Dual-Mode Propulsion System Enabling CubeSat Exploration of the Solar System

NASA Innovative Advanced Concepts

2014 NIAC Symposium

Stanford University

February 4 – 6, 2014

Nathan Jerred, Troy Howe & Steven Howe

Center for Space Nuclear Research

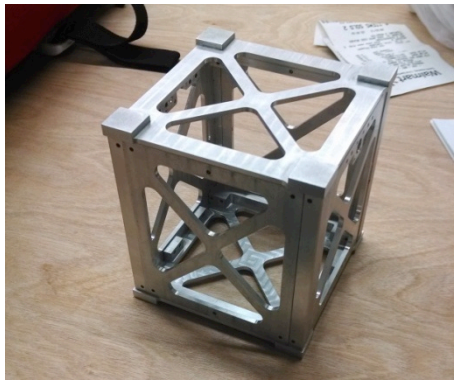
Adarsh Rajguru

University of Southern California

CENTER FOR SPACE NUCLEAR RESEARCH

Inspiration

- Limited budgets make large missions more difficult to fund
  - *achieve more science-per-dollar*
- Develop technologies enabling reliable, compact exploration platforms
  - *long-lived and long-ranged mobile platform*
  - *off-the-shelf propulsion system*
  - *propulsion system for micro-satellite payloads*
- Target small launch vehicles
  - *less than 1,000 kg to LEO*
- Enable affordable deep space exploration



1U CubeSat frame

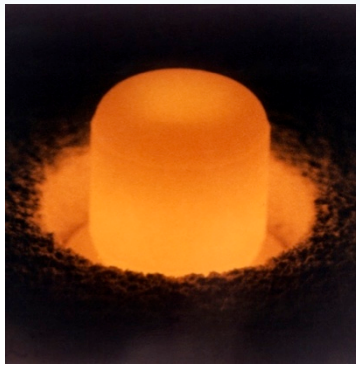
GOAL

- *deliver 6U CubeSat payload to Enceladus orbit (~15 kg)*
- *develop an appropriately sized propulsion system concept*
- *develop the mission architecture*

Concept
energy source

- Radioisotope Decay
  - *proposed radioisotope: $^{238}PuO_2$*
    - *heritage use within NASA*
  - *high specific energy:* ***1.6 x 10^6 MJ/kg (thermal)***
    - *Chemical propellants: 10 MJ/kg (thermal)*
    - *RTG: 9.6x10^4 MJ/kg (6%) vs Li-ion: 0.72 MJ/kg (electric)*
  - *poor specific power:* ***0.392 W/g*** *[$^{238}PuO_2$]*

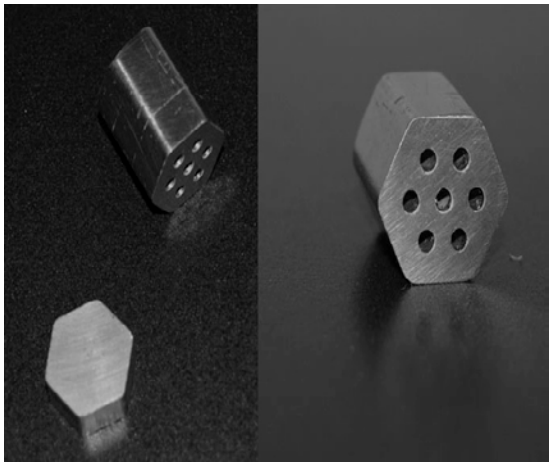
$^{238}PuO_2$ Pellet



DOE

- Fuel Containment
  - *fuel encapsulated in a tungsten-based matrix*
    - *provides high strength & toughness*
  - *can provide great energy density*

Tungsten-based CERMET



O'Brien et al.

NIAC NASA Innovative Advanced Concepts

Center for Space Nuclear Research

Approach

- Radioisotope-Based Core
  - *decay energy accumulated within central core, i.e, thermal capacitor*
  - *direct propellant heating for propulsion*
    - *radioisotope thermal rocket (RTR)*
  - *thermal energy converted to electrical energy*
    - *electric propulsion*
    - *power generation*
  - *accumulated energy is depleted through each impulse*

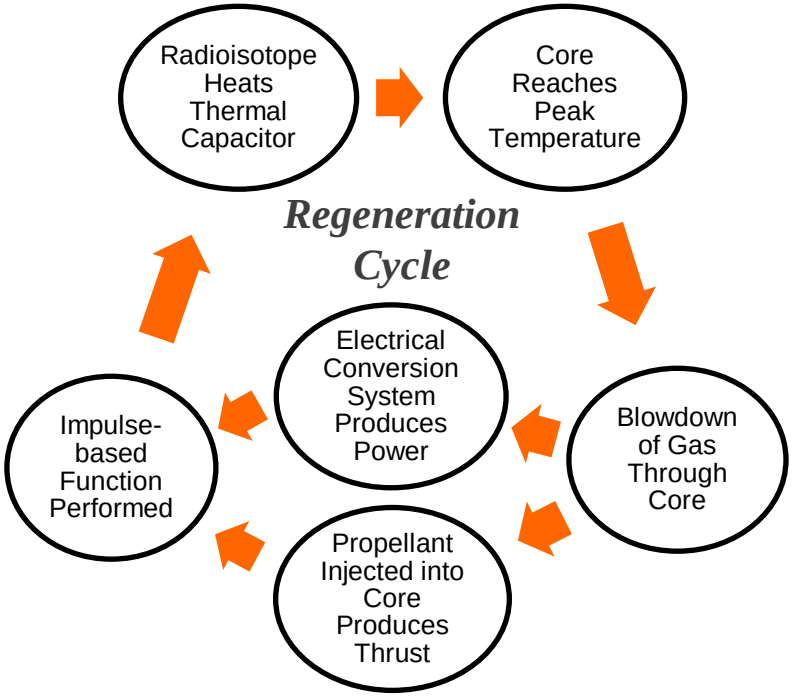
artistic rendering of the concept in Earth orbit

