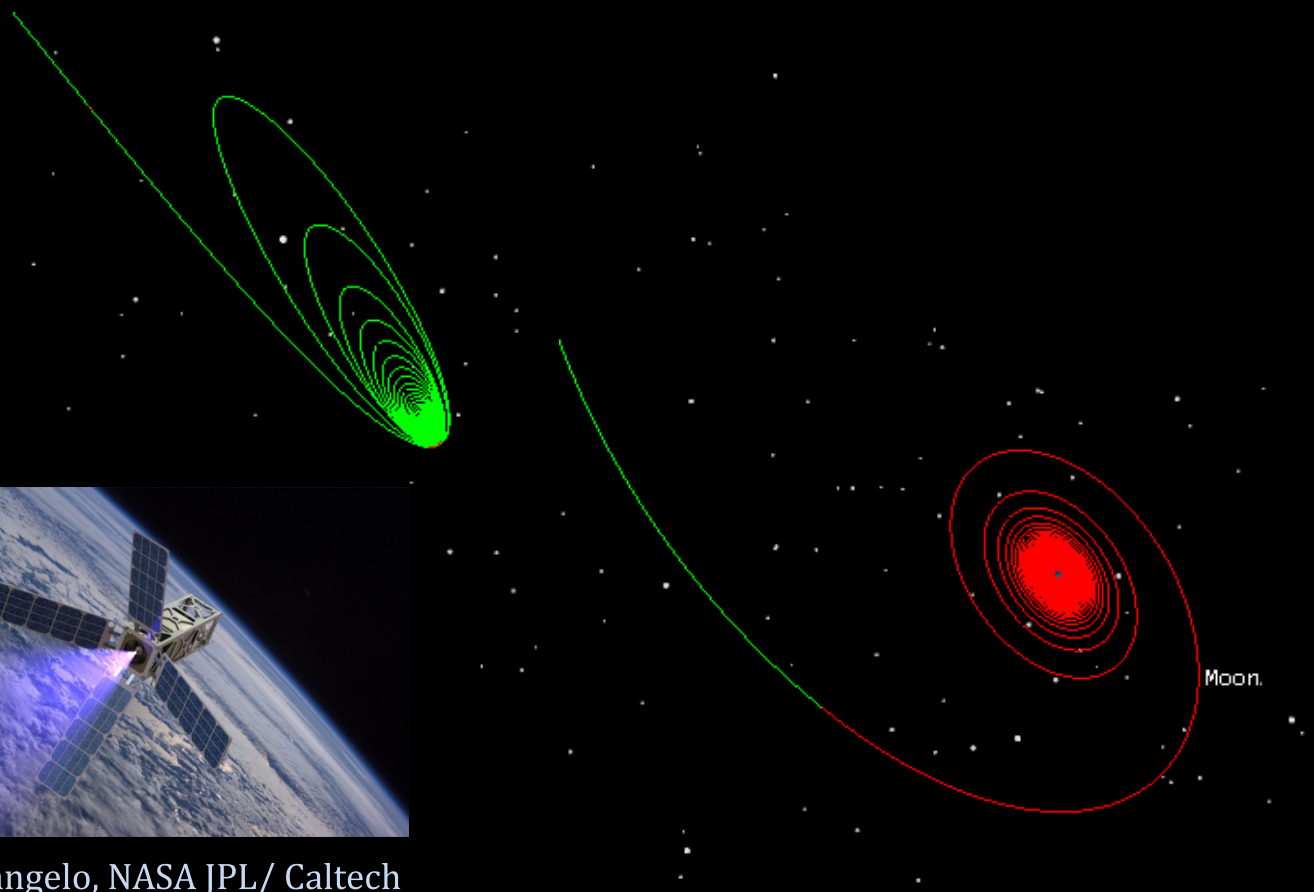
SPACE 2014

Small Spacecraft System-level Design and Optimization for Interplanetary Trajectories



Sara Spangelo, NASA JPL/ Caltech
Derek Dalle, University of Michigan
Benjamin Longmier, University of Michigan
SPACE Conference, August 2014

How Far Can CubeSats Go (Alone)?

SPACE 2014

- Can CubeSats go beyond Low Earth Orbit (LEO)? ***Yes***
  - Is there a fundamental size, mass, power, cost limitation? ***No!***

- Enabling factors:
  - Miniaturized thruster technology (CAT) with high ΔV capabilities
  - Miniaturized attitude control technology (Blue Canyon's XB1)
  - Heritage and experience operating CubeSats in LEO
  - Optimal use of volume and mass, and scheduling of available energy and time

Photo Credit: NASA Website

Past Work & Objectives

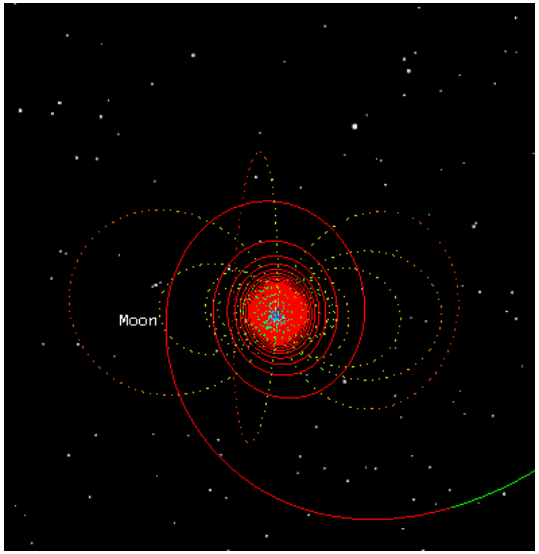
SPACE 2014

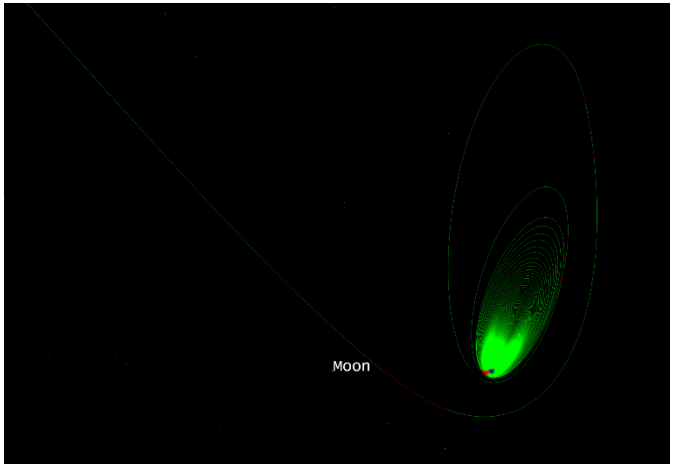
Previous work:

- Demonstrated the feasibility of escaping Earth orbit with the CubeSat Ambipolar Thruster (CAT) in a 3U CubeSat starting in LEO
- Optimized trajectories to minimize time, fuel, and radiation exposure
- Considered different orbit maneuvers (constant thrust, optimal thrust at perigee)
- Assessed trade-off between required fuel, batteries, and time to escape Earth orbit

The goal of this work is to:

- Demonstrate LEO to Earth-escape maneuvers:
- Investigate transfers from GEO to interplanetary destinations (Mars, Jupiter, Saturn)
- Identify trade-offs/ sensitivities to propellant, power, mass, transfer time





CAT: Large ΔV Engine Capability

SPACE 2014

CAT: CubeSat Ambipolar Thruster

- Uses high-density plasma source
- Achieves high ΔV *and* high thrust/power
- Fits within small spacecraft form-factor (<0.1 U)
- Awarded a CSLI* Launch on PATRIOT mission awarded in 2014
- Successful Kickstarter Campaign resulting in seed funding ($100 K)
- Some commercial funding supporting tech development

Design of a 3U 3U CubeSat with CAT engine performing initial testing in Low Earth Orbit.

