Optimizing Orbit Transfer Time using Thrust and Attitude Control for a CubeSat with Interplanetary Applications

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