Concept
energy storage

- Thermal Capacitor Qualifications
  - *high thermal storage, high thermal conductivity, high melting temperature*
  - *sensible heat → based on a material's heat capacity*
  - *latent heat → based on energy needed to change a material's phase*

- Core Material
  - *Silicon identified as a suitable material*
    ΔH_{fusion} = 1.8 MJ/kg
    T_{melt} = 1687 K

Allows for an operational temperature of around 1700 K

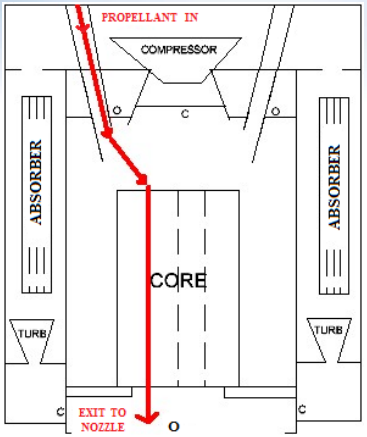


proposed cycle process

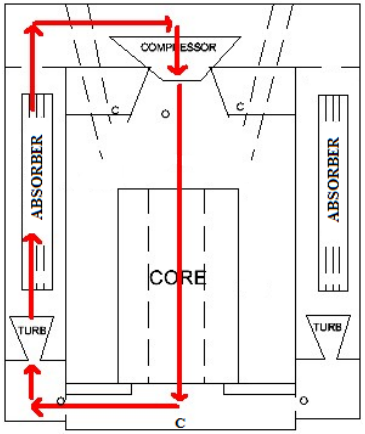
Concept
dual-mode

- Direct Propellant Heating
  - *provides thermal-based propulsion*
  - *heated propellant expelled through nozzle creating thrust*
- Energy Conversion
  - *closed loop Brayton cycle*
    - *10's of kW per pulse*
    - *multiple pulses per day*

dual Brayton engines



thermal propulsion flow schematic

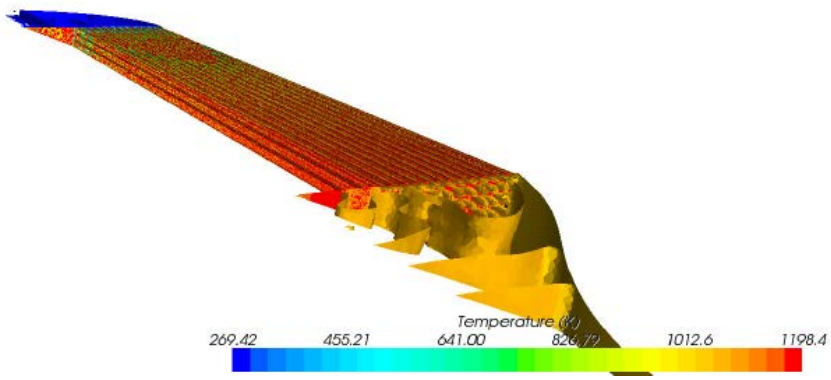


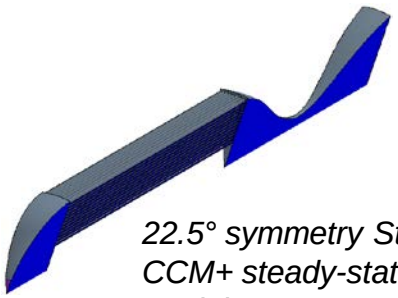
electrical conversion flow schematic

Optimization

- Previous work performed on a Martian exploration probe

Starr-CCM+ steady-state simulation of Mars Hopper concept





22.5° symmetry Starr-CCM+ steady-state model

- Thermal interactions within the core
  - *details core sizing such as isotope loading, capacitor size, insulation, etc.*
- Thermal hydraulics between the core and flowing gas
  - *details performance of propellant yielding chamber pressure & temperature aiding in nozzle development*
  - *details performance of working fluid for energy conversion to help determine power generation and efficiencies*

Mission Design
instrumentation & communication

Proposed Instrument Package

Instrument	Mass (g)	Volume (cm^3)	Power (W)
Thermal Imaging Radiometer	1,000's	1,000's	10's
Infrared Spectrometer	100's	100's	-
High Resolution Camera (MAC)	100's	100's	1's
Mass Spectrometer	100's	10's	1's