*CSLI: NASA's CubeSat Launch Initiative

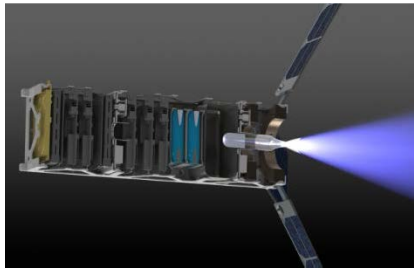
Photo Credit: PEPL

CAT: Large ΔV Engine Capability

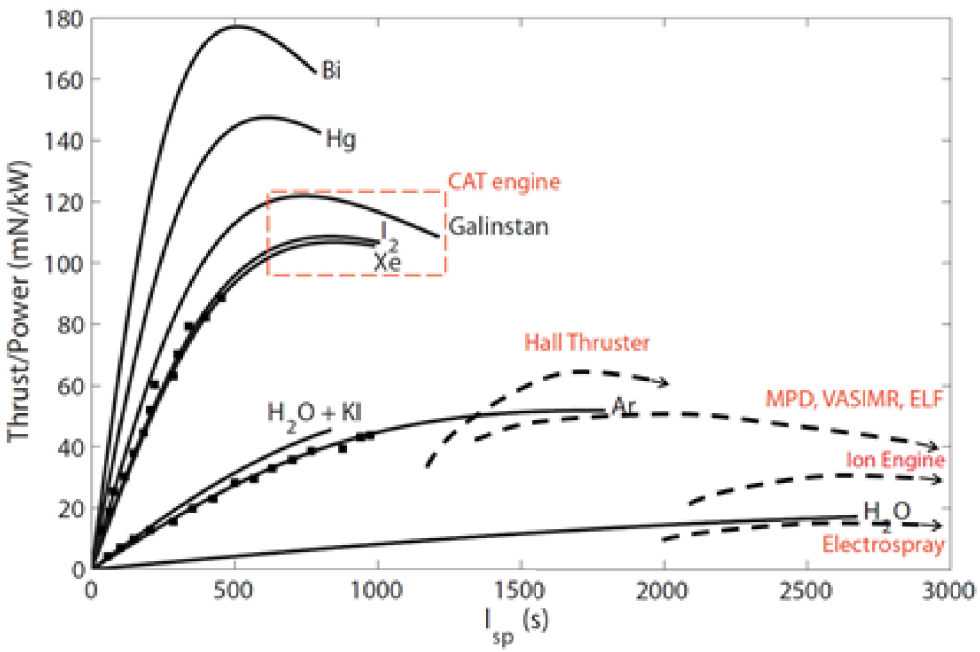
SPACE 2014

CubeSat Ambipolar Thruster (CAT)

- Mass: <0.5 kg, Volume: 0.1U
- Iodine fuel (I_2), I_{sp}=1010 sec, Density=5 g/cm^3
- Operating power levels: 3-300 W
- 50-60% thruster efficiency



CAT engine with CubeSat subsystems



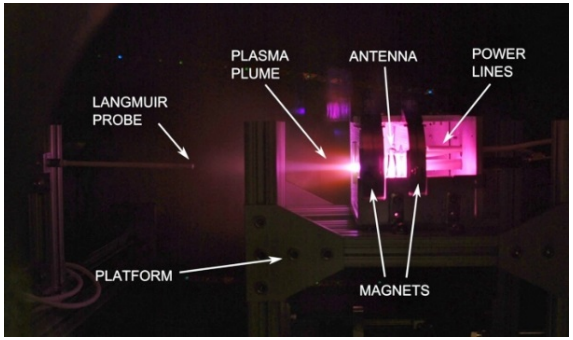


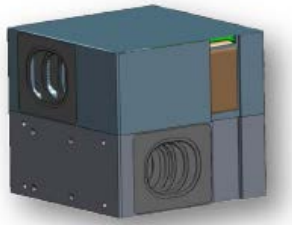
Image Credit: PEPL

Vehicle Description

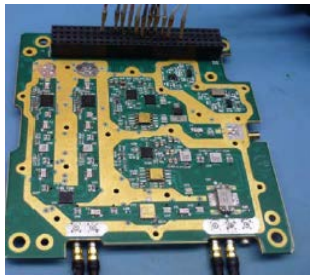
SPACE 2014

3U CubeSat Spacecraft State-of-the-Art Components

- CAT Thruster & Propellant Tank (~10% mass propellant)
- IRIS Transponder for interplanetary navigation and telecom
- Blue Canyon XB1 Bus (GNC, C&DH, Telecom, Power, ACS)
  - Pointing: 7.2 arcsec accuracy, 1 arcsec stability,
  - <2.5 kg, ~1 U, <2.5 W
- Aluminum 3U CubeSat Structure
- Deployable MMA Solar Arrays (~75 W in sun)
- 4 x Li-Ion batteries (~15 Whr/each) in addition to 25 Whr in XB1



XB1 Blue Canyon System



JPL Iris Transponder

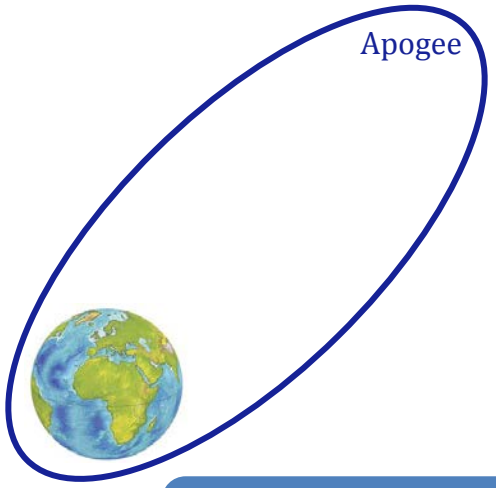
Table 1. Representative Master Equipment List (MEL) for a 6U spacecraft, where a 1 U is 1000 cm^3

Component	Mass (kg)	Volume (1 U fraction)	Max Power (Watts)
Thruster	0.5	0.1	3-300
Fuel Tank	1.0	0.1	0
IRIS Transponder	0.4	0.4	12.5
Antennas (Telecom)	0.05	0.05	-
Blue Canyon XB1 Bus (Telecom, GNC, CDH, EPS)	1.2	1.0	2.5
Extra Batteries (4 x 18650 Li-Ions)	0.04	0.05	-
Structure/Shielding	0.9	0.19	-
Deployable Solar Panels	0.8	0.5	-
Dry Spacecraft Total	6.15	3.68	17-315

Image Credit: Blue Canyon, JPL

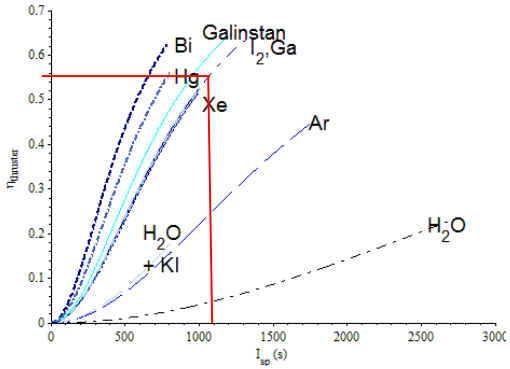
Multidisciplinary Modeling Framework

SPACE 2014



Orbit Design

- ΔV boosts orbit altitude
- Conservation of energy and angular momentum

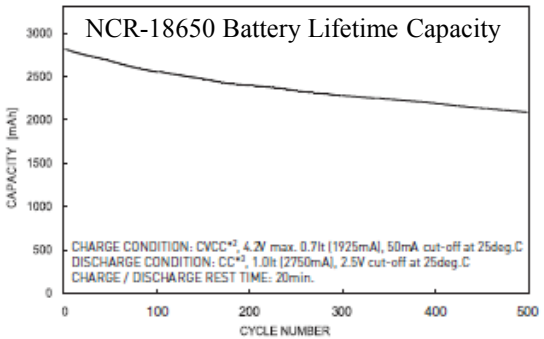


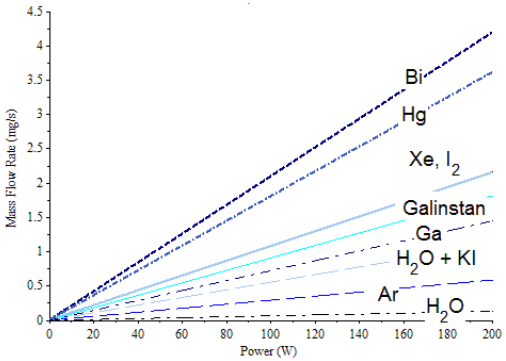
Energy

- Battery Capacity
- Solar Power Collection
- Eclipse Management

Propulsion

- ΔV from Rocket Equation (mass flow from power, time)





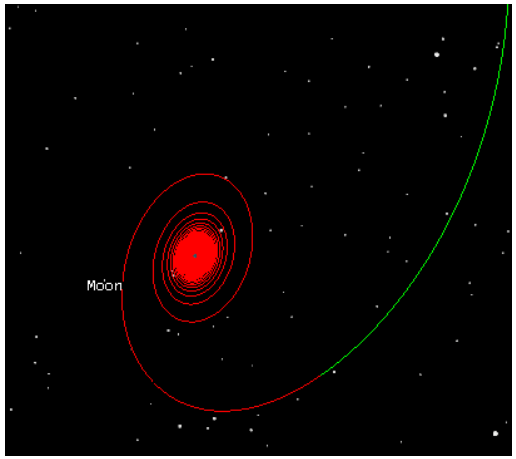
We also analyze attitude control and radiation, but not as part of the optimization problem

Multidisciplinary Modeling Framework

SPACE 2014

Goal: Find the trajectory to minimize time to escape Earth's sphere of influence (radius: 925,000 km) and reach interplanetary destinations (Mars, Jupiter, Saturn)

Decisions: Power setting for constant thrusting maneuvers



Constraints:

- Thruster power settings (3-300 W)
- Battery capacity constrains available energy
  - Battery capacity degradation*
- Positive energy balance throughout each orbit
- CubeSat form factor (3 U-6 U)

Dynamics:

