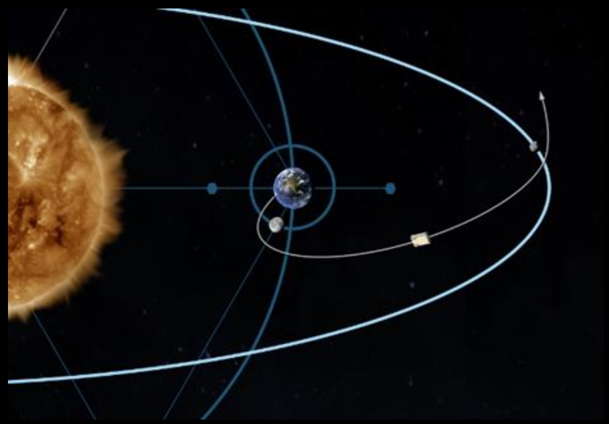
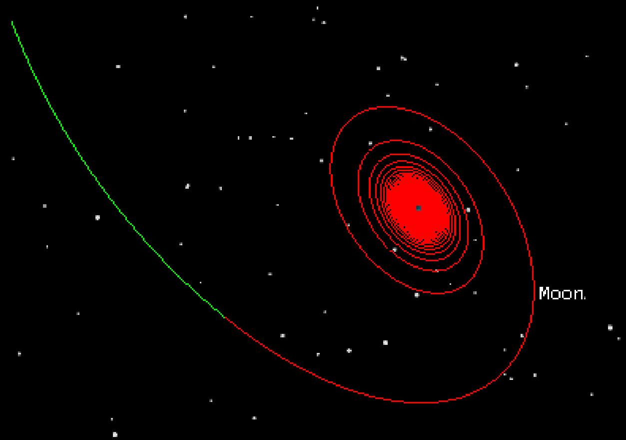
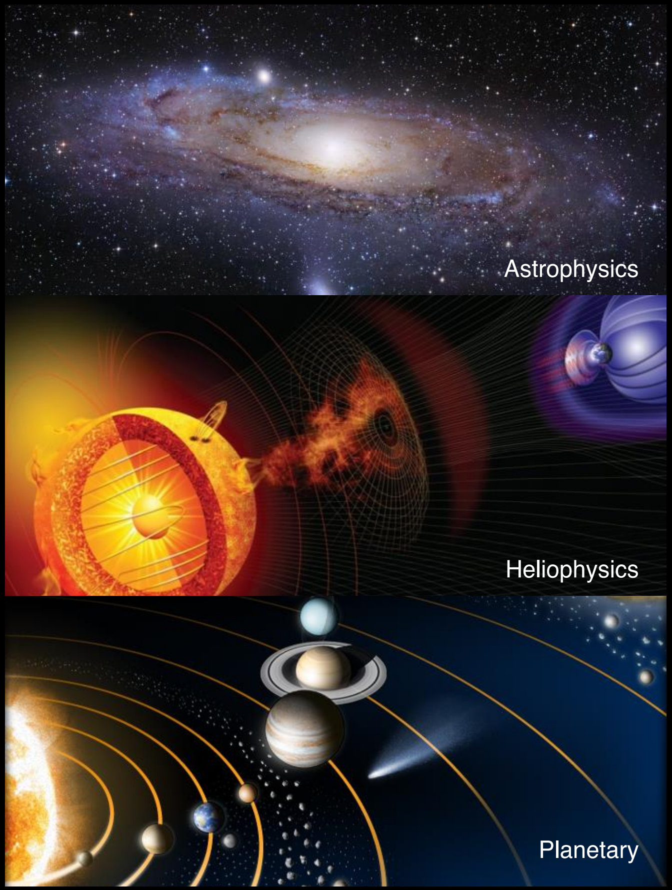
Design Optimization of Small Spacecraft with Electric Propulsion Systems

ISSC 2015







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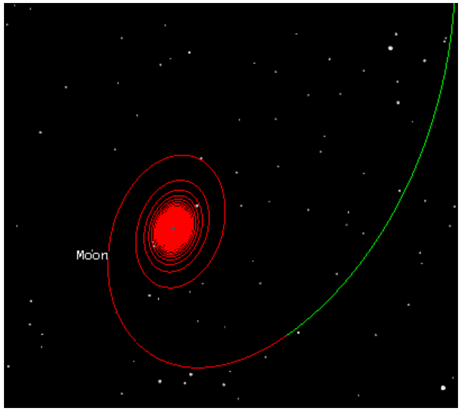
Ben Longmier, University of Michigan; Derek Dalle, NASA/Ames

Interplanetary Small Satellite Conference, April 2015, Santa Clara, CA

Past, Present & Future of CubeSat Propulsion Systems



Past: Low Earth Orbit (LEO) CubeSats “passive drifters”

Present: Current State of the Art

- Cold gas systems for small $\Delta V<100$ m/s, de-sats
- Large electric propulsion (EP) systems ~10 kg

Future: Several emerging EP solutions for CubeSats

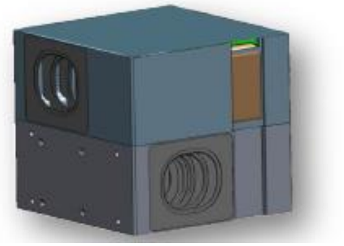
Game-changing and enabling/enhancing a broad class of missions:

- Significant ΔV primary propulsion
  - Change orbit, create constellations, drag makeup in LEO
  - Deorbit CubeSats or other debris in LEO
  - Ability to perform formation flight (large apertures)
  - Large maneuvers to transfer to comets, asteroids, planets!
  - Ability to “capture” or create constellations around bodies
  - Hover, proximity operations, land on small bodies, rings, etc.
- Attitude control maneuvers
  - De-saturate reaction wheels, reaction wheel replacement, etc.

Goal of this talk: Identify niches for electric propulsion small satellite technologies and optimal applications using systems-level perspective

Heritage and Enabling Technology

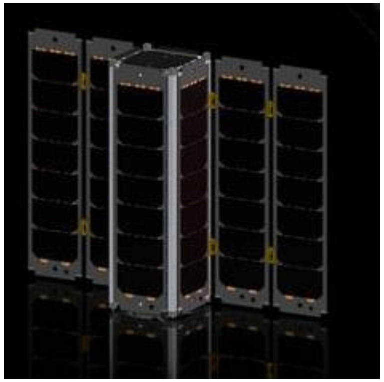
- Significant flight experience and heritage in LEO and high-TRL components
- Telecommunication and Navigation systems
  - High-rate X/Ka-Band radios (10+ Mbps in LEO)
  - Iris Transponder (JPL) and high gain antennas
- High-accuracy attitude control technology
  - Blue Canyon's XACT: 7.2 arcsec accuracy, 1 arcsec stability, <2.5 kg, ~1 U, <2.5 W
  - VACCO Cold Gas Systems (ΔV<80 m/s in 3U CubeSat)
- Solar arrays that are deployed and gimbaled for Sun-tracking
  - Deployable Solar Arrays (eHAWK arrays up to 130 W/kg)
- Integrated Computers, GNC, and Bus Architectures
  - BCT XB1 Bus (GNC, C&DH, Telecom, Power, ACS)
  - Radiation-tolerant flight computers (LEON, etc.)
  - Companies offering buses like Tyvak, Blue Canyon, etc.
- Aluminum 3U CubeSat Structure (radiation shielding)



XB1 Blue Canyon System

eHAWK MMA Solar Arrays (130 W/kg)

Clyde Space Double Deployed 2-Sided 30 W Solar Panels



ISIS 3U CubeSat Al Structure

Image Credit: Clyde Space, ISIS, Blue Canyon, MMA

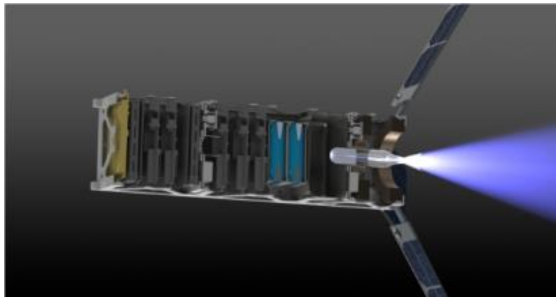
Overview of Emerging Small Spacecraft EP Systems

Thruster* (Point Design)	Power	Thrust	I_{sp}	Mass
Units	W	mN	sec	kg
CAT Plasma	100	10	1010	0.5
Busek's BHT-200	275	13	1375	10
Busek's MEP (HARPS)	2	0.1	1500	0.1
Mini Helicon Plasma	5	0.185	2000	0.1
MIT iEPS	40	2.28	2000	0.1
Busek's Ion (BIT-1)	10	0.1	2150	0.05
MIT MEMs Ion MEP	10	0.1	3000	0.16
MiXI Ion	40	1.43	3000	0.25
Busek's Ion (BIT-3)	60	1.4	3500	0.2
JPL's MEP	8.16	0.174	3744	0.16

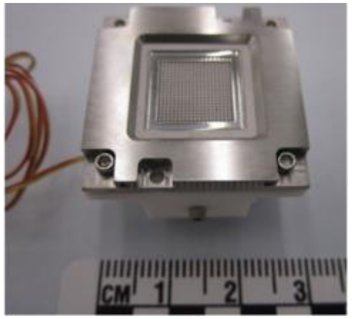
*Thruster specs based on publically available information

Notes:

- Busek BHT-200 has high TRL (9 in LEO)
- Most other EP thrusters have TRL < 6



UMich/Aether's CubeSat Ambipolar Thruster (CAT)