6U payload frame

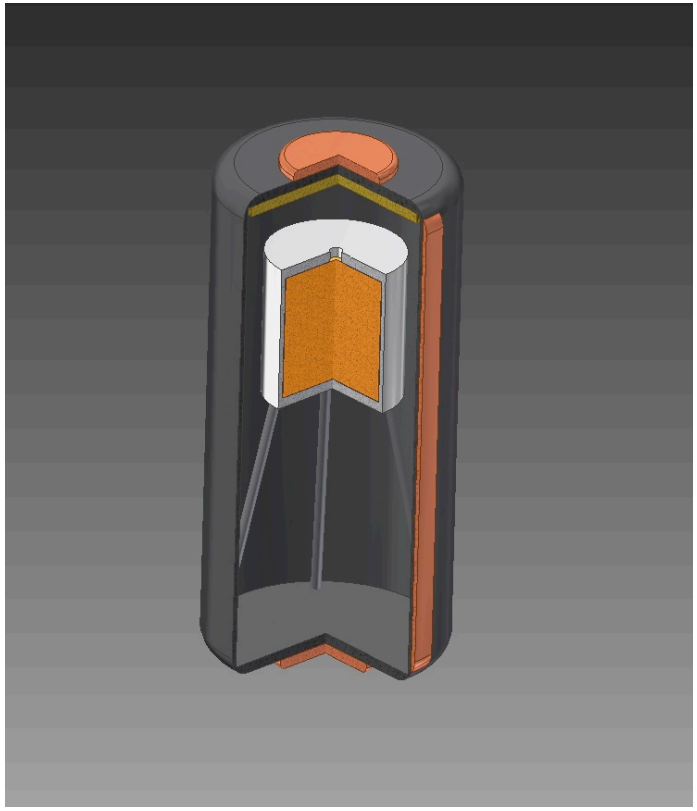
communication system

Communication System

Item	Symbol	Units	Value
Frequency	f	GHz	27.50
Transmission Power	**Pt**	**Watts**	**25000**
Transmission Antenna Dia.	Dt	m	1.5
Trans. Antenna Gain (net)	Gt	dBi	50.16
Prop. Path Length	S	km	$1.27(10)^9$
Space Loss	Ls	dB	-303.30
System Noise Temp.	Ts	K	84.10
Data Rate	**R**	**Mbps**	**1**
SNR	Eb/No	-	5.94

Payload Power

A compact Radioisotope Thermo-Photovoltaic (RTPV) power source ~being studied at the CSNR

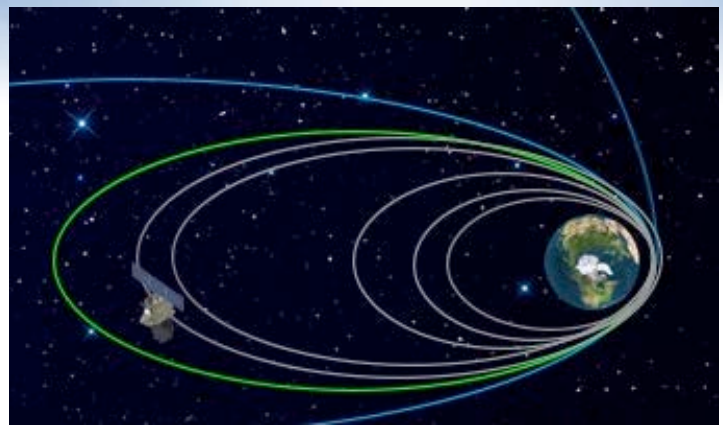


estimated RPS for
5 We – 400g, "D cell" size

- Solid-state energy conversion
  - *continuous energy production*
  - *no moving parts & long mission lifetimes*
- Encapsulation method allows for modification in the source size
  - *can be sized to accommodate payload needs*
- Tungsten matrix shields harmful radiation
  - *power sources can be housed among instrumentation*
  - *provides high temperature strength*
- Emitter material or coating can be integrated with the CERMET fuel
  - *increase conversion efficiency*

Trajectory

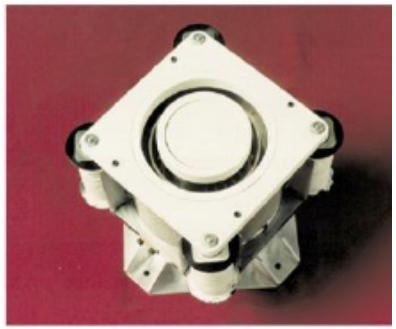
- Thermal Propulsion Mode
  - *provides high thrust & moderate Isp*
  - *impulse function allows for phasing maneuvers (perigee pumping) to achieve Earth escape*
- Electric Propulsion Mode
  - *provides high Isp for interplanetary travel*
  - *allows for shorter transit times*
  - *four 2.2 kW Hall Effect thrusters*



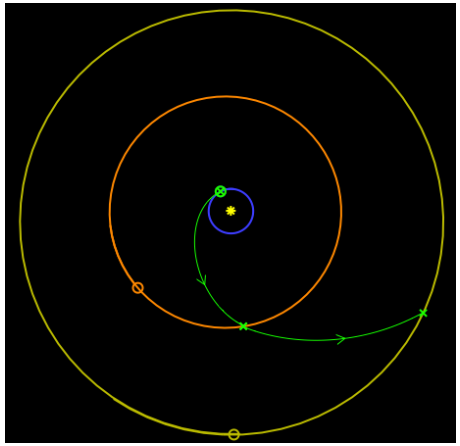
graphic of Earth-based phasing maneuvers