- Propulsion:
  - Mass flow rate is linearly related to power setting
  - ΔV computed by rocket equation based on current mass, mass flow rate
- Orbit: Conservation of angular momentum and energy to compute orbit changes
- Energy: Model realistic collection and consumption (solar, nominal, thrust)

Inputs: Initial altitude, solar panel and nominal power, initial dry/ fuel mass

*Reference: http://www.duckware.com/blog/tesla-elon-musk-nytimes-john-broder-feud/index.html

Analytic Model: Propulsion, Orbit, Energy

SPACE 2014

Propulsion Dynamics

$$\Delta V = V_{ex} \ln\left(\frac{m_i}{m_f}\right), \qquad \dot{m} = \delta P,$$

$$V_{ex} = g I_{sp},$$

Orbit Dynamics

$$\dot{v}_r = \frac{v_\theta^2}{r} - \frac{\mu}{r^2} + a \sin\gamma, \qquad v = \sqrt{v_r^2 + v_\theta^2},$$

$$\dot{v}_\theta = a\cos\gamma - v_r\theta, \qquad h = v_\theta r,$$

$$\dot{\theta} = \frac{v_\theta}{r}, \qquad e = \sqrt{\frac{(2\mu - rv^2)rv_r^2 + (\mu - rv^2)^2}{\mu}},$$

$$\gamma = \tan^{-1}\left(\frac{v_r}{v_\theta}\right), \qquad r_a = \frac{h^2}{\mu(1-e)},$$

$$a = -\frac{V_{ex}\dot{m}}{m} \qquad r_p = \frac{h^2}{\mu(1+e)}.$$

Energy Dynamics

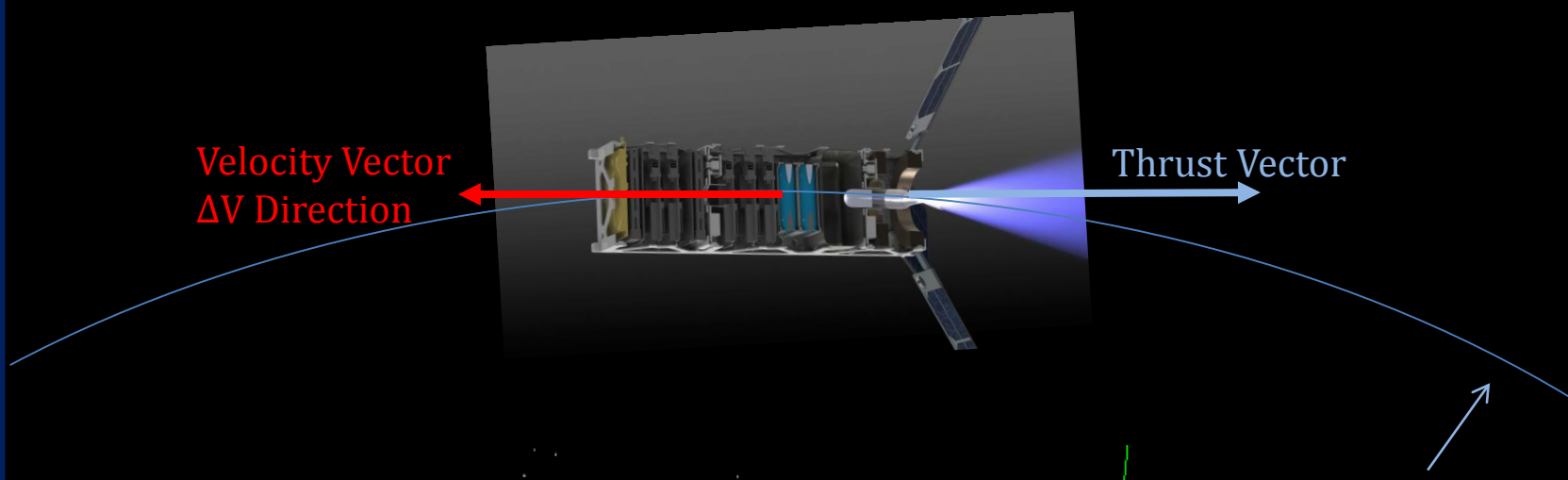
$$E_{batt} \geq P_{t,i} t_{t,i}, \qquad \forall i \in I,$$

$$E_{batt} \geq P_{n,o} t_{e,o}, \qquad \forall i \in I.$$

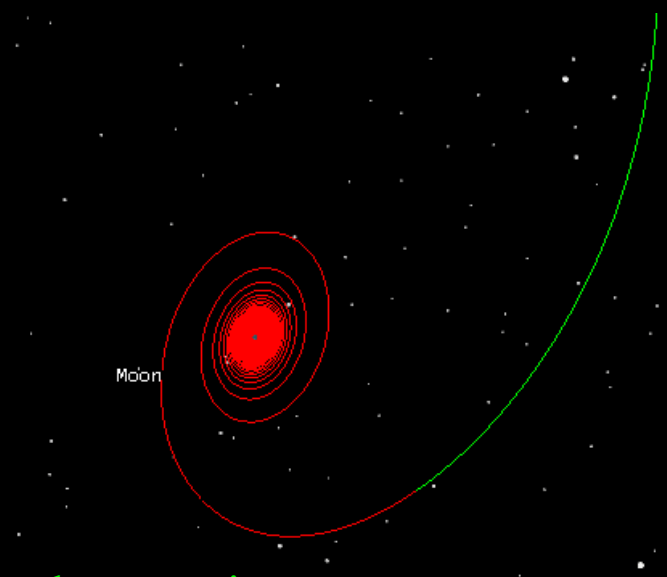
$$P_{s,o} t_{s,o} \geq P_{t,o} t_{t,o} + P_{n,o}(t_{s,o} + t_{e,o}), \qquad \forall o \in O,$$

Orbit-boosting in Earth Orbit: Constant Thrusting

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



Resulting spiral out trajectory



Red shows thrust/ Green shows cruise

Earth Escape ΔV and Escape Time

SPACE 2014

Worst-case ΔV requirements for different destinations

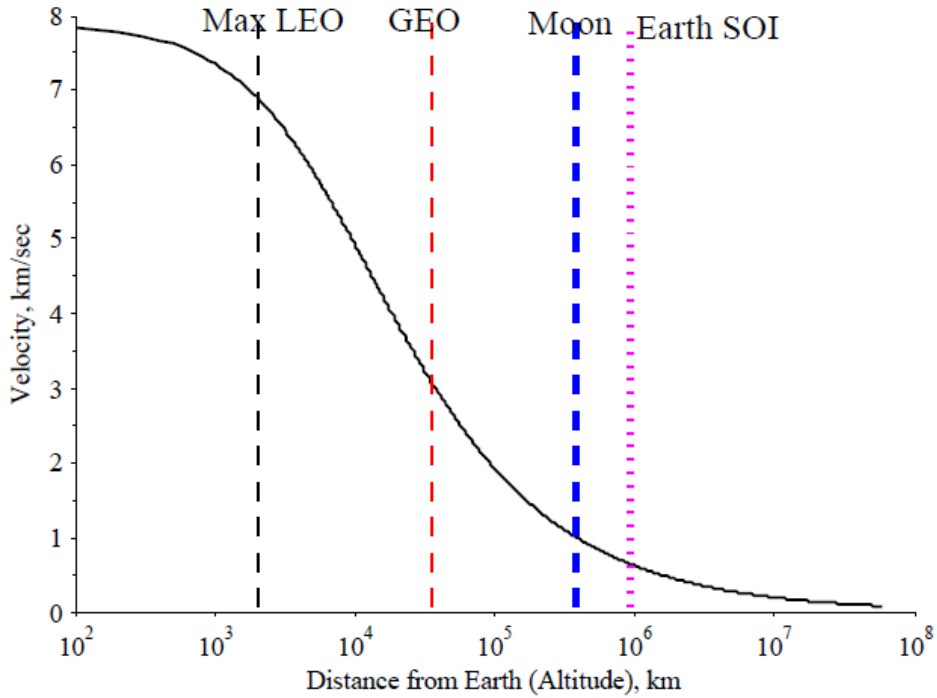


Table 3. Time required for different destinations and propellants, assuming orbit starts in 500 km circular orbit with 10 W constant thrust and a spacecraft dry mass of 2.5 kg.