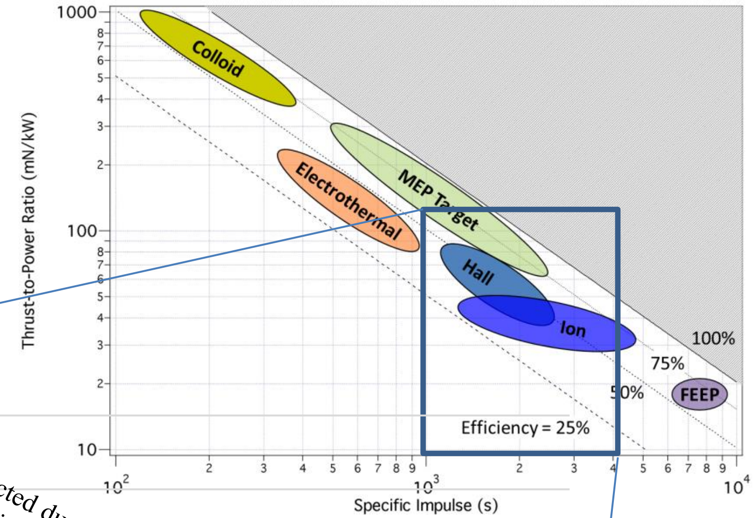


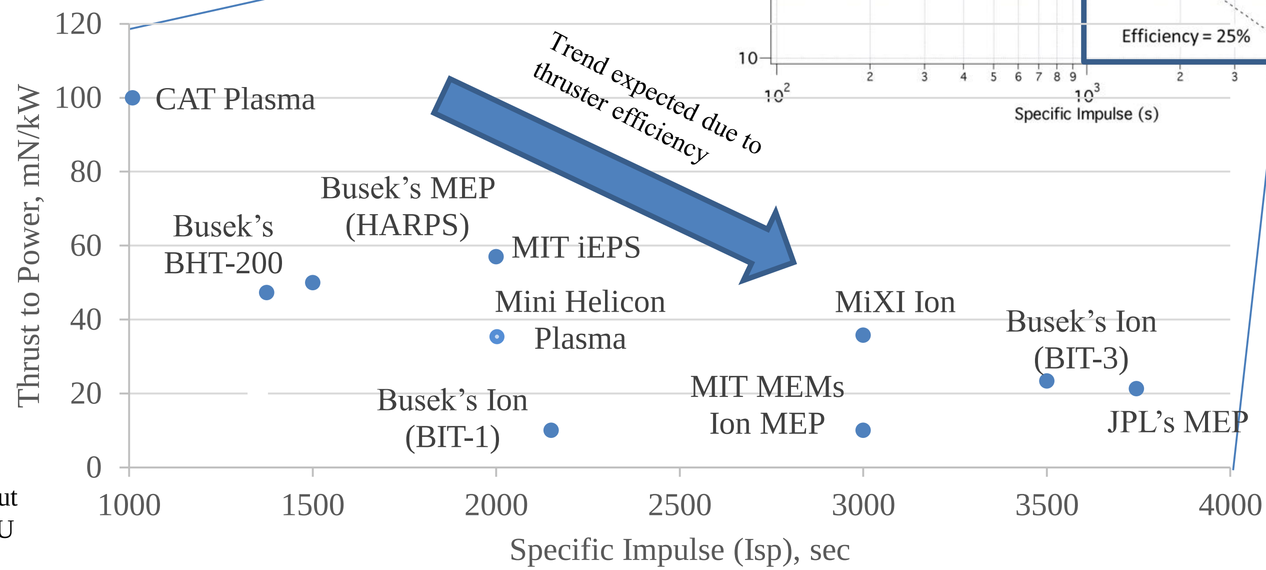
JPL's Indium MEP Thruster

Where do Emerging EP Systems “Fit”?

Small Electric Propulsion (EP) technologies:

- Aim to provide high thrust, maximize thrust to power ratio ΔV
- Provide long duration thrusting
- Satisfy small spacecraft mass, volume, power, thermal constraints



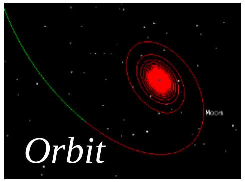


Power is input power to PPU

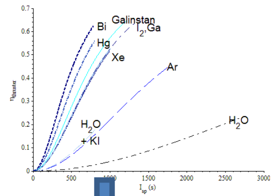
- CAT Plasma: low I_{sp} maximizes Thrust to Power; MEP mid-range I_{sp}
- Most propulsion systems actually span a “range” but point design plotted

Reference: “Review of MEP Technology”, Marrese-Reading, John Zimer, et a.., MEP A-Team Study, Sept. 17, 2014

Multidisciplinary Systems Modeling Approach



Propulsion System Model (thrust, I_{sp}, mass)



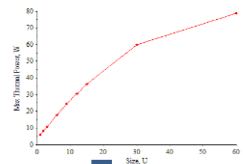
Assumptions:

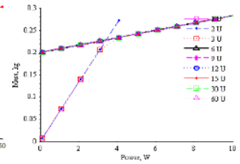
- Thrusters fire perfectly in desired direction.
- Spacecraft mass includes propulsion system (propellant, etc.), bus, and solar panels.
- Mass margin includes payload and PPU mass.
- Thrusters are modular and can be fractional
- Solar panels sized for continual thrusting.

Trajectory (ΔV, time, thrust)

Compute Power, # Thrusters, Propellant Mass

Thermal and Power Model





Size Solar Arrays

Check Feasibility of Thermal and Power Systems

$$M_{sc}=M_{prop}+M_{bus}+M_{sp}$$

Masses:

M_{sc} Spacecraft :

M_{prop} Propulsion System

M_{bus} Bus

M_{sp} Solar Panels

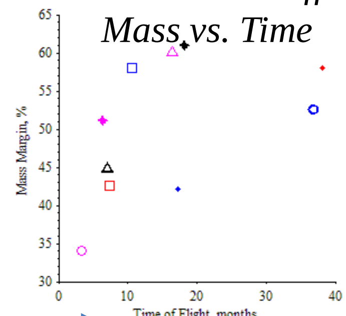
M_{max} Maximum mass for given size (~2 kg/U)

Compute Total System Mass (thrusters, propellant, power system)

Mass Margin = $(M_{max}-M_{sc})/M_{max}$

Check Total Mass Feasibility and Compute Margins

Pareto Trade-offs Mass vs. Time

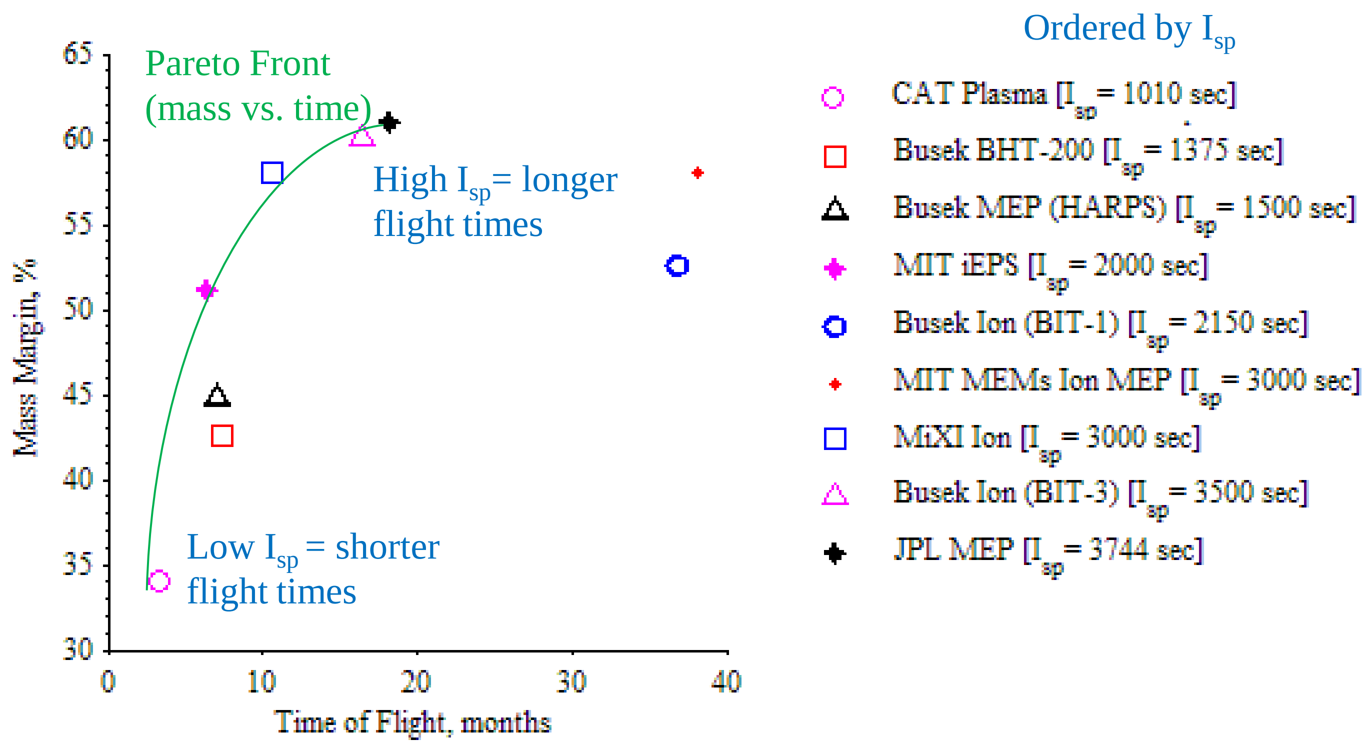


Thrust, I_{sp}, propellant

Trajectories from: S. Spangelo, D. Landau, N. Aurora, S. Johnson, T. Randolph, “Defining the Optimal Requirements for the Micro Electric Propulsion Systems for Small Spacecraft Applications", Journal of Spacecraft and Rockets, Under Review.