Earth – Jupiter – Saturn

- Earth Departure: **Jan-18-2018**
- Saturn Arrival: **July-11-2023**
- Total Duration: **5.48 years**
- Injection ΔV ($Earth_{escaped}$): **6.42 km/s**
- Jupiter Gravity Assist: **0.12 km/s**
- Post Injection ΔV ($Saturn_{capture}$): **0.54 km/s**
- Total ΔV required by propulsion system = **6.96 km/s**



AeroJet BPT-2000 Hall Effect thruster



possible trajectory to Saturn

Performance

PRELIMINARY RESULTS

based on Earth escape

Operation

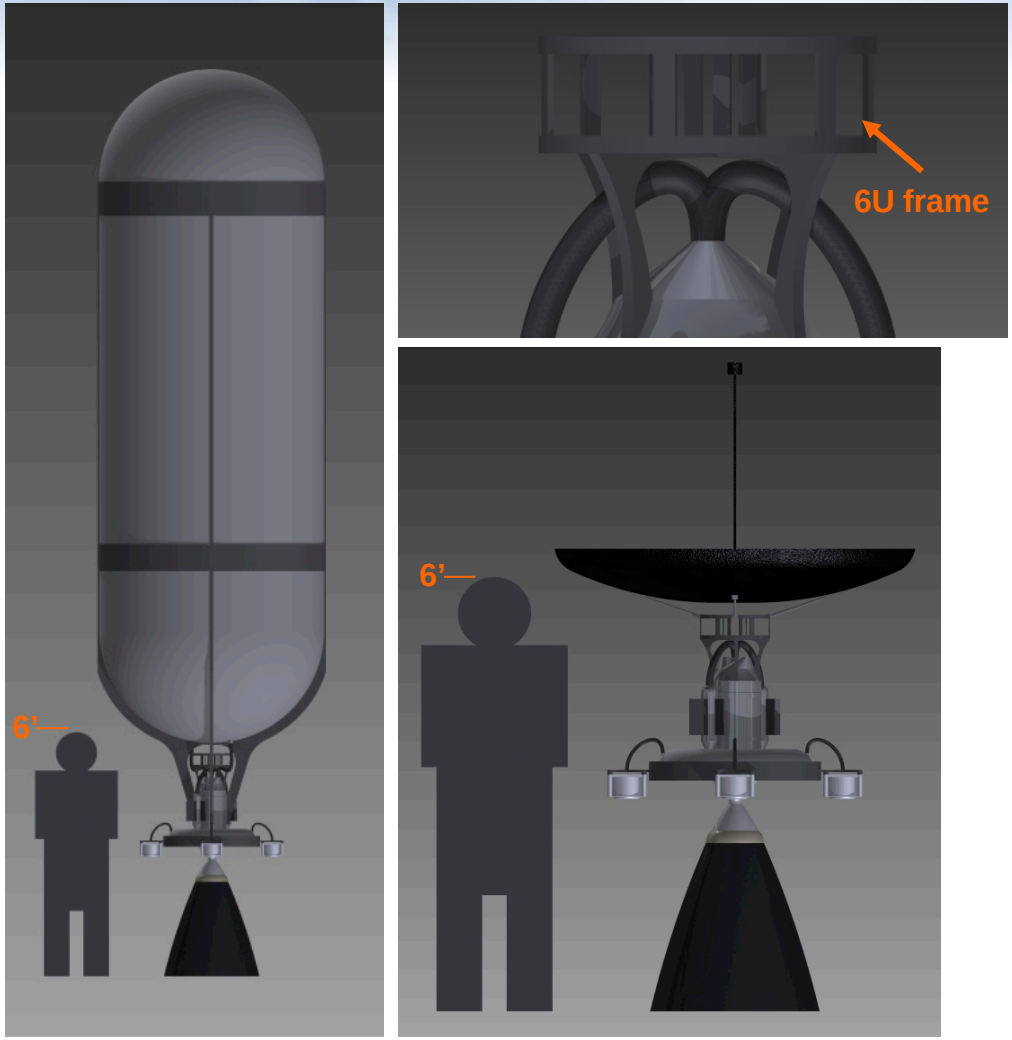
$time_{blowdown}$: 360 s
$time_{heatup}$: 0.2 day
$time_{escape}$: 80 days
ΔV_{escape}: 3.8 km/s
temp: 1700 K

Core

30 cm length
20 cm diameter
2 mm dia. flow channels
18.1 kg Silicon
7.16 kg $^{238}PuO_2$
89.8 kW_t core power
27 kW_e electric power
31.1 kg TOTAL