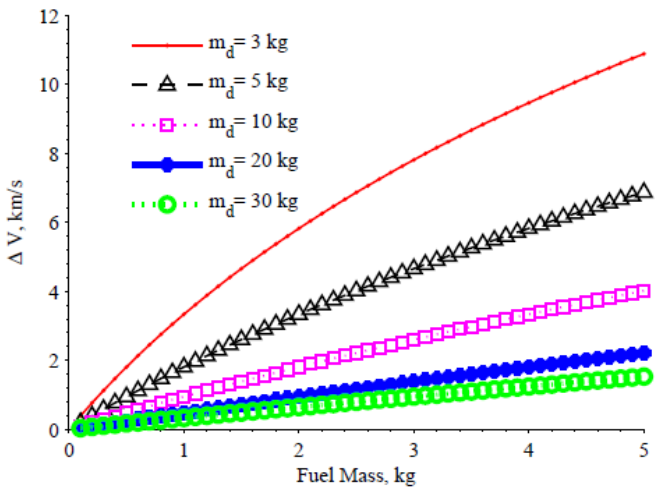
Parameter	Units	Maximum LEO	GEO	Mean Moon	Earth's SOI
Distance from Earth	km	2,000	35,700	384,000	925,000
ΔV	km/ sec	0.6	4.4	6.5	6.9
I_2 Propellant	days	32	194	259	269
Ga Propellant	days	36	226	308	321
H_20 Propellant	days	202	1,369	1,933	2,026

Earth Escape Maneuvers: Mass and Power Sensitivity

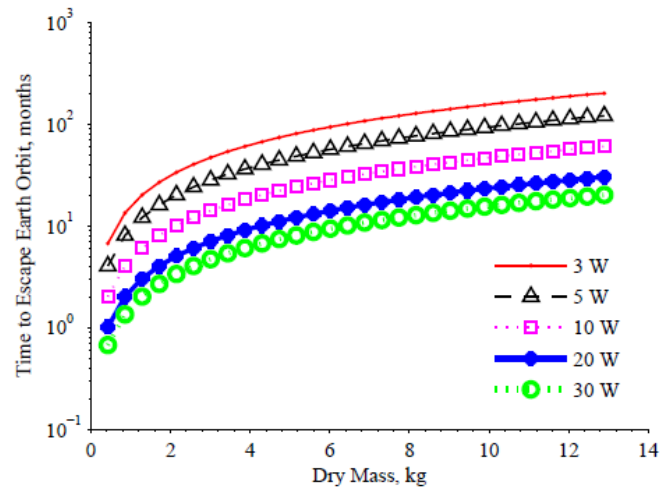
SPACE 2014

Table 2. Fuel mass required for different destinations and propellant types, assuming initial orbit of LEO 500 km altitude with 10 W constant thrust and a spacecraft dry mass of 2.5 kg.

Parameter	Units	Maximum LEO	GEO	Mean Moon	Earth's SOI
Distance from Earth	km	2,000	35,700	384,000	925,000
ΔV	km/s	0.6	4.4	6.5	6.9
I_2 Propellant	kg	0.3	1.8	2.4	2.5
Ga Propellant	kg	0.2	1.4	1.9	2.0
H_20 Propellant	kg	0.1	0.8	1.1	1.1



a) ΔV with a dry spacecraft mass of 2.5 kg



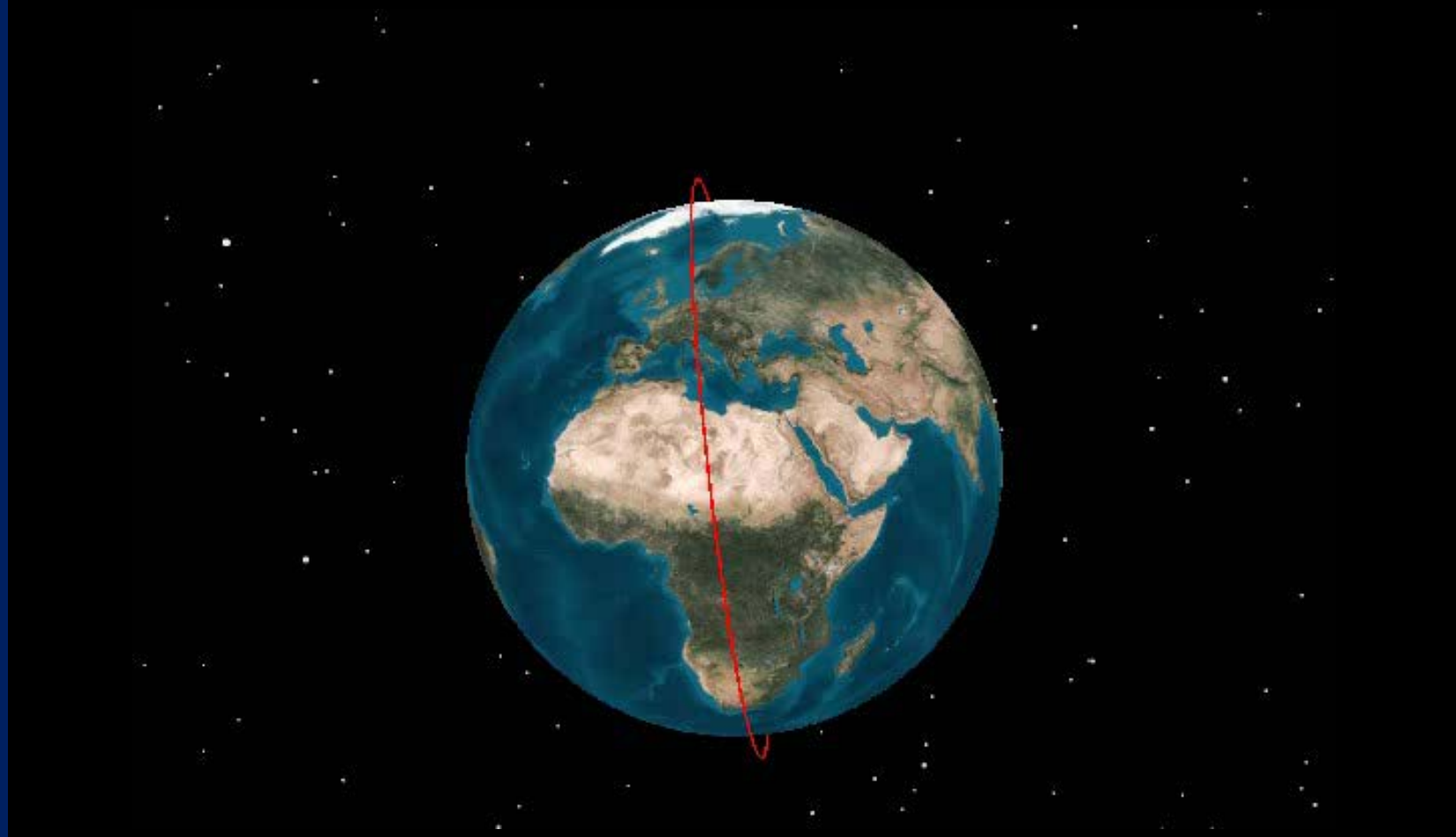
b) Earth escape time for initial 500 km circular orbits (requires $\Delta V = 6.96$ km/s)

Significant advantages with lower-mass and higher-power systems

Orbit-boosting for Earth Sun-Sync Orbits

SPACE 2014

Starting from "worst-case" 500 km Sun Synchronous orbit ($\beta^* \approx 60^o$), with 25 W power setting, altitude is boosted such that there is no eclipse (>750 km) in <3 days!



Total Escape Time: 108 days,
Total Escape Fuel: 2.5 kg

β^*: Angle between orbital plane and vector to Sun

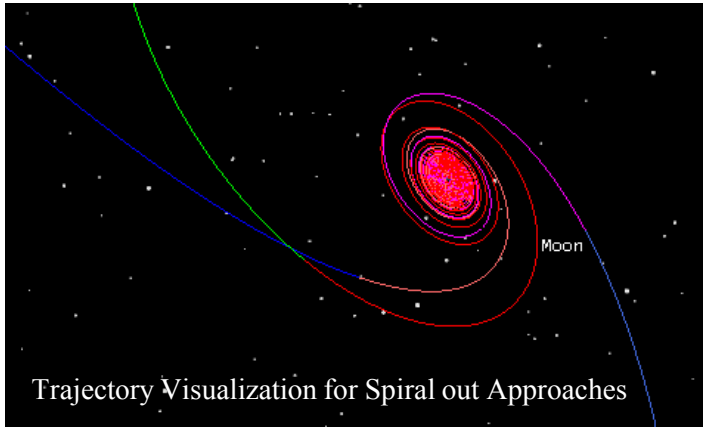
Earth Escape Maneuvers: Sensitivity to Power Level

SPACE 2014

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

Constant Thrust Power Values	10 W	20 W	25 W
Fuel Quantity	2.5 kg	2.5 kg	2.5 kg
Time	269 days	134 days	108 days
Energetic Feasibility	Feasible in any orbit	Only feasible in (terminator) sun sync orbit	Only feasible in (terminator) sun sync orbit
Number of Orbits	1322	681	545
Total Accumulated Ionizing Dose with 82.5 Mils after 1 year	29.99 krad	15.01 krads	12.12 krads

- Considerable time savings to escape Earth with increased power values
- 10 W case is only feasible case for all orbits
- 20-25 W cases feasible with sun sync orbits (unable to maintain enery balance)
- Alternative approach: increase power value as altitude increases (eclipse fraction decreases)