Comparison of EP Thrusters for ΔV=1 km/sec Maneuver

6 U (12 kg, max ~56 W) CubeSat with continuous thrusting

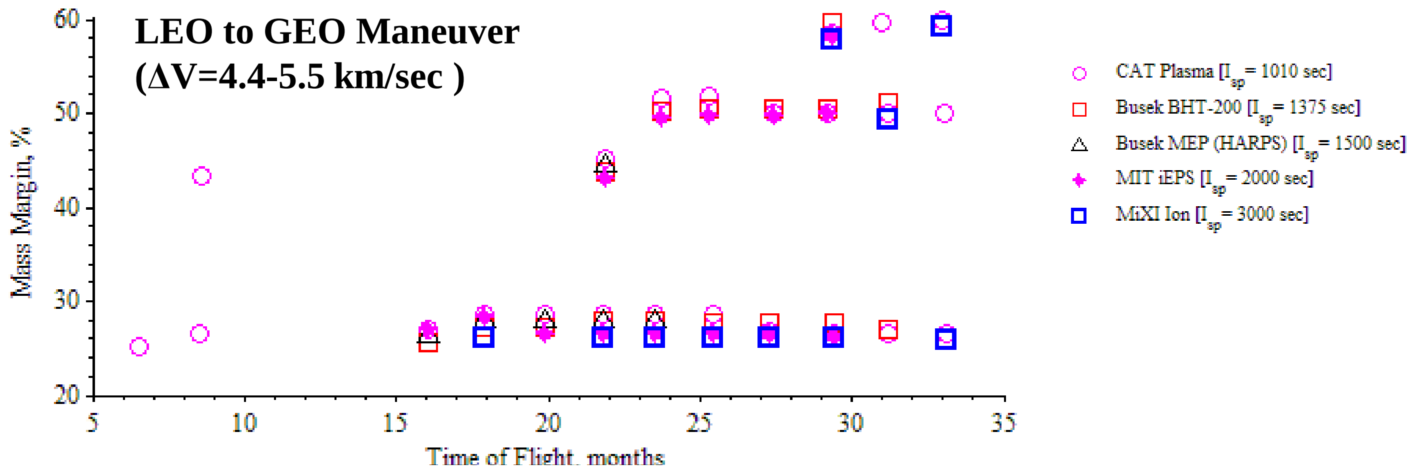


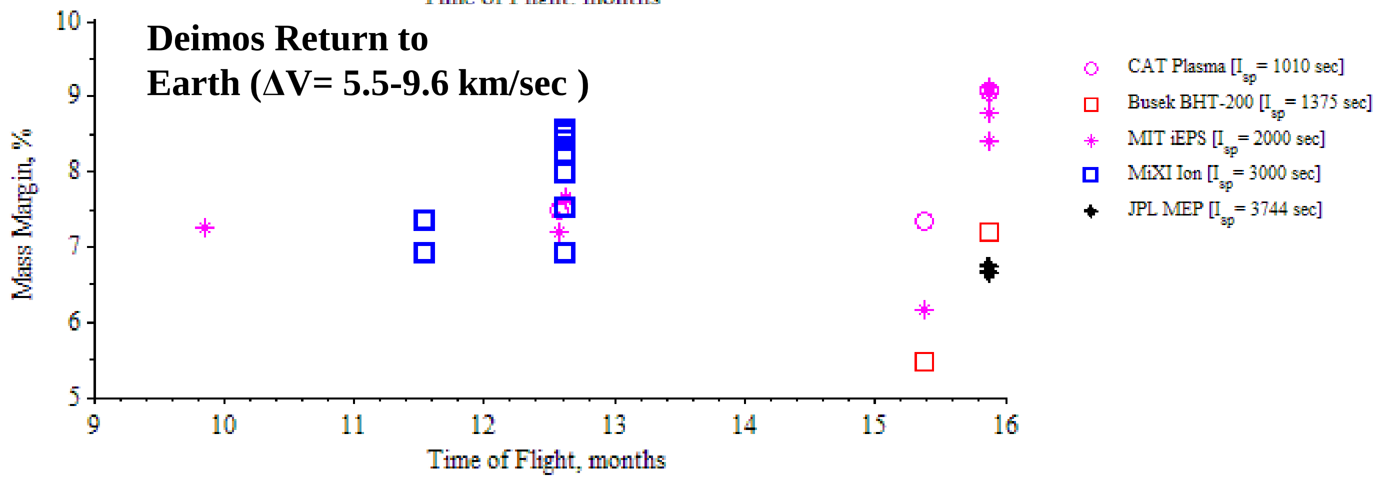
- *CAT Plasma minimizes flight time, JPL MEP maximizes mass margin*
- *Pareto front dominated by highest thrust-to-power thrusters for a given I_{sp}*

Note: Results are for the published "point design" and each thruster will operate across a range of values.

Comparison of EP Thrusters for Large Orbit Transfers

12 U (max~ 24 kg, max ~102 W) CubeSat with optimal thrusting for data sets with diverse constrained I_{sp} values (times)

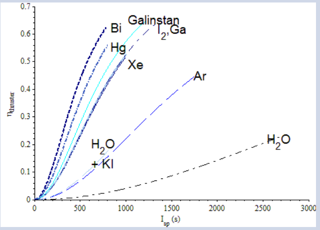




Best results for CAT, MIT iEPS, MiXI which have low I_{sp} and on Pareto front

Note: Results are for the published "point design" and each thruster will operate across a range of values.

Optimization Formulation for Earth-Escape CubeSats

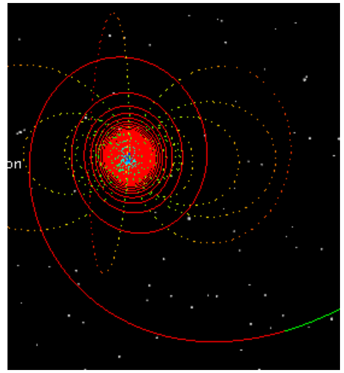
Element	Optimization Parameters
Objectives	Minimize time, propellant, or radiation exposure
Decisions	• Thrusting strategy (where/when, level) • Vehicle design (solar power, batteries, size)
Constraints	• Starting orbit: 500 km circular LEO polar • Final orbit: escape Earth orbit (SOI: 925,000 km) • Maintain positive energy balance and survive eclipses • CubeSat form-factor (3-6U, <6-12 kg)
Assumptions	• Bus power consumption <5 W, Max power collection: 25 W • Applicable to any small spacecraft EP technologies
Dynamics (MBSE Framework)	*Orbit* $\dot{r} = 2a\sqrt{\frac{r^3}{\mu}},$ $a = -\frac{\dot{m}V_{ex}}{m},$ $V_{ex} = gI_{sp},$ *Propulsion*  *Energy* $P_{s,i}t_{s,i} \geq (P_{t,i} + P_{n,i})t_{t,i},$ $E_{batt} \geq (P_{t,i} + P_{n,i})t_{e,i},$

S. Spangelo and B. Longmier, "Optimization of CubeSat System-level Design and Propulsion Systems for Earth-Escape Missions", Journal of Spacecraft and Rockets, accepted December 2014.

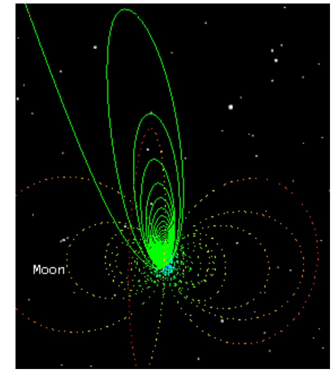
CAT: Optimal Solutions for Different Objectives

3U Earth-escape trajectory starting in 500 km circular orbit with deployable arrays

Case 1: Constant Thrust in Velocity Direction



Case 2: Optimized Variable Thrust/Time at Perigee



Comparison of solutions for various goals

Optimization Goal	Initial Orbit Sun Synchronous (P_{av}: 25 W)	Initial Orbit Not Sun Synchronous (P_{av}: 11 W)
Minimize Time	Case 1- 108 days	Case 2- 175 days
Minimize Propellant	Case 2- 1.34 kg	
Minimize Propellant & Battery Mass/ Volume	Case 1- 2.5 kg/ 0.5 U	
Minimize Radiation (~6mm Al assumed)	Case 2- 1.03 krad (Case 1: 1.1-3.9 krad)	

S. Spangelo and B. Longmier, "Optimization of CubeSat System-level Design and Propulsion Systems for Earth-Escape Missions", Journal of Spacecraft and Rockets, Accepted 2014.

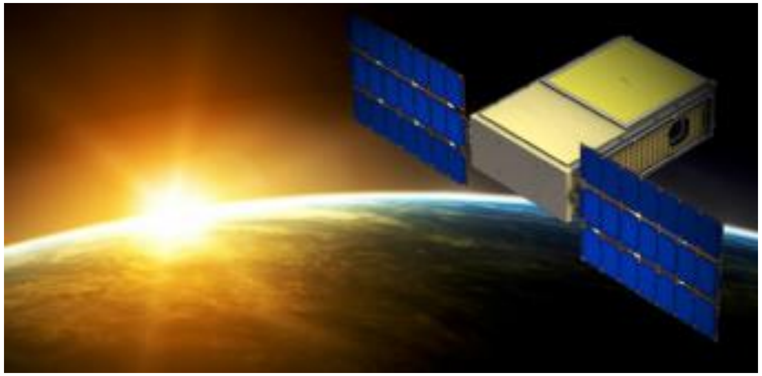
Summary & Future Work

Summary

- Framework for evaluating diverse emerging thruster technologies
- Pareto fronts for ideal thruster for different objectives
- Trade-offs between mass, time, radiation metrics for diverse thrust strategies
- Best performance for high thrust-to-power CAT, MIT iEPS, JPL MEP, MiXI

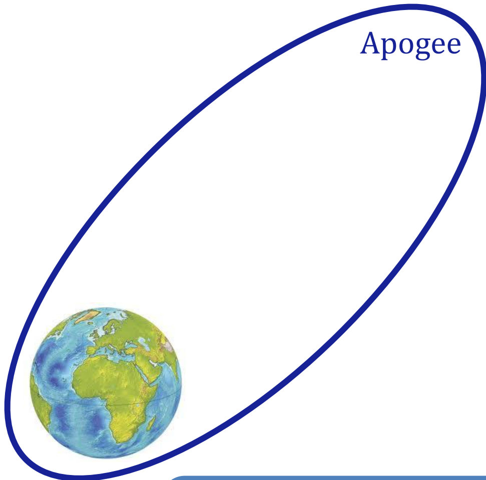
Future Work

- Model and simulate radiation, and attitude control in optimization problem
- Model realistic operations (thrust strategy, radiation, lifetime, etc.)
- Consider higher-fidelity orbit transfer models and lifetimes issues
- Comparison to solar sail technologies, chemical systems, etc.