Performance

propellant: Hydrogen
Isp : 694 s
blowdowns: 400
$\Delta V_{per\ burn}$: 0.0095 km/s
$thrust_{per\ burn}$: 26.39 N
prop. $mass_{per\ burn}$: 1.39 kg
prop. mass: ~558 kg TOTAL



artistic rendering of proposed concept

Center for Space Nuclear Research

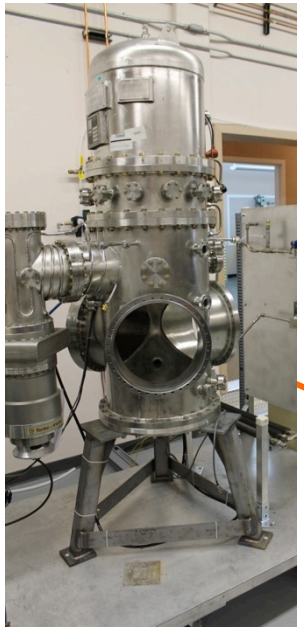
Future Work

Phase I

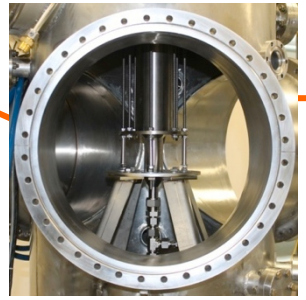
- Further Optimization
  - *concept design*
  - *mission design*
- Simulations
  - *thermal hydraulics*
  - *trajectory optimization*
- Technology Comparison
  - *chemical propulsion*
- Mission Comparison
  - *Enceladus Orbiter via Decadal Survey, etc.*

Phase II

- Detailed Concept Optimization
- Technology Demonstration
  - *energy storage within thermal capacitor*
  - *energy conversion with closed-loop Brayton*
  - *thrust demonstration with propellant & nozzle*

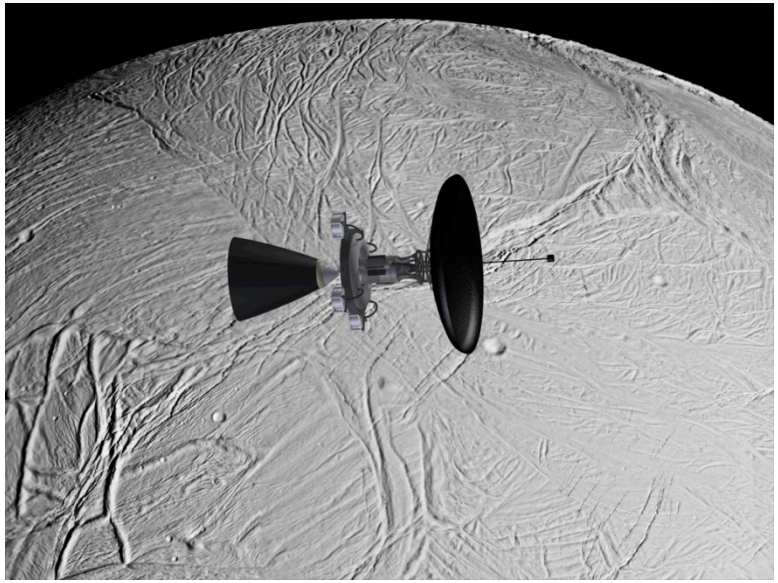


available experimentation hardware



Summary

- A reliable interplanetary propulsion system based on radioisotopic energy is possible
  - *provides versatility to use available power for propulsion or electrical power production*
- Extends the realm of CubeSat-based exploration and experimentation
- Enables a broader range of researchers and research institutions



artistic rendering of concept in Enceladus orbit

- The development of a low mass, low cost propulsion system is achievable
  - *smaller launch vehicles → cheaper launch costs*

Enables 'Public Access' To Outer Planet Exploration!!

Acknowledgments

- Funding Provided By
  - *NIAC Phase I Award → Fiscal Year 2014*

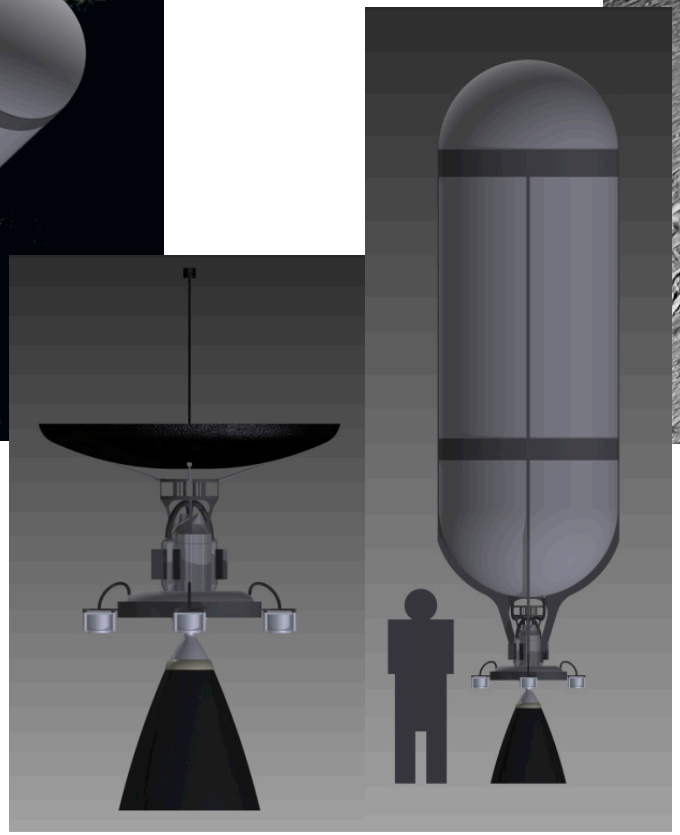
- Co-Investigators
  - *Troy Howe & Adarsh Rajguru*
  - *Steven Howe → Advisor*