Trajectory Visualization for Spiral out Approaches

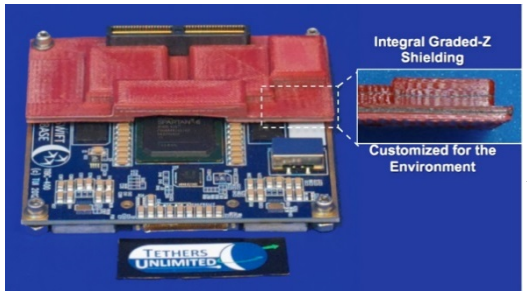
Earth Escape Maneuvers: Radiation Effects & Mitigation

SPACE 2014

Radiation dosage after one year of mission operation (starting 500 km orbit)

Aluminum shielding thickness	Case and Description	
	Constant Thrust (10 W)	Constant Thrust (25 W)
82.5 Mils (2.1 mm)	29.99 krad	12.12 krad
232.5 Mils (5.9 mm)	3.89 krad	1.96 krad
457.5 Mils (11.6 mm)	2.20 krad	1.11 krad
Predicted Al thickness for Accumulated Dosage <5 krad*	5.5 mm	4.5 mm

- CubeSat components shown to fail between 5-10 krad*
- Radiation effects can be mitigated by enclosing sensitive chips with Al or other protection schemes
  - e.g. 3D-printed custom radiation protection solutions



Tether's Unlimited Radiation Shielding

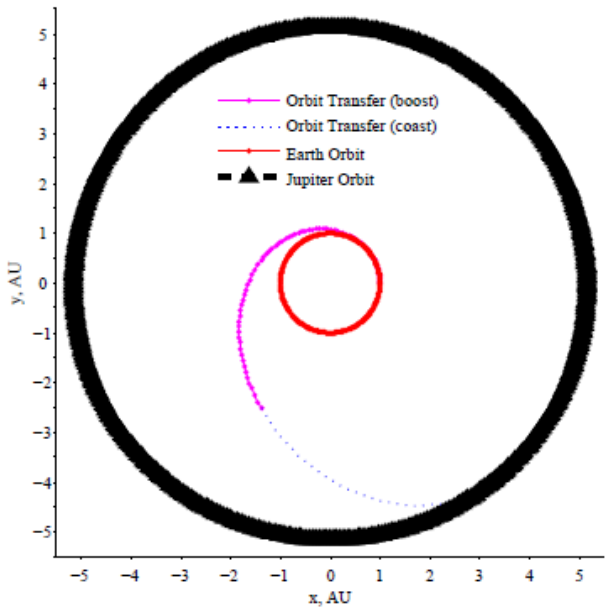
ISIS 3U CubeSat Al Structure

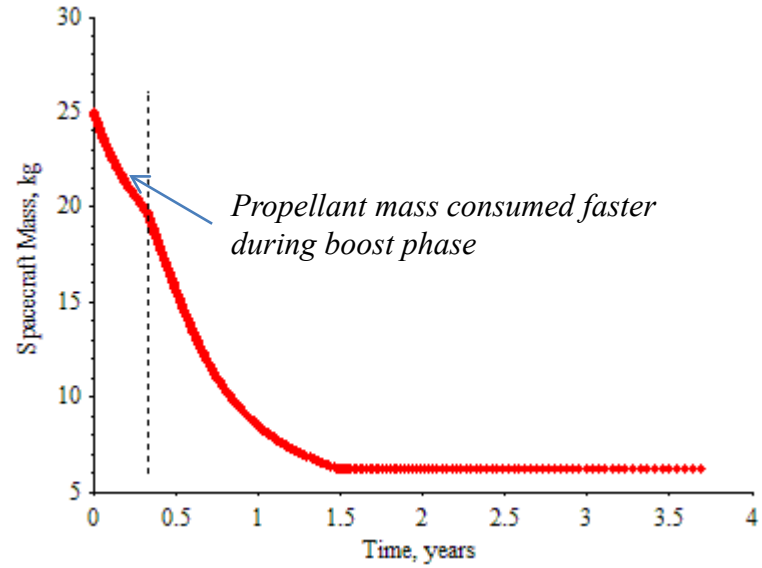
*Total Dose Test Results for CubeSat Electronics, Finchel et al.

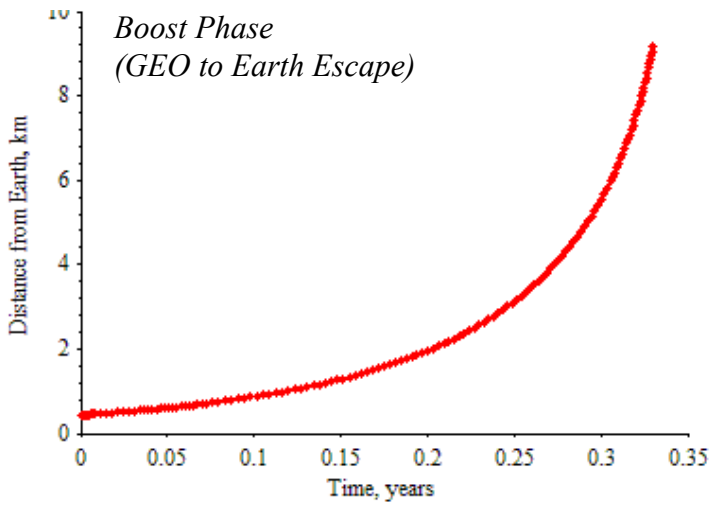
Interplanetary Transfer from GEO to Jupiter

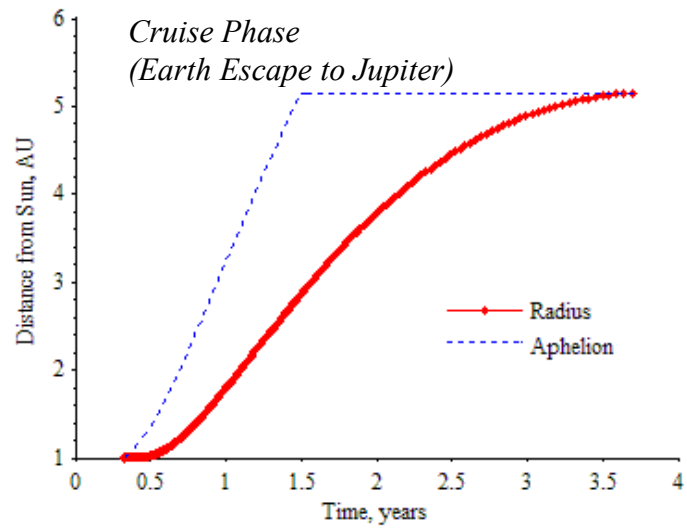
SPACE 2014

Transfer from GEO to Jupiter assuming 11 kg initial mass, 6.2 kg dry mass CubeSat