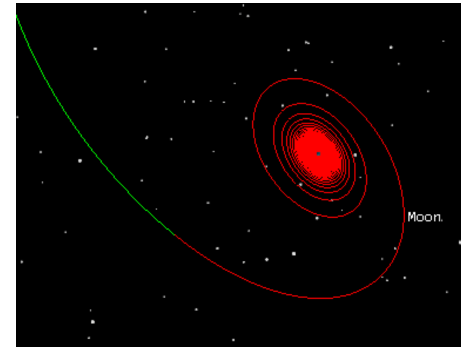


Back-up Slides

Multidisciplinary System Level Constraints and Interactions

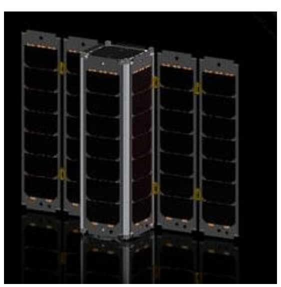


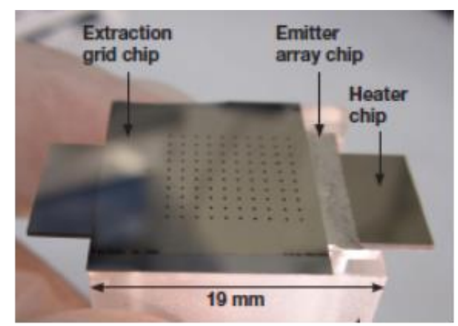
Trajectory Design
- I_{sp}, Thrust
- ΔV to achieve orbits
- Thrust time, trip time



Mass, Power , Thermal
- Battery Capacity
- Solar Power Collection
- Thermal Radiation

Propulsion
- Number of thrusters to achieve required thrust
- Propellant mass

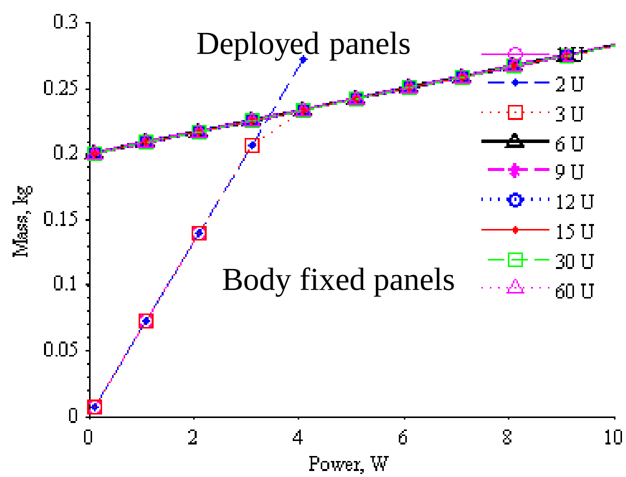


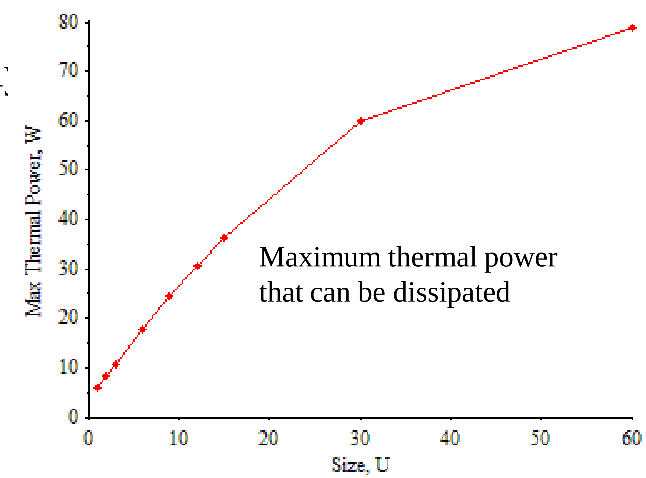


JPL's Indium MEP thruster

Modeling Assumptions

- Systems-level integrated models (trajectories, spacecraft, propulsion)
- Approach generally applicable to all MEP technologies
- The thrusters generate thrust perfectly in the desired direction.
- Attitude control is accomplished by on-board reaction wheels in the case of primary propulsion and by the thrusters when they perform attitude control.
- The PPU, heater, and neutralizer are sized to accommodate the thruster.
- There are no solar eclipses or occultations in the trajectories.
- The solar panels are sized to support continuous thrusting and nominal bus.
- The spacecraft volume and mass are constrained by conventional CubeSat form-factors for small spacecraft and extrapolated for larger ones.
- We investigate study 6-12 U (12-24 kg) spacecraft sizes
- The payload system power is ignored, although this is expected to be significantly less than the thruster power.
- When multiple thrusters are operated simultaneously they each have the same per
- Wh … sivel the … ical.





Model- MEP Propulsion System

- Micro Electro Spray (MEP) technology
- Liquid metal propellant micro-fabricated with Indium propellant
- Capillary-force driven propellant management system with no pressurization, valves, or moving parts
- Small, compact, scalable technology pushing limits of microfabrication technique

$$\dot{m} = I_e/(Q/M), \qquad (3)$$

$$V_{ex} = I_{sp} g, \qquad (4)$$

$$V_b = \frac{V_{ex}^2}{2\eta_b (Q/M)}, \qquad (5)$$

$$T = \frac{I_b V_{ex}}{(Q/M)}. \qquad (6)$$

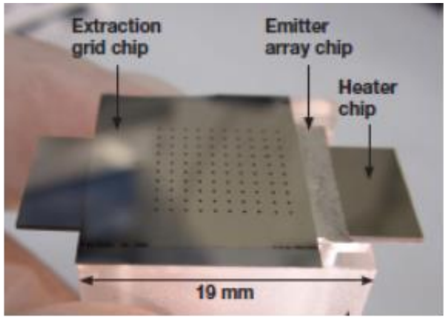
$$I_b = \eta_e n_e I_e, \qquad (7)$$

$$P_b = I_b V_b, \qquad (8)$$

$$P_{PPU} = \frac{P_b}{\eta_{PPU}}, \qquad (9)$$

$$\eta_n = \frac{\eta_b}{V_n}, \qquad (10)$$

$$P_n = \frac{I_b}{\eta_n} + V_n I_b. \qquad (11)$$



JPL's Indium MEP thruster

$$P_p = N_t(1 + m_p)(P_{PPU} + P_n + P_h(s)), \qquad (12)$$

Total Propulsion System Power

$$\eta_S = \frac{N_t T V_{ex}}{2P_p}. \qquad (13)$$

Propulsion System Efficiency

Modeling: Power System Mass

- Power system scales with required power to support propulsion system

Parameters:
- P_s: Average power consumption of the system
- P_{max}: Maximum Power Generated by Fixed (fix) or Deployed (dep) panels
- M_{sp}: Solar Panel Mass for Fixed (fix) or Deployed (dep) panels

$$P_s = (1 + m_s)(P_t d_t + P_b) + P_p d_p,$$

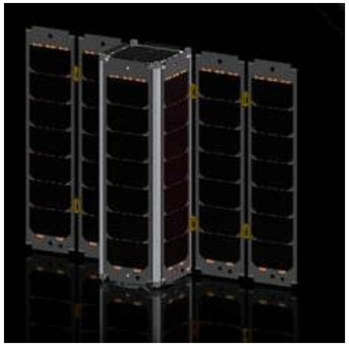
$$P_s \leq P_{max,fix} U,$$

$$M_{sp,fix} = \frac{P_s}{\gamma_{fix}}.$$

$$P_s \leq P_{max,dep} U,$$

$$M_{sp,dep} = m_{sp,gim} + \eta_{sp} \frac{P_s}{\gamma_{dep}},$$

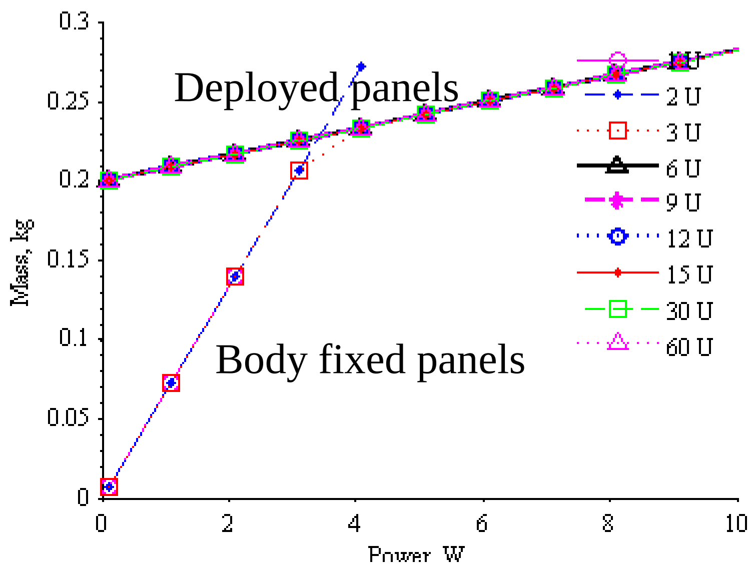
Clyde Space Double Deployed 2-Sided 30 W Solar Panels

