Used NASA year 2020 specific power projections for PV</td>
<td>• Removed thermal energy collection and storage system</td>
<td>• 3 – 10 mN Thrust</td>
</tr>
<tr>
<td></td>
<td>• Added photovoltaic panels and batteries</td>
<td>• Removed thermal energy collection and storage system</td>
</tr>
<tr>
<td></td>
<td>• Used NASA year 2020 specific power projections for PV</td>
<td>• Added photovoltaic panels and batteries</td>
</tr>
<tr>
<td></td>
<td></td>
<td>• Used NASA year 2020 specific power projections for PV</td>
</tr>
</tbody>
</table>

**Identical Mass Fractions** \( M_{\text{Propulsion & Power}} = 58\% \)

**Total \( \Delta V \) and Delivery Time are Primary Comparison Metrics**
Ethridge Orbit Calculations

Two Impulse
One Perigee Burn
One Apogee Burn

$T/W > 0.01$

Multi Impulse
More Than One Perigee Burns and More Than One "Insertion" Burns Near Final Apogee

$T/W < 0.1$

Continuous Burn
Spiral Trajectory

$T/W < 0.001$

LEO to GEO

$14000 \leq \Delta V \leq 17000 \text{ FPS}$

Trip Time $< \text{ DAY}$

$14000 \leq \Delta V \leq 19200 \text{ FPS}$

Trip Time $> \text{ DAYS}$

$\Delta V \approx 19200 \text{ FPS}$

Trip Time $> \text{ WEEK}$
Kennedy Orbit Calculations

- 10% “On Time”
- 100 kg satellite
- Launch from Ariane 5 into GTO (350 x 35,717 km - 7°)
- Transfer to 0° at 116 °E
- Assumes ½ N thrust and 400 s $I_{sp}$
- 48 kg for JUST solar thermal engine and propellant

<table>
<thead>
<tr>
<th>Start Date</th>
<th>1 April 2005, 00:00:00.00 (Julian Date 2453461.5)</th>
</tr>
</thead>
<tbody>
<tr>
<td>End Date</td>
<td>6 May 2005, 09:11:16.96 (JD 2453496.88)</td>
</tr>
<tr>
<td>Elapsed Time</td>
<td>35 days, 9 hrs., 11 min.</td>
</tr>
<tr>
<td>Number of Maneuvers</td>
<td>58 (51 apogee kicks, 7 plane changes at node crossings).</td>
</tr>
<tr>
<td>Two-orbit “hold” of 42 hrs., 20 min. introduced after apogee kick 48 to attain proper orbital phasing at GEO</td>
<td></td>
</tr>
<tr>
<td>Total Velocity Change</td>
<td>1,761 m/s</td>
</tr>
<tr>
<td>Propellant Consumption</td>
<td>36.184 kg</td>
</tr>
<tr>
<td>Final Mass</td>
<td>63.816 kg</td>
</tr>
<tr>
<td>Engine “On-Time”</td>
<td>80 hrs., 33 min.</td>
</tr>
</tbody>
</table>
Secondary Concentrator

- Defined by parametric equations given by *Welford and Winston 1978*

- **CANNOT** be used to increase power due to low f/d ratio for the concentrator

- **CAN** be used to increase concentration ratio

<table>
<thead>
<tr>
<th>Diameter</th>
<th>Max Angle</th>
<th>Minimum Diameter</th>
<th>% Area Increase</th>
<th>Max Ratio</th>
<th>Minimum Spot</th>
<th>Spot Change</th>
<th>C Ratio Change</th>
</tr>
</thead>
<tbody>
<tr>
<td>70</td>
<td>33.0</td>
<td>1.66</td>
<td>0%</td>
<td>3.24</td>
<td>0.914</td>
<td>0.84</td>
<td>5%</td>
</tr>
<tr>
<td>75</td>
<td>35.162</td>
<td>1.81</td>
<td>15%</td>
<td>2.99</td>
<td>1.046</td>
<td>1.09</td>
<td>-16%</td>
</tr>
<tr>
<td>80</td>
<td>37.7</td>
<td>2.09</td>
<td>32%</td>
<td>2.67</td>
<td>1.220</td>
<td>1.49</td>
<td>-29%</td>
</tr>
</tbody>
</table>
Ammonia and Hydrazine

\[ 3N_2H_4 \rightarrow 4NH_3 + N_2 - 336 \text{ kJ} \]

\[ 4NH_3 \rightarrow 2N_2 + 6H_2 + 184 \text{ kJ} \]

Net 1.6 MJ/kg

<table>
<thead>
<tr>
<th>( \alpha_D )</th>
<th>Isp</th>
</tr>
</thead>
<tbody>
<tr>
<td>0</td>
<td>253</td>
</tr>
<tr>
<td>0.2</td>
<td>274</td>
</tr>
<tr>
<td>0.4</td>
<td>289</td>
</tr>
<tr>
<td>0.8</td>
<td>312</td>
</tr>
<tr>
<td>1</td>
<td>322</td>
</tr>
</tbody>
</table>

- Incomplete dissociation will lower performance
- Equilibrium calculations for 1500 K solar thermal thruster (Colonna et. al. 2005)
- Note, hydrazine thrusters typically have \( \alpha_D \approx 55\% \)

\[ V_e = I_{sp} g_o = \sqrt{\frac{T_o R}{M} \frac{2 \gamma}{\gamma - 1} \left[ 1 - \left( \frac{p_e}{p_o} \right)^{(\gamma - 1) / \gamma} \right]} \]
For each 200 km Orbit

**Energy Values**

- **Solar Concentrator**
  - 1.57 m²
  - 1300 W/m²
  - 50 min “on sun”
  - 6110 kJ Total

- **Incoming Sunlight**

- **Fiber-optic Transmission**
  - 70% Efficiency
  - 4280 kJ to TSM

- **High Temperature Phase Change TSM**

- **Ammonia Propellant**
  - Ammonia at $\alpha = 100\%
  - 540 kJ Out
  - $I_{sp} = 330$ s

- **25% Loss**
  - 1070 kJ

- **Insulation Losses**

- **20% Efficiency**
  - 2136 kJ Out

- **Waste Heat**

- **Thermophotovoltaics**

- **Electricity Output**
  - 534 kJ Out

- **Electric Output**

- **Heat Exchanger**

- **Propulsion Power Out**

**η_{total} \approx 17.5%**
Tech Comparison Metrics

• Satellite is sized for a 200 km circular orbit
  - Storage sized for approx. 36 min eclipse

• Assumes 20% total electrical system efficiency

• Assumes 70% thermal collection efficiency

• Approximates impulsive burn profile with a 5% firing rule