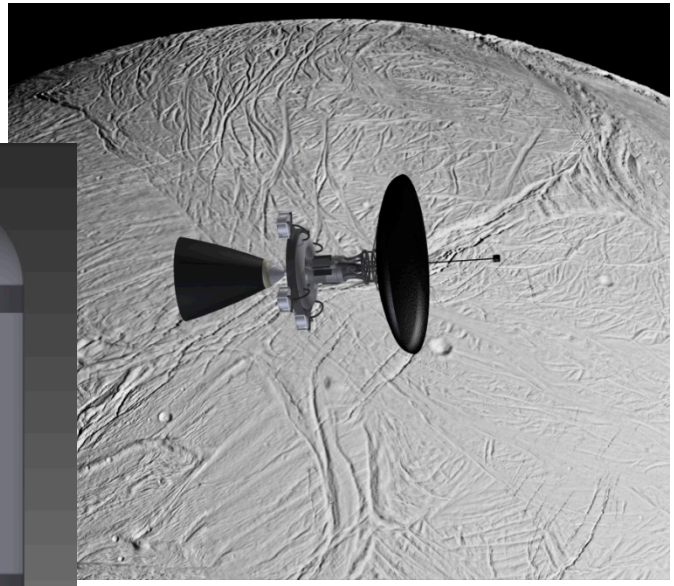
NIAC Group → Thank you for program support

artistic rendering of proposed concept

THANK YOU!!

Questions??





References

O'Brien R. C., Ambrosi R. M., Bannister, N. P., et al., Spark Plasma Sintering of simulated radioisotope materials within tungsten cermets, Journal of Nuclear Materials, 2009, 393(1), 108-113.

O'Brien R. C., Ambrosi R. M., Bannister, N. P., et al., Safe radioisotope thermoelectric generators and heat sources for space applications, Journal of Nuclear Materials, 2008, 377(3), 506-521.

Mattarolo, G., "Development and Modeling of a Thermophotovoltaic System," Thesis, Electrical Engineering and Computer Science Dept., University of Kassel. Kassel, Germany (2007).

Carl M. Stoots. "Emissivity Tuned Emitter For RTPV Power Sources." *Nuclear and Emerging Technologies For Space,The Woodlands, TX,03/21/2012,03/23/2012*. (2012).

Larson W. J., Wertz J.R., Space Mission Analysis and Design, 3rd Edition, Microcosm Press, 2005.

Curtis H. D., Orbital Mechanics for Engineering Students, Elsevier Aerospace Engineering Series, 2005, pg 268-273.

Gaskell, D. R. *Intro. to the Thermodynamics of Materials, 4th Ed.* (2003) 587.

Kelley, K. K. "The Specific Heats at Low Temperatures Of Crystalline Boric Oxide, Boron Carbide And Silicon Carbide". *Journal of the American Chemical Society*. 63 (1941) 1137-9.

Kantor, K., P. B. Krasovitskaya, R. M. Kisil, O. M. Fiz. "Determining The Enthalpy And Specific Heat Of Beryllium In The Range 600-2200" *Phys. Metals and Metallog*. 10 (6) (1960) 42-4. Mcl-905/1, Ad-261792.

Booker, J. Paine, R. M. Stonehouse, A. J. Wright. "Investigation Of Intermetallic Compounds For Very High Temperature Applications". Air Development Division (1961) 1-133. Wadd Tr 60-889, Ad 265625.

Pankratz, L. B. K. K. Kelley. *Thermodynamic Data for Magnesium Oxide* U S Bur Mines. Report. 1-5 (1963); Bm-Ri-6295.

Kandyba, K., V. V. Kantor, P. B. Krasovitskaya, R. M. Fomichev, E. N. Dokl "Determination Of Enthalpy And Thermal Capacity Of Beryllium Oxide In The Temperature Range From 1200 – 2820" *Aec-Tr-4310*. (1960) 1-4.

Hedge, J. C., J. W. Kopec, C. Kostenko, J. I. Lang. *Thermal Properties Of Refractory Alloys*. Aeronautical Systems Division. (1963) 1-128; (Asd-Tdr-63-597, Ad 424375)

X-123CdTe (X-Ray & Gamma-Ray Detector System) – http://www.amptek.com/x123cdte.html

Argus Infrared Spectrometer – http://www.thoth.ca/spectrometers.htm

NanoCam C1U (High Resolution Camera) – http://gomspace.com/index.php?p=products-c1u