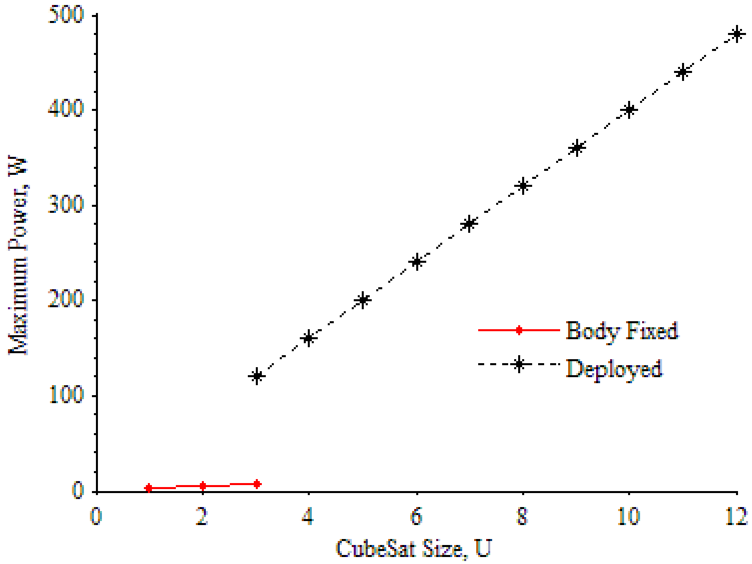


eHAWK MMA Solar Arrays (130 W/kg)

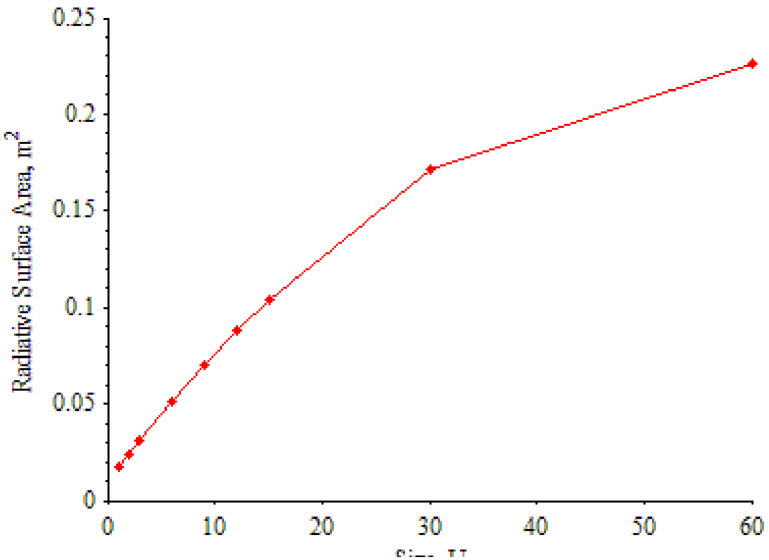
Modeling: Spacecraft Components Scaling

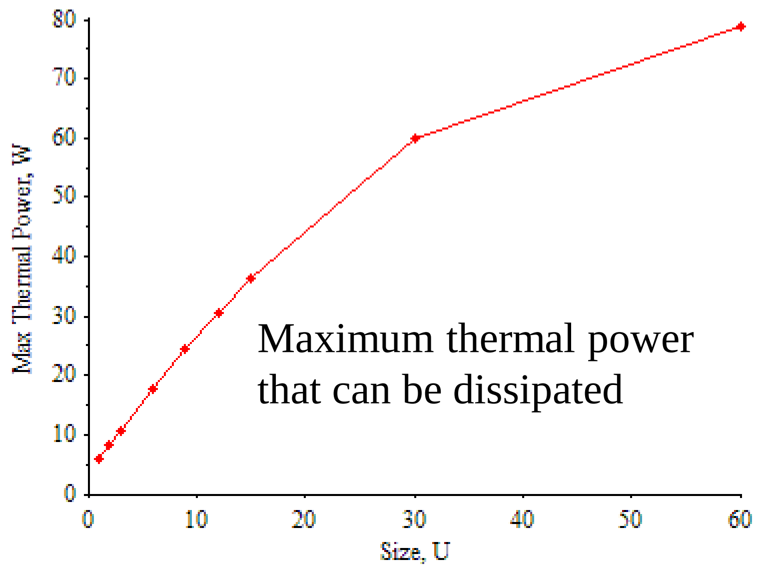
Scaling for Solar Panels





Scaling for Thermal System





Approach- Goals, Decisions, Constraints

Objective: Maximize payload mass fraction $M_p M_i$

- M_p: Payload mass of all components except propulsion system
- M_i: Mass of initial wet spacecraft

Why? To maximize capability to carry science instruments/spacecraft components

Decision Variables:

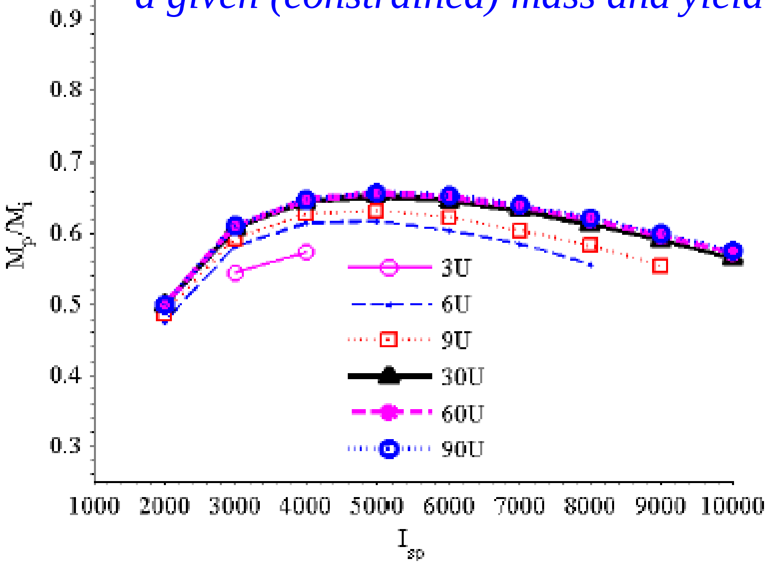
- Number of MEP thrusters (or size)
- Specific Impulse (I_{sp}) which is directly related to thrust level
  - Driving system interface parameter that defines trades between mission design (trip time), power system, and propulsion system (mass, power)
  - Substantial driver in the propulsion system design and defines required operating voltages and total propellant throughput

Constraints:

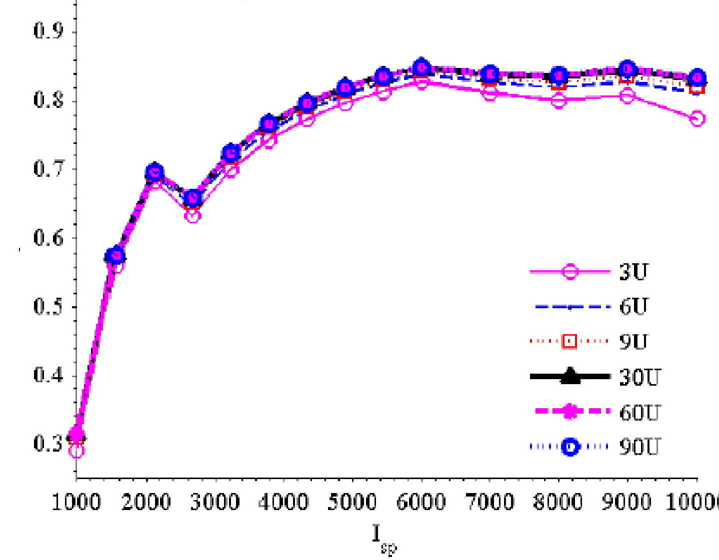
- Due to voltage limitations, $I_{sp} < 7000$ sec; lower voltages preferred to reduce complexity of high-voltage systems
- Short lifetimes preferred; missions with lifetimes less than 3 years

Results- LEO Transfer Cases

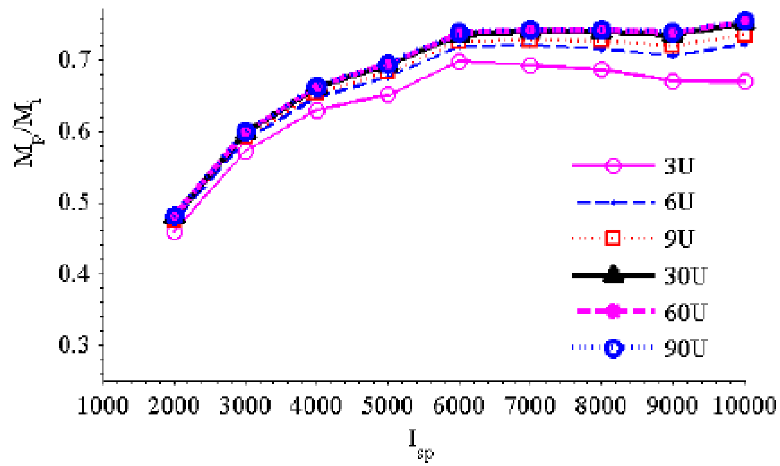
For higher ΔV missions, higher I_{sp} values are optimal to achieve sufficient thrust for a given (constrained) mass and yield higher thrust/trip times.