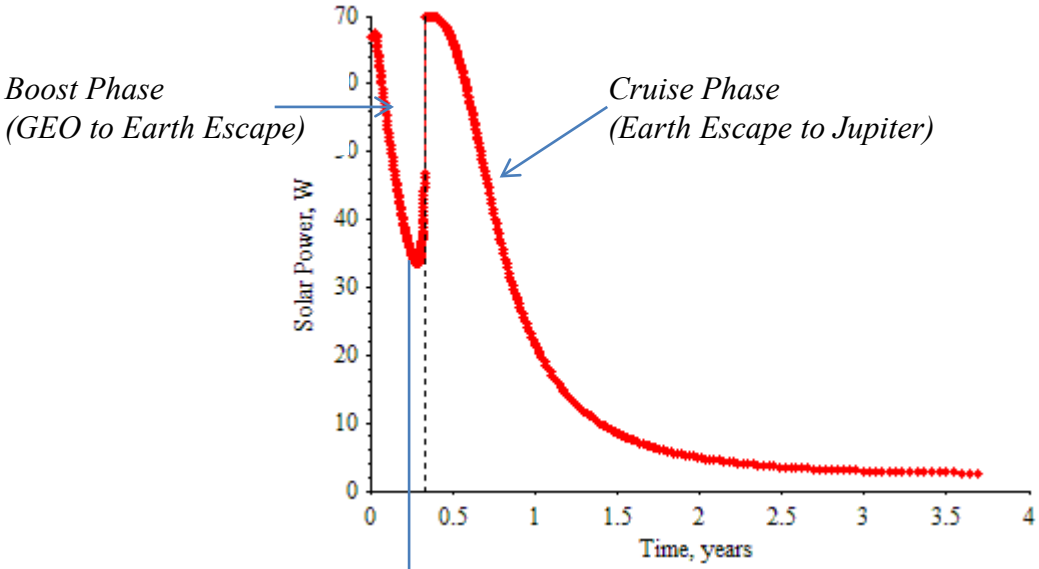


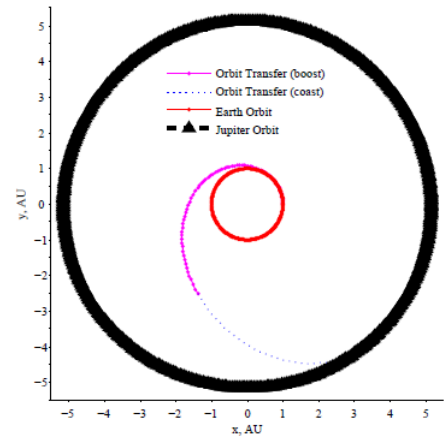


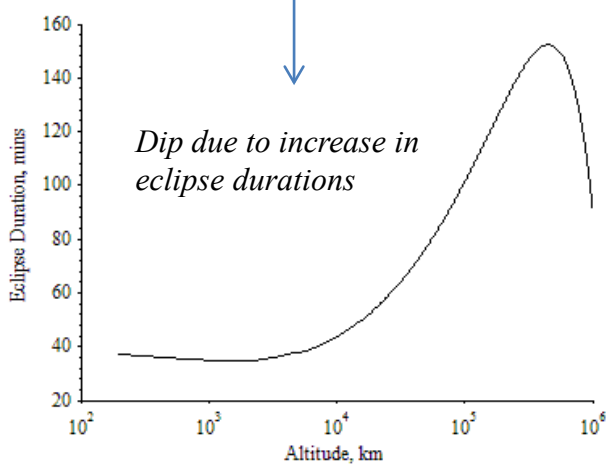
Eclipse Trends Impact Available Power

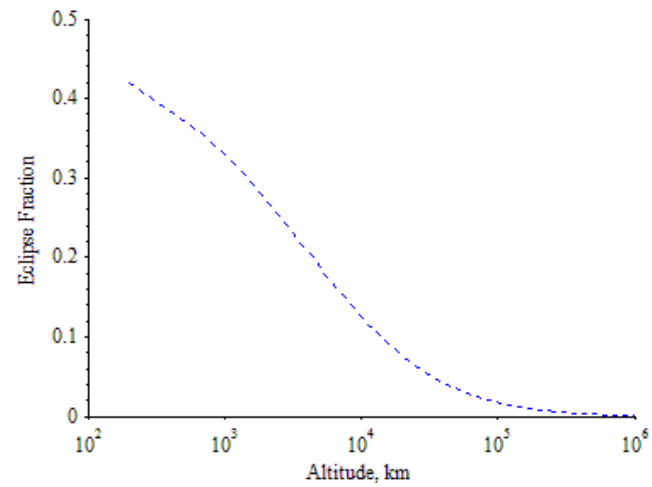
SPACE 2014

Available power during transfer depends on eclipse trends (boost) and distance from Sun (cruise)







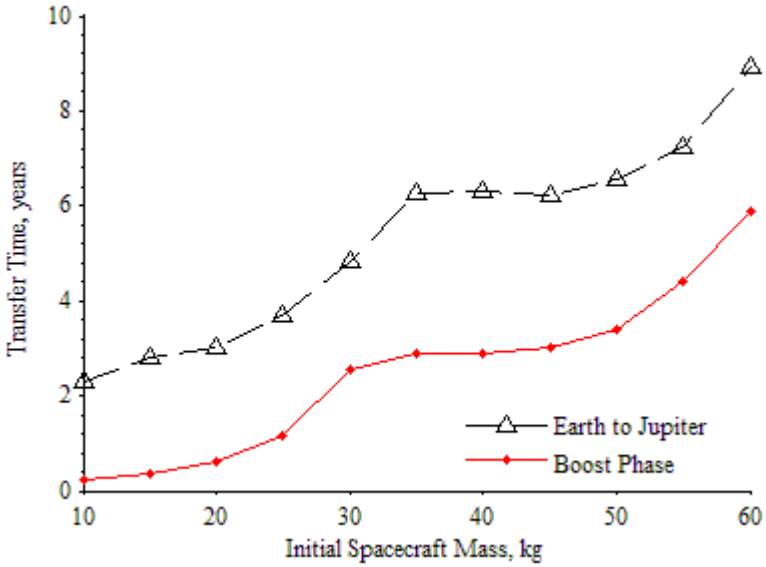


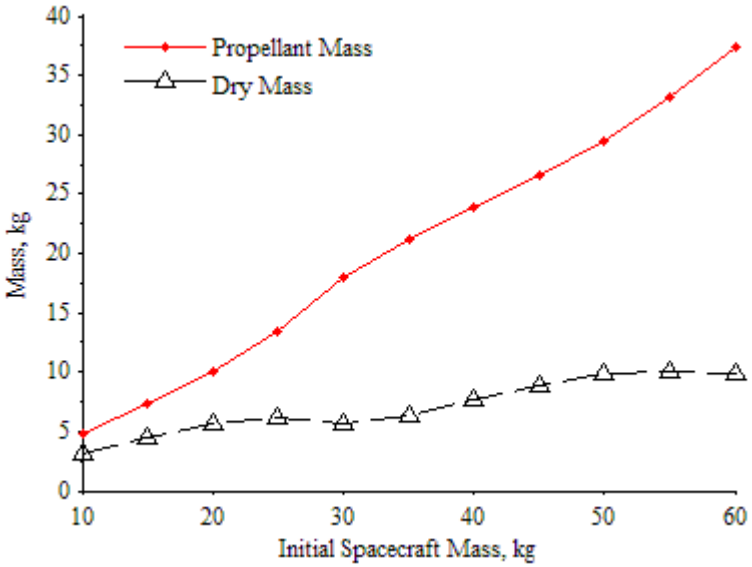
Although eclipse fraction decreases, eclipse time increases due to change in orbital period

Interplanetary Transfer: Mass Sensitivity

SPACE 2014

Sensitivity to initial mass for 70 W CubeSat from GEO to Jupiter





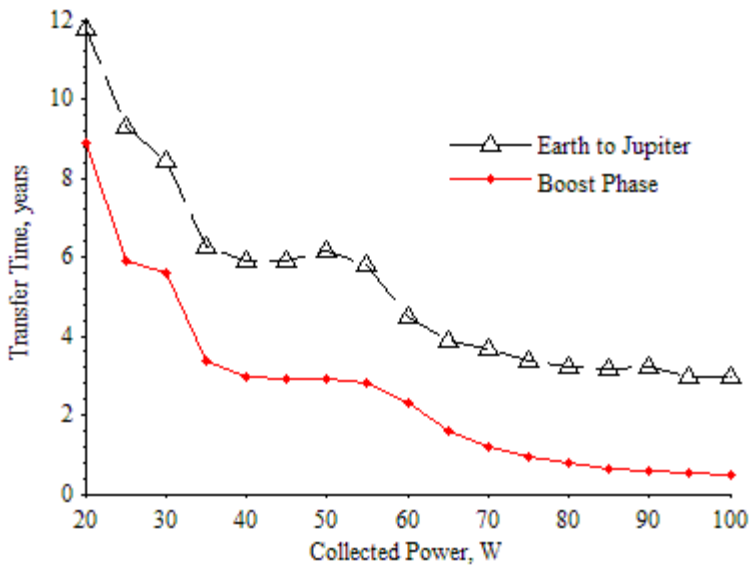
"Knee" in curve shows where more mass doesn't improve transfer times

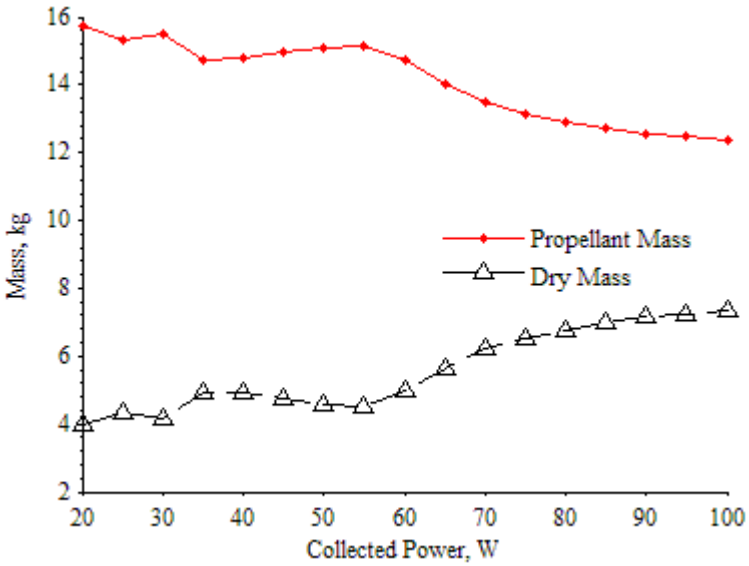
Mass fraction grows significantly with higher mass