Low Voltage Gated Electrostatic Mass Spectrometer (LVGEMS) – http://www.techbriefs.com/component/content/article/16137

http://www.universetoday.com/106288/indias-mars-orbiter-mission-mom-requires-extra-thruster-firing-after-premature-engine-shutdown/

http://trajbrowser.arc.nasa.gov/

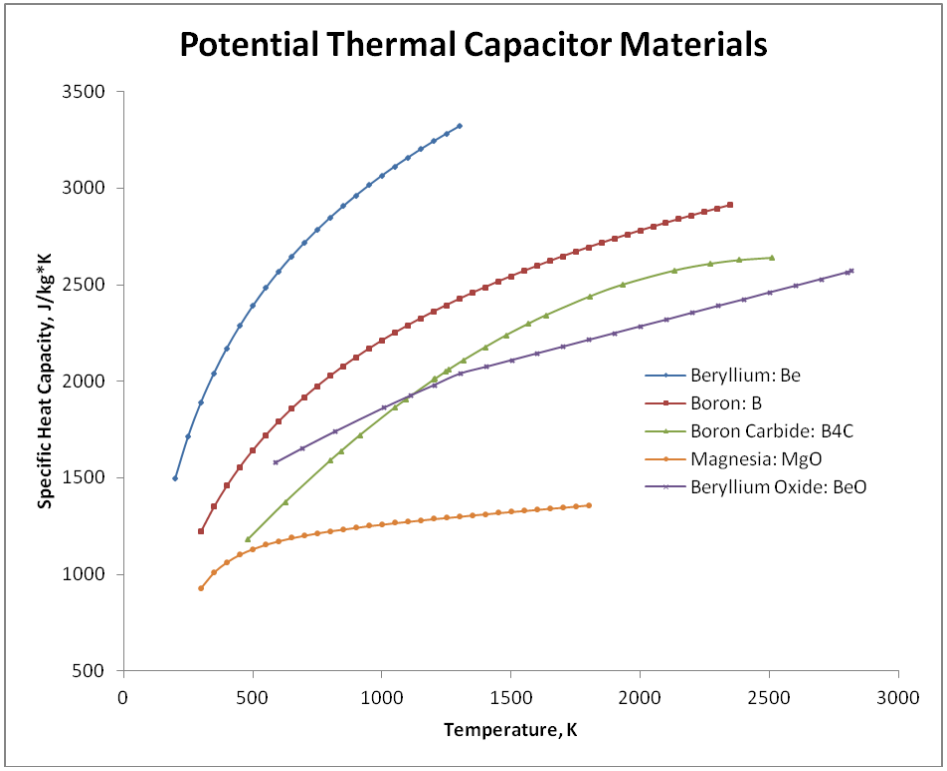
Appendix

Concept
core material

Thermal Capacitor Qualifications

– *high thermal storage, high thermal conductivity, high density, high melting temperature*

Specific Heat verse temperature of possible core materials



Silicon identified as a suitable material

Energy Storage Potential:
ΔH_{fusion} *=1.8 MJ/kg (1700 K)*

Allows for an operational temperature of 1700 K

Mission Design
communication

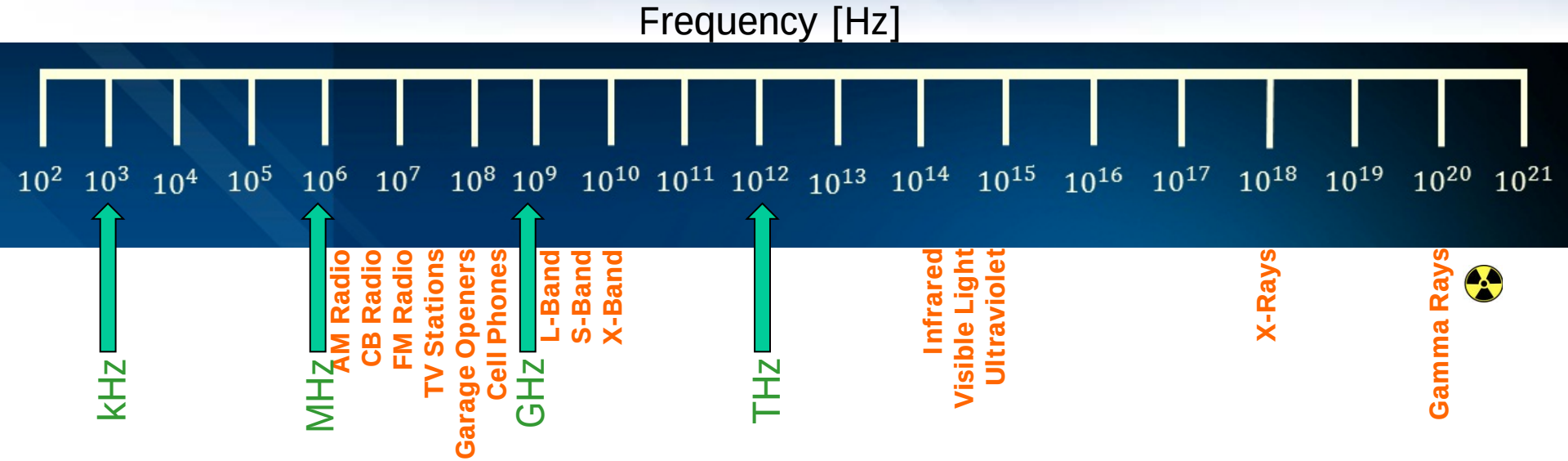
link budget between spacecraft and DSN

Item	Symbol	Units	Value
Frequency	f	GHz	27.5
Transmitter Power	Pt	Watts	2500
Transmitter Line Loss	Lt	dB	-1.10
Transmit Antenna Diameter	Dt	m	1.5
Transmit Antenna Gain (net)	Gt	dBi	50.16
Propagation Path Length	S	km	$1.27(10)^9$
Space Loss	Ls	dB	-303.30
Propagation & Polarization Loss	La	dB	-0.06
Receive Antenna Diameter	Dr	m	34.00
Receive Antenna Pointing Loss	Lpr	dB	-6.87
Receive Antenna Gain	Gr	dBi	77.22
System Noise Temperature	Ts	K	84.10
Data Rate	R	Mbps	**1**
SNR	Eb/No	-	5.94