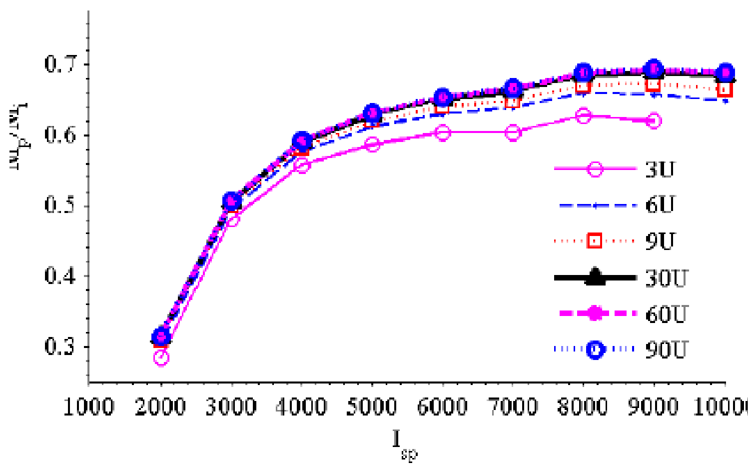
(a) LEO Fly Over ($\Delta V = 0.5$ km/sec)



(d) LEO to GEO ($\Delta V = 4.4 - 5.5$ km/sec)



(e) LEO ISS to Polar ($\Delta V = 6.7 - 7.6$ km/sec)



(f) LEO ISS to Equatorial ($\Delta V = 9.3 - 11.3$ km/sec)

Approach- Mission Data Sets

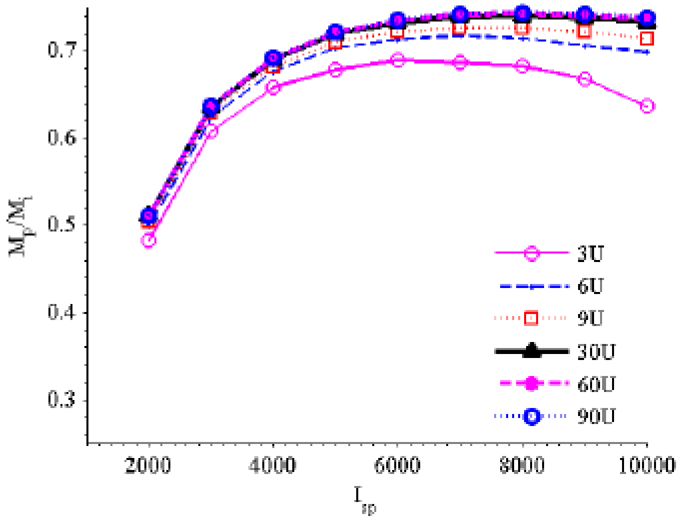
Science-driven broad mission categories to cover upcoming mission opportunities. All categories apply to both LEO and interplanetary applications

#	Category	Propulsion Type	Example Applications
1	Orbit Changing	Primary Propulsion	-Altitude Raising (LEO, GTO, GEO, lunar crossing, other planets) -Inclination Changing (LEO or other) -Interplanetary transfers (Mars sample return)
2	Constant Acceleration for Orbit Maintenance/ Proximity Operations	Primary Propulsion	-Drag Make-up (LEO or other) -Hovering over/ near body

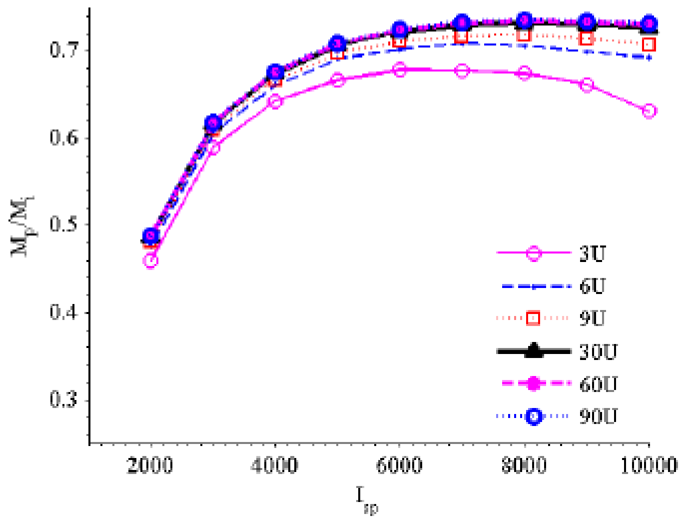
We also studied precision pointing but results are outside scope of this paper.

Results- Interplanetary Transfer Cases

Reducing returns with increase in I_{sp} near the optimal

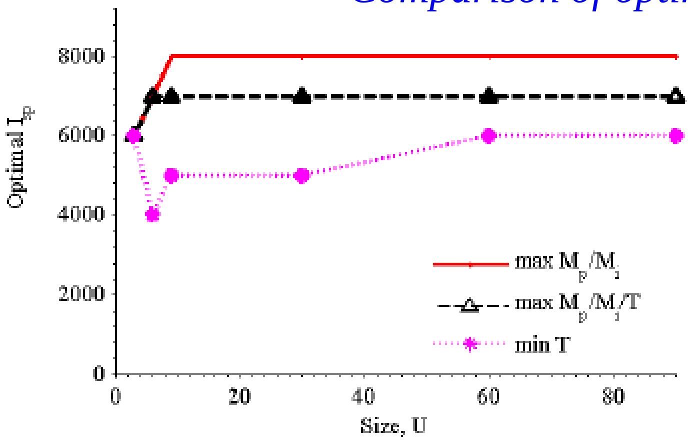


(a) Deimos ($\Delta V = 4.6 - 8.6$ km/sec)

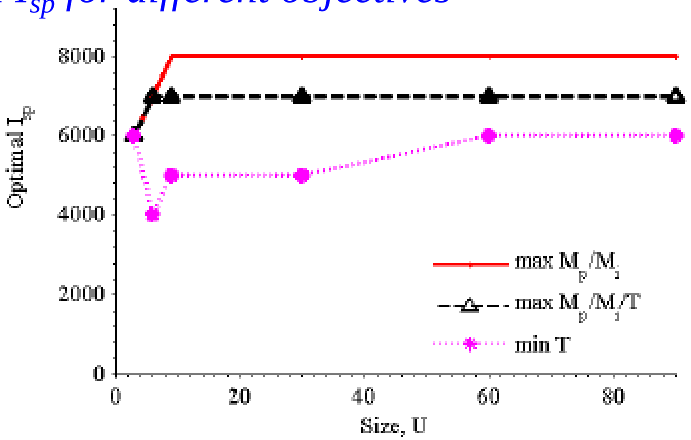


(b) Phobos ($\Delta V = 5.4 - 9.4$ km/sec)

Comparison of optimal I_{sp} for different objectives



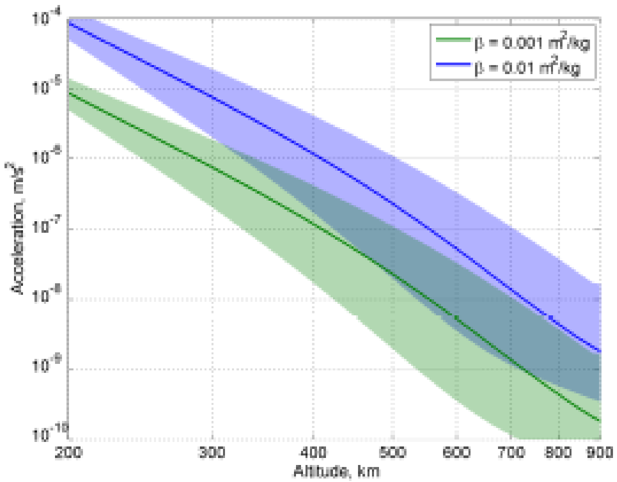
(a) Deimos ($\Delta V = 4.6 - 8.6$ km/sec)



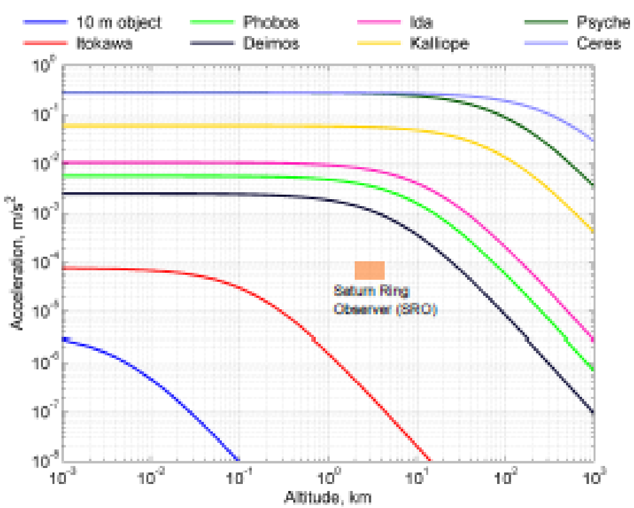
(b) Phobos ($\Delta V = 5.4 - 9.4$ km/sec)