Interplanetary Transfer: Power Sensitivity

SPACE 2014

Sensitivity to maximum power for 25 kg CubeSat from GEO to Jupiter





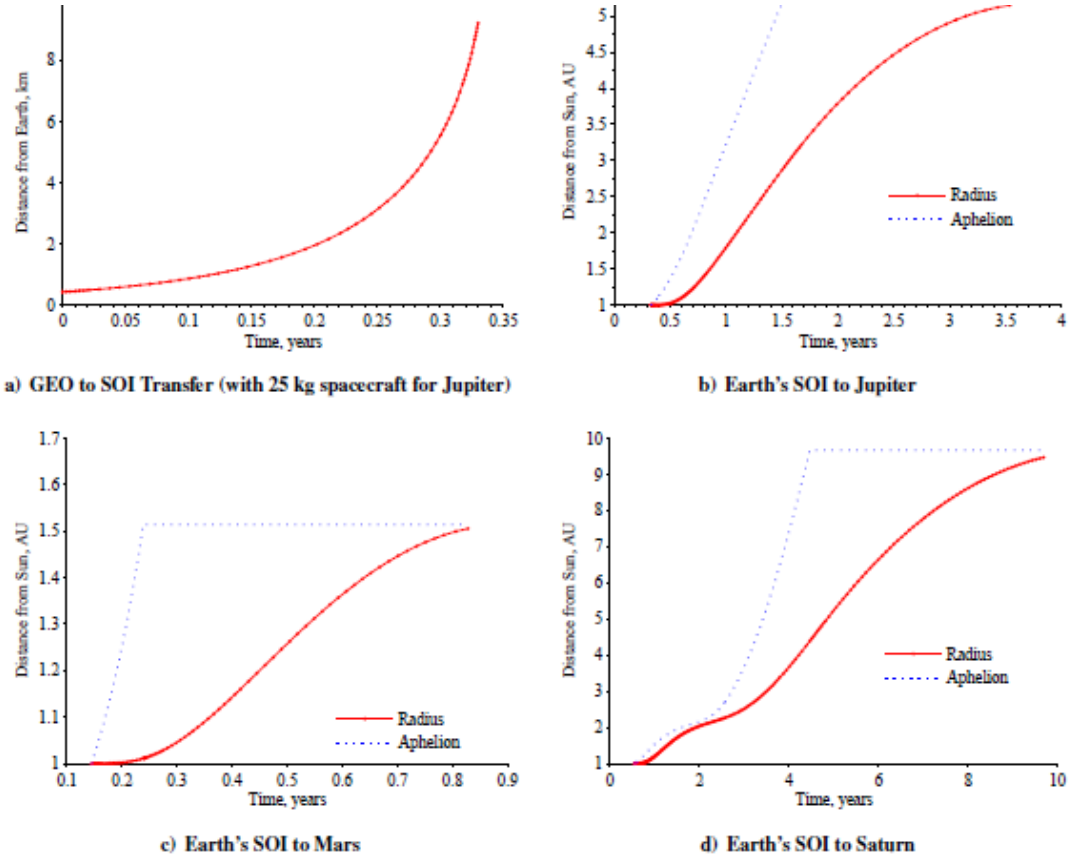
Plateau shows where higher power doesn't improve transfer times

Mass fraction decreases with higher power values

Transfers from GEO to Mars, Jupiter, Saturn

SPACE 2014

Possible to transfer to Mars, Jupiter, and Saturn with reasonable transfer times and masses!



a) GEO to SOI Transfer (with 25 kg spacecraft for Jupiter)

b) Earth's SOI to Jupiter

c) Earth's SOI to Mars

d) Earth's SOI to Saturn

Table 5. Properties of travel to interplanetary destinations assuming starts in equatorial GEO, assuming initial power collection of 70 W at 1 AU, a dry spacecraft mass of 6.2 kg and I_2 propellant, and worst-case travel distances.

Planet Destination	Mars	Jupiter	Saturn
Distance from Earth	0.52 AU	4.20 AU	8.54 AU
Travel Time GEO to Earth's SOI	0.15 years	0.33 years	0.55 years
Travel Time to Destination	0.83 years	3.54 years	9.69 years
Initial Spacecraft Mass	11.00 kg	25.00 kg	42.00 kg
Fuel Mass	4.56 kg	12.50 kg	35.68 kg
Solar Power Percentage at Destination	44.1%	3.7%	1.1%

Additional Challenges and Future Work

SPACE 2014

Additional Mission Design Challenges

- Power system upgrades to manage higher power values
- Thermal issues with high-powered thrust maneuvers
- Communication throughout mission, impacts power, pointing at large ranges
- Attitude control errors impacting trajectory and efficiency
- Radiation-mitigation strategies

Future Work

- Integrated vehicle and trajectory optimization, including:
  - Power, thermal, radiation, and attitude control models
- Consider higher-fidelity orbit transfer models
- Analyze other orbit transfers and destinations
  - Lunar flybys, transfers to Moon, Mars, and beyond…

Back-up Slides

How Much Mass Is Needed?

Motivation
Problem
CAT Engine
Feasibility
Strategies
Conclusions

Work through Rocket Equation for I_2 fuel and CAT

Parameter	Symbol	Input/ Equation	Value	Units
Specific Impulse	I_{sp}	Input	1010	sec
Exhaust Velocity	V_{exh}	$V_{exh} = I_{sp} \cdot g$	9908	km/sec
Dry Spacecraft Mass	m_s	Input	2.5	kg
Propellant Mass	m_p	Input	2.5	kg
Initial Mass	m_i	$m_i = m_s + m_p$	5.0	kg
Final Mass	m_f	$m_f = m_s$	2.5	kg
Delta V Capability	ΔV	$\Delta V = V_{exh} \ln\left({m_i}/{m_f}\right)$	7.0	km/sec

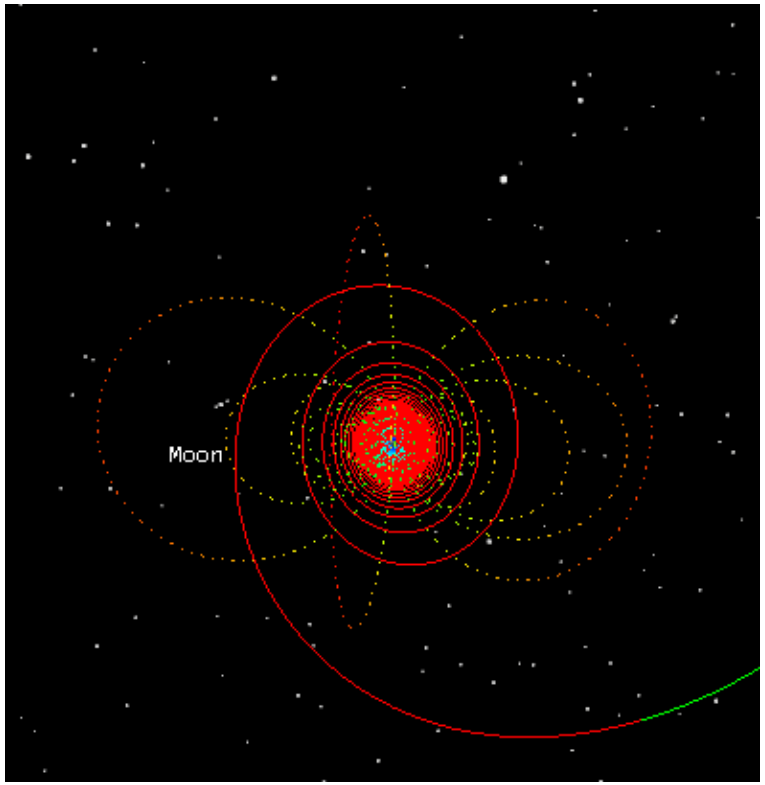
Ideal Rocket Equation

We can escape Earth's Sphere of Influence (ΔV~ 7km/sec) with ~2.5 kg of fuel!

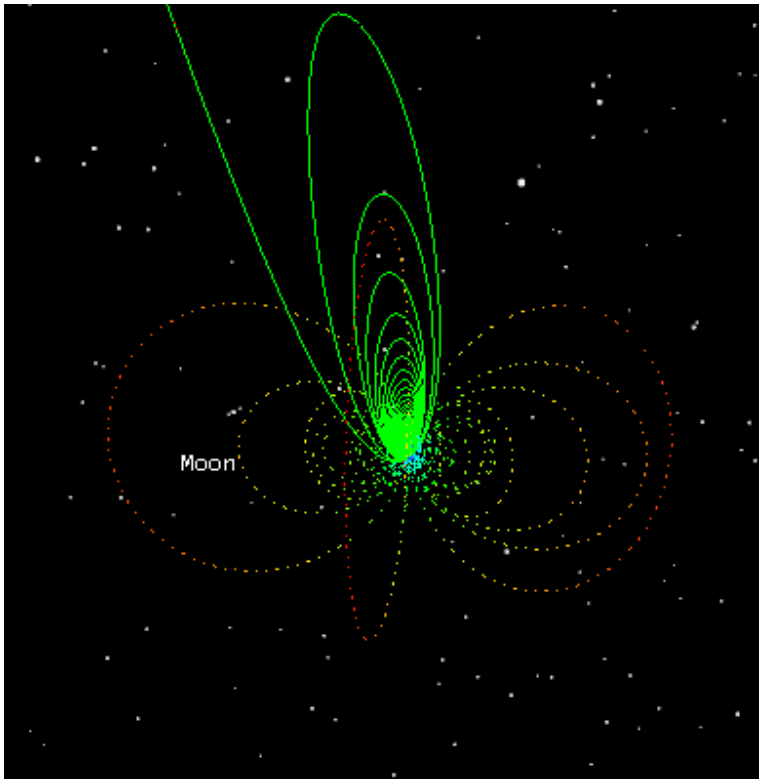
g: gravity constant = 9.81 m/sec^2

Case 1 and 2: Radiation Effects

- Total Accumulated Radiation Dosage measured with STK's SEET*
- Case 1 spends more time in radiation belts than Case 1