table describing communication budget

**designed for 9.47 AU distance*

Communication
frequency bands



Frequency Band	Frequency Range	
	Uplink	Downlink
	GHz	GHz
UHF	0.2 - 0.45	0.2 - 0.45
L	1.635 - 1.66	1.535 - 1.56
S	2.65 - 2.69	2.5 - 2.54
C	5.9 - 6.4	3.7 - 4.2
X	7.9 - 8.4	7.25 - 7.75
Ku	14.0 - 14.5	12.5 - 12.75
Ka	27.5 - 31	17.7 - 19.7
SHF / EHF	43.5 - 45.5	19.7 - 20.7

Spacecraft→
DSN HEA

Communication
link budget design

Idaho National Laboratory

Net Gain of Transmission Antenna

Receiver Losses

Net Gain of Receiver Antenna

Transmission Power (W)

Transmission Losses

Atmospheric Losses

$$\frac{E_b}{N_0} = \frac{P_t L_t L_r G_t G_r L_a L_s}{k T_s R}$$

Space Loss

Data rate (bps)

Signal to Noise Ratio

System Noise Temperature (K)

Boltzmann Constant (J/K)

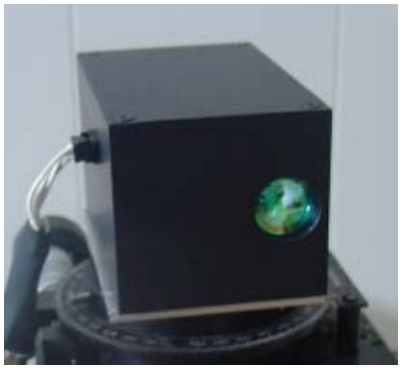
Center for Space Nuclear Research

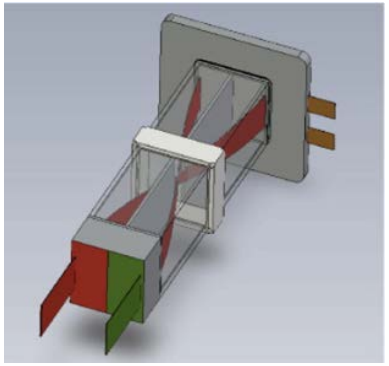
Mission Design
instrument package

proposed instrument package for Enceladus

Instrument	TRL	Mass (kg)	Volume (cm^3)	Peak Loading Power (W)
X-123CdTe (X-Ray & Gamma-Ray Detector System)	7	0.18	175	2.5
Argus Infrared Spectrometer	9	0.23	180	-
NanoCam C1U (High Resolution Camera)	8	0.166	501	0.66
Low Voltage Gated Electrostatic Mass Spectrometer (LVGEMS)	7	0.25	32	0.5

table describing possible instrument package

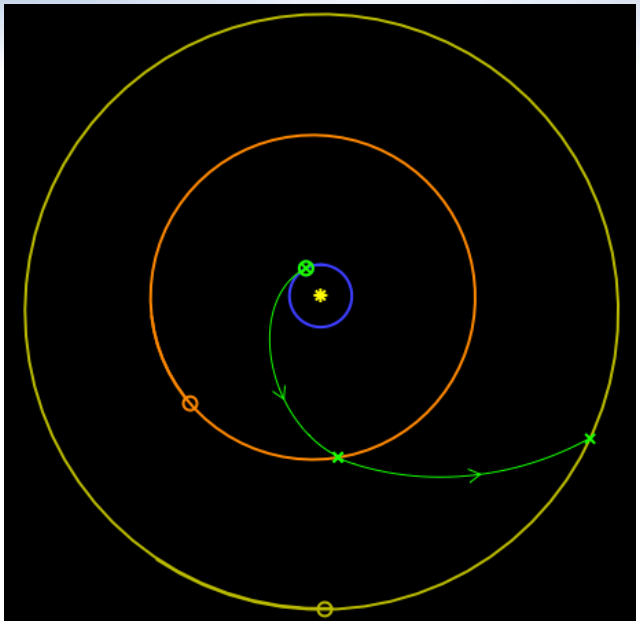




*designed for about a 6U CubeSat payload

Trajectory

- Electric Propulsion
  - *4 ion thrusters → redundancy*
    - *Xenon propellant*
  - *allows for shorter transit times*



possible heliocentric trajectory to Saturn

Earth – Jupiter - Saturn

- **Earth Departure: Jan-18-2018**
- **Saturn Arrival: July-11-2023**
- **Total Duration: 5.48 years**
- **Launch Vehicle Injection C3 required: 80.7 km^2/s^2**
- **Declination of the launching asymptote: 4 degrees.**
- **Injection delta V(Earth escape delta V requirement):6.42 km/s**
- **Jupiter Flyby: Jan-08-2020**
- **Gravity Assist: 123 m/s (0.12 km/s)**
- **Post injection delta V (Saturn capture): 0.54 km/s**
- **Total delta V required by propulsion system = 6.96 km/s**