Results- Constant Acceleration Cases

Required Accelerations

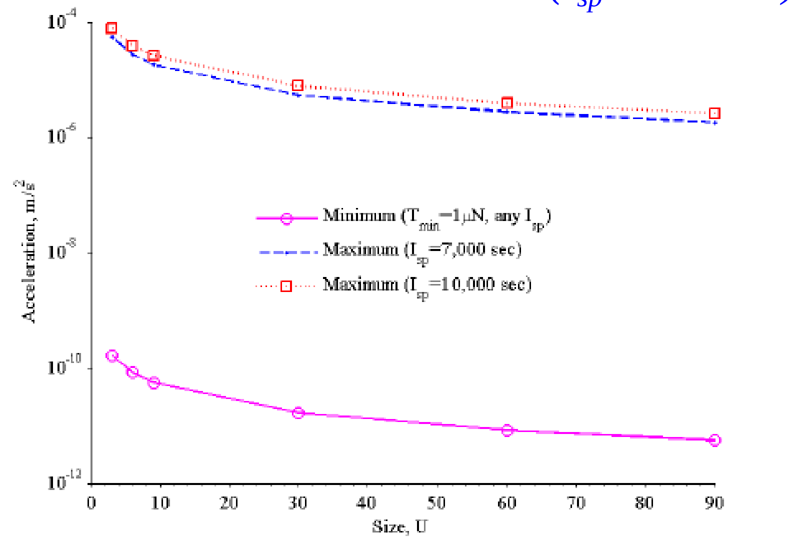


(a) Drag Make-Up

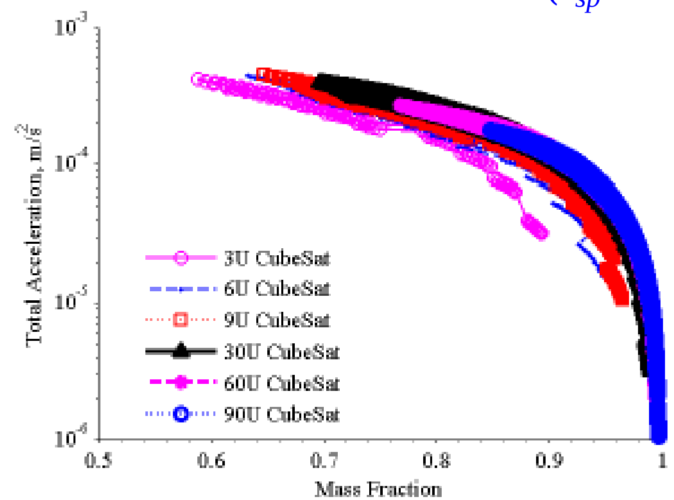


(b) Hovering

Minimum Accelerations (I_{sp}=7000 sec)



Maximum Accelerations (I_{sp}=7000 sec)



Summary & Recommendations

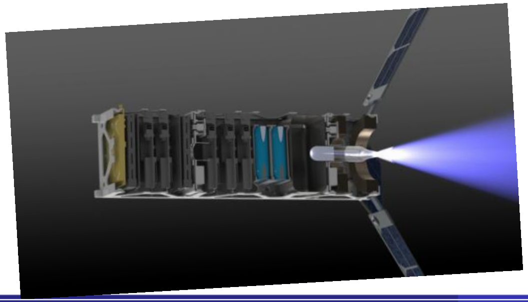
Summary:

- Orbit transfers: I_{sp} of 4000-5000 sec accomplishes ≥89% of optimal
  - Higher I_{sp} (~10000 sec) optimal for high ΔV LEO transfers
  - Higher I_{sp} (~7000 sec) optimal for Mars transfer cases
- Maximum (constant) acceleration is accomplished for I_{sp} ≥3000 sec

Name	Optimal I_{sp} to Maximize M_p/M_i	Percentage of Optimal M_p/M_i (I_{sp}=4000 sec)	Percentage of Optimal M_p/M_i (I_{sp}=5000 sec)
Fly Over	5000	96 %	100 %
GTO to Lunar Flyby	7000	94 %	96 %
LEO to GEO	6000	90 %	95 %
ISS to Polar	10,000	86 %	91 %
ISS to Equatorial	9000	85 %	89 %
Deimos Return	7000	92 %	96 %
Phobos Return	7000	92 %	95 %

Recommendation:

- Minimize thruster/PPU architecture with high thrust-to-mass ratio as currently a large fraction of propulsion system dry mass (driving to higher I_{sp})

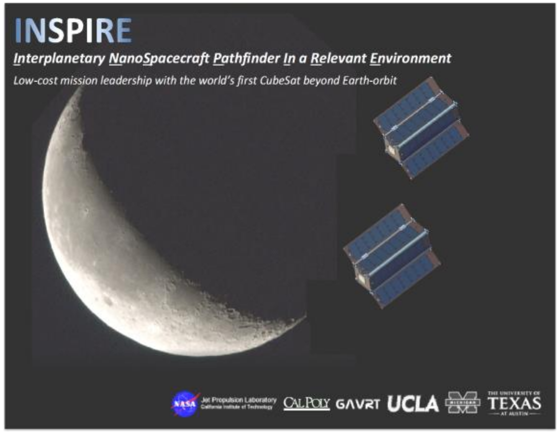


Results-

TABLE III
LEO AND DEEP SPACE ORBIT TRANSFERS STUDIED. ALL LEO ORBITS ARE CIRCULAR AND GEOS ARE EQUATORIAL.

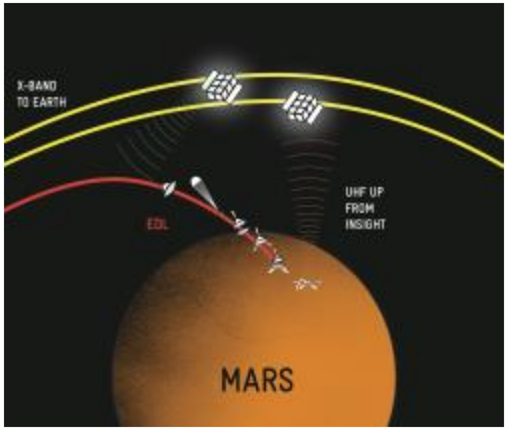
Name	Initial Orbit	Final Orbit	ΔV range	Burn Time (days)	Time of Flight (days)
Fly Over	See Fig. 3(b)	Polar, 574 km	0.5-0.8 km/s	60	60
GTO to Lunar Flyby	GTO	Earth Escape	1.7-3.4 km/s	50-559	120-735
LEO to GEO	Equatorial, $a = 500$ km	GEO	4.4-5.5 km/s	77-658	178-917
ISS to Polar	$i = 52°$, $a = 420$ km	Polar, $a = 420$ km	6.7-7.6 km/s	82-543	121-553
ISS to Equatorial	$i = 52°$, $a = 500$ km	Equatorial, $a = 420$ km	9.3-11.3 km/s	85-711	123-735
Deimos Return	Deimos	Earth Orbit	4.6-8.6 km/s	17-738	719-1206
Phobos Return	Phobos	Earth Orbit	5.4-9.4 km/s	21-869	723-1336

Active Interplanetary CubeSat Projects

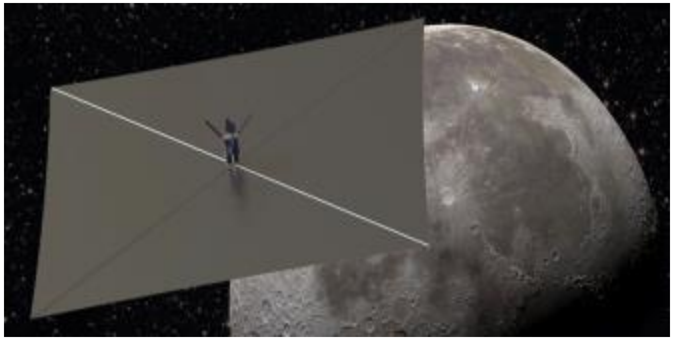


INSPIRE (JPL)[1]
Navigation demonstration with the IRIS radio beyond the Moon

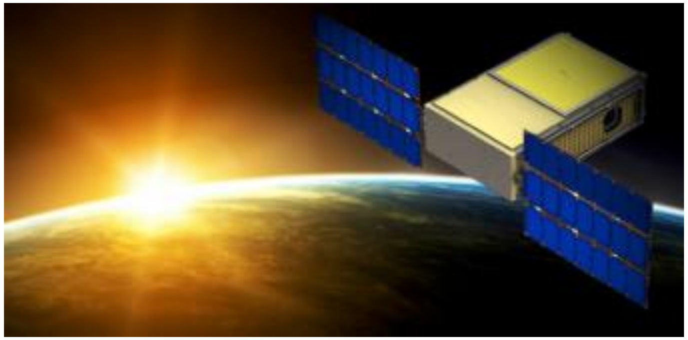
NEA Scout (MSFC/JPL)[2,3]
Asteroid characterization mission [EM-1]



MarCO (JPL)[2]
InSight insertion real-time relay



Lunar Flashlight (JPL/MSFC)[2,3]
Lunar orbiter to search for ice in lunar craters [EM-1]

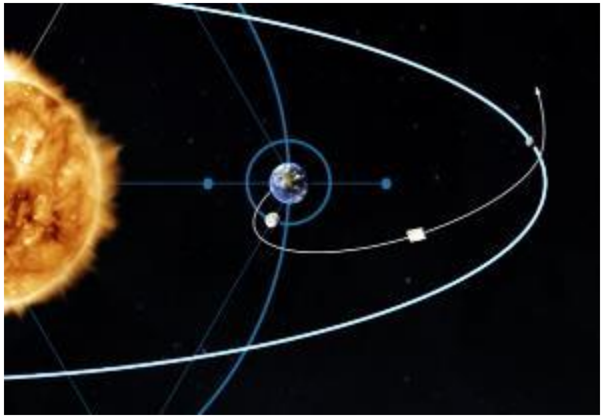


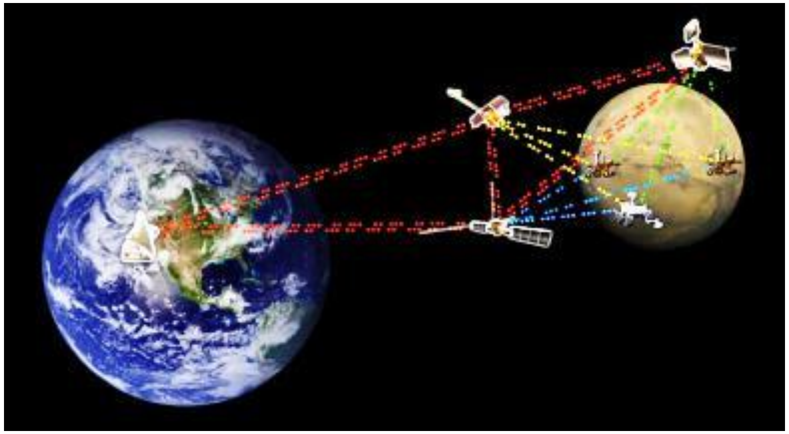
BioSentinel (Ames)[2,3]
Biosensor to study impact of radiation on living organisms [EM-1]