Case 1: Constant Thrust in Velocity Direction (10W)

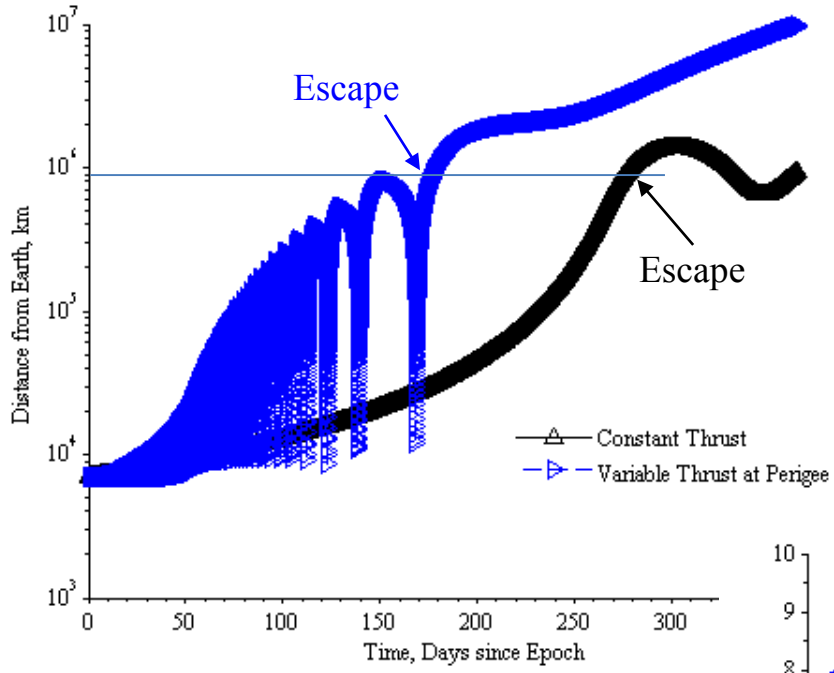


Case 2: Optimized Variable Thrust/Time at Perigee

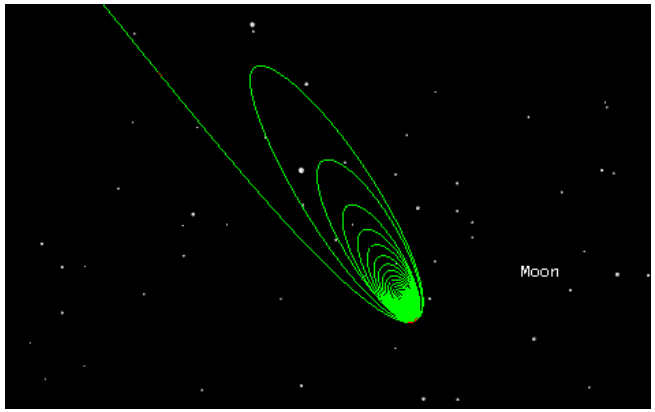
Red shows thrust/ Green shows cruise

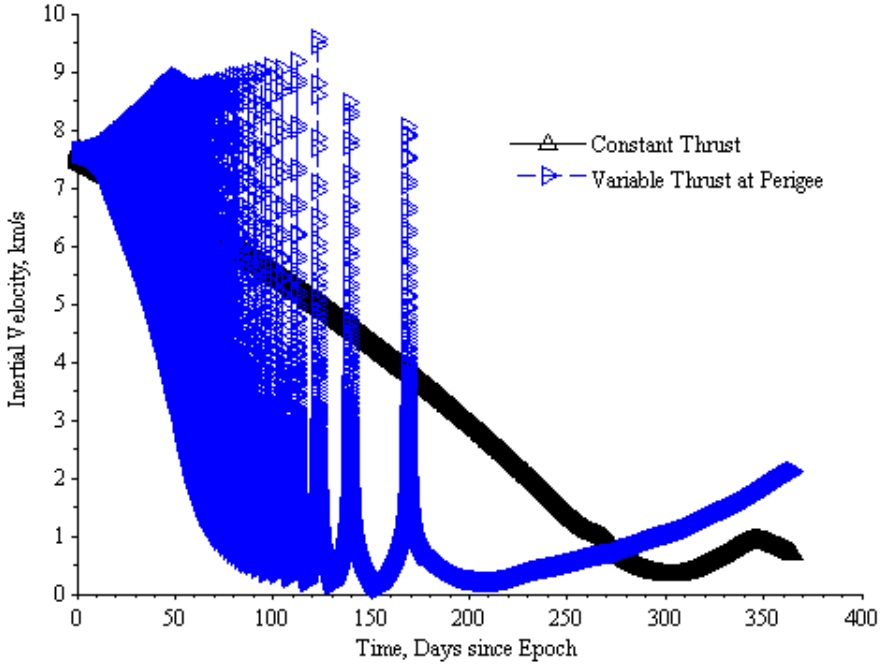
*SEET: Space Environment and Effects Tool

Case 1 and 2: Properties of Solutions

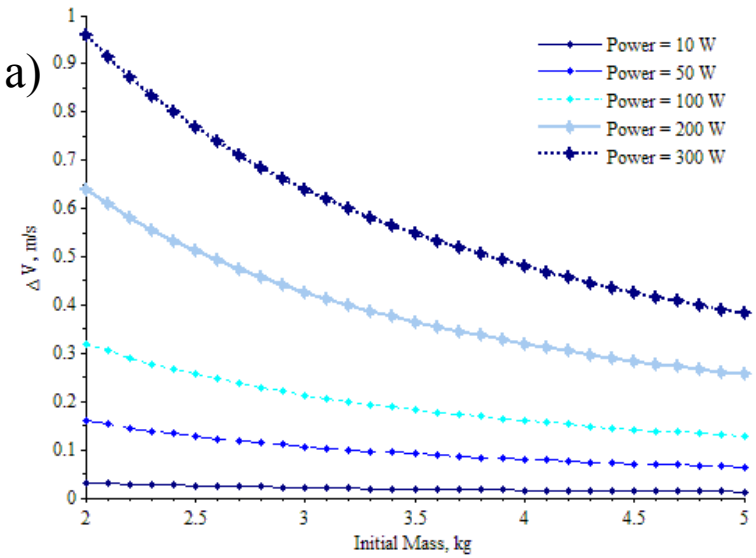


Different approaches for achieving Earth escape (apogee raise vs. apogee and perigee raise), which is more efficient (fuel + time + radiation exposure)



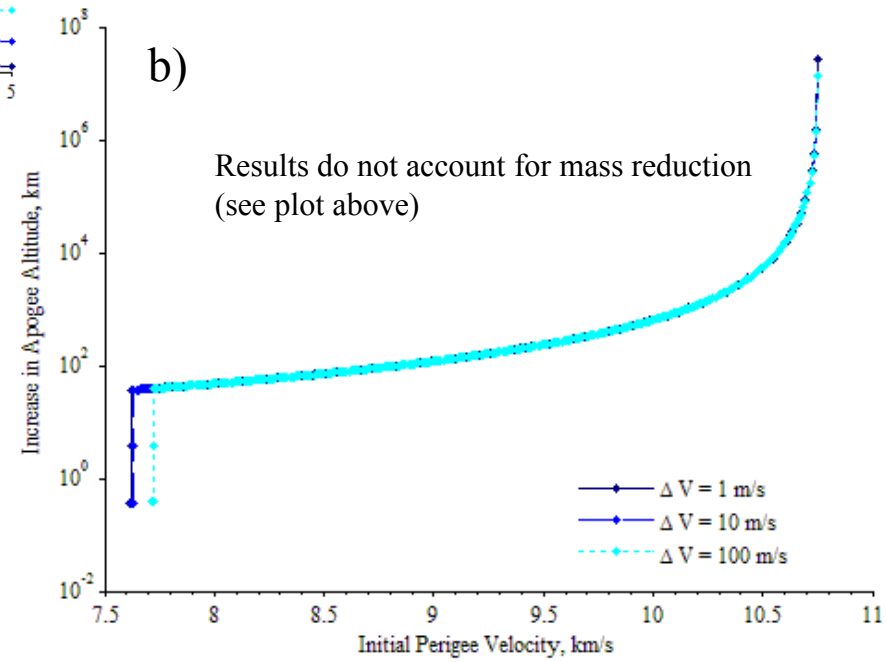


Case 2: Model Insights from Sensitivity Analysis



Greater ΔV with less initial mass (nearly twice the ΔV with half the mass, 2.5 vs 5 kg, for 300 W maneuver)

Greater apogee boost when perigee is greater (several orders of magnitude!)



Summary: Multiple advantages of thrust maneuvers "later" in mission, but must first get to higher altitudes!