[1]JPL/NASA Planetary Science Division, [2]JPL, [3]NASA's Advanced Exploration Systems (AES)

Future Impact of Science-Driven Small Spacecraft

- ***Performing significant ΔV and high-precision attitude control enables:***
  - Escaping Earth-orbit, transferring to Moon, Mars, asteroids, coments, beyond
  - Creating and maintaining formation flight/constellations (*e.g.* large apertures)
- ***Autonomous Operations enabling:***
  - Autonomous navigation by imaging asteroids (*e.g.* DS1)
  - Agile Science for on-board autonomy to locate Earth, detect objects (*e.g.* plumes)
  - Dynamic observation planning, disruption-tolerant networking (DTN)
- ***Science Mission Applications to perform SMEX/Discovery-class science:***
  - Multi-spacecraft architectures: constellations, mother-daughtership, swarms, formation flying to perform distributed temporal/spatial measurements
  - Pre-cursor missions to explore dirty/dangerous/unknown environments (comets, asteroids, moons, Earth-Sun Lagrange points)

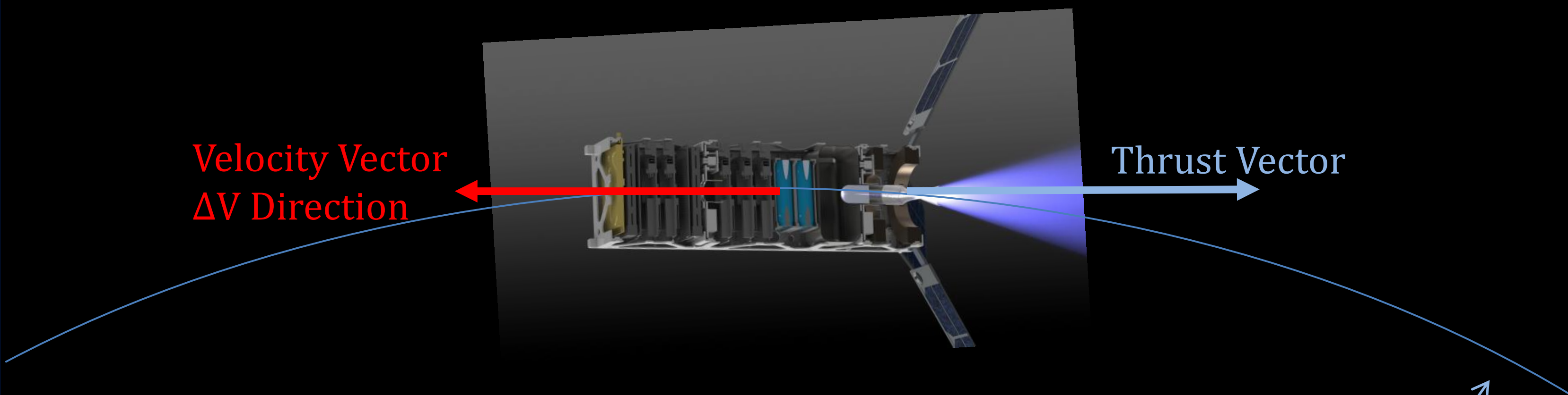


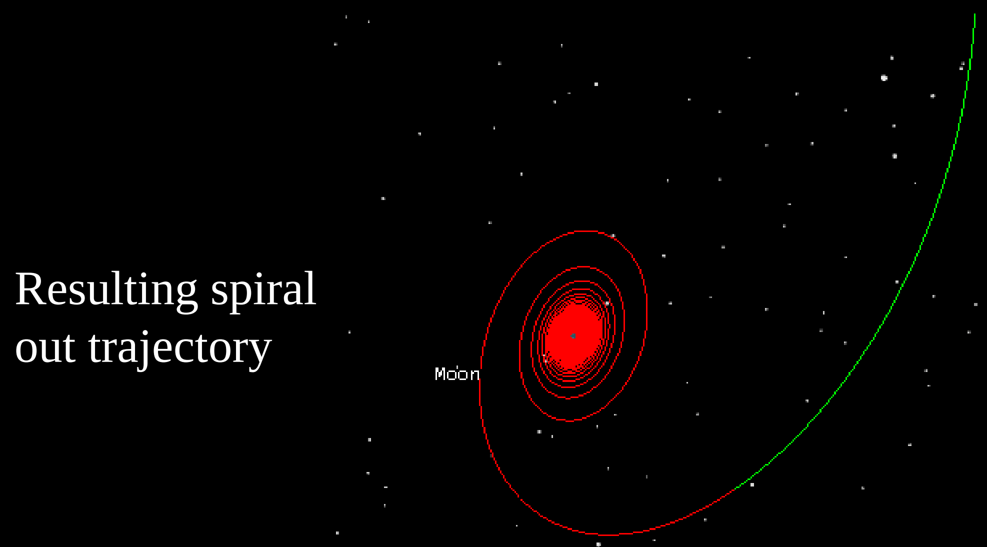


Agile Science Reference: D. R. Thompson, S. A. Chien, J. C. Castillo-Rogez

Case 1: Constant Thrusting in Velocity Direction

Simplest and (usually) most time efficient approach to raise altitude

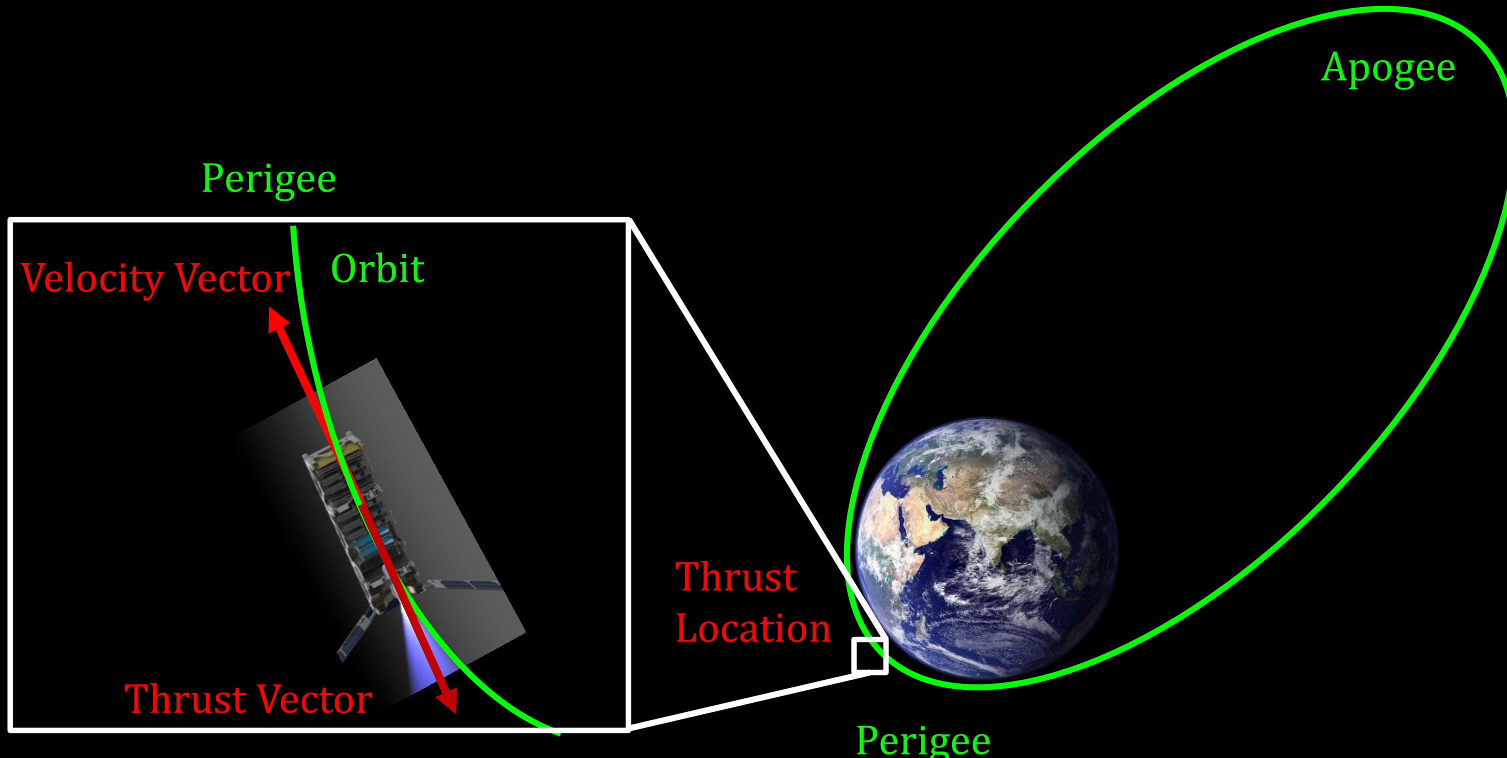




Red shows thrust/ Green shows cruise

Case 2: Problem Description

The scheme where we thrust only at perigee exploits the fact that increasing the ΔV at perigee (gravity well) results in greater apogee raises.



This approach may be more (time and fuel) efficient relative to the constant thrust approach (Case 1).

Case 1: What's the Impact of Attitude Control Errors?

Results given for orbit starting in 500 W circular orbit until Earth-escape (925,000 km)

- Even with γ=20°, only requires an additional 13.1 days (10W)/ 5.2 days (25 W)
- Orbit shape and precession will also change with cross-track ΔV component

Angular Error (γ)	Actual/ Ideal Thrust Ratio	Increase in Time Constant Thrust (10 W)	Increase in Time Constant Thrust (25 W)
1°	0.9998	0.02%	0.02%
5°	0.9962	0.4%	0.4%
10°	0.9848	1.5%	1.5%
20°	0.9397	6.4%	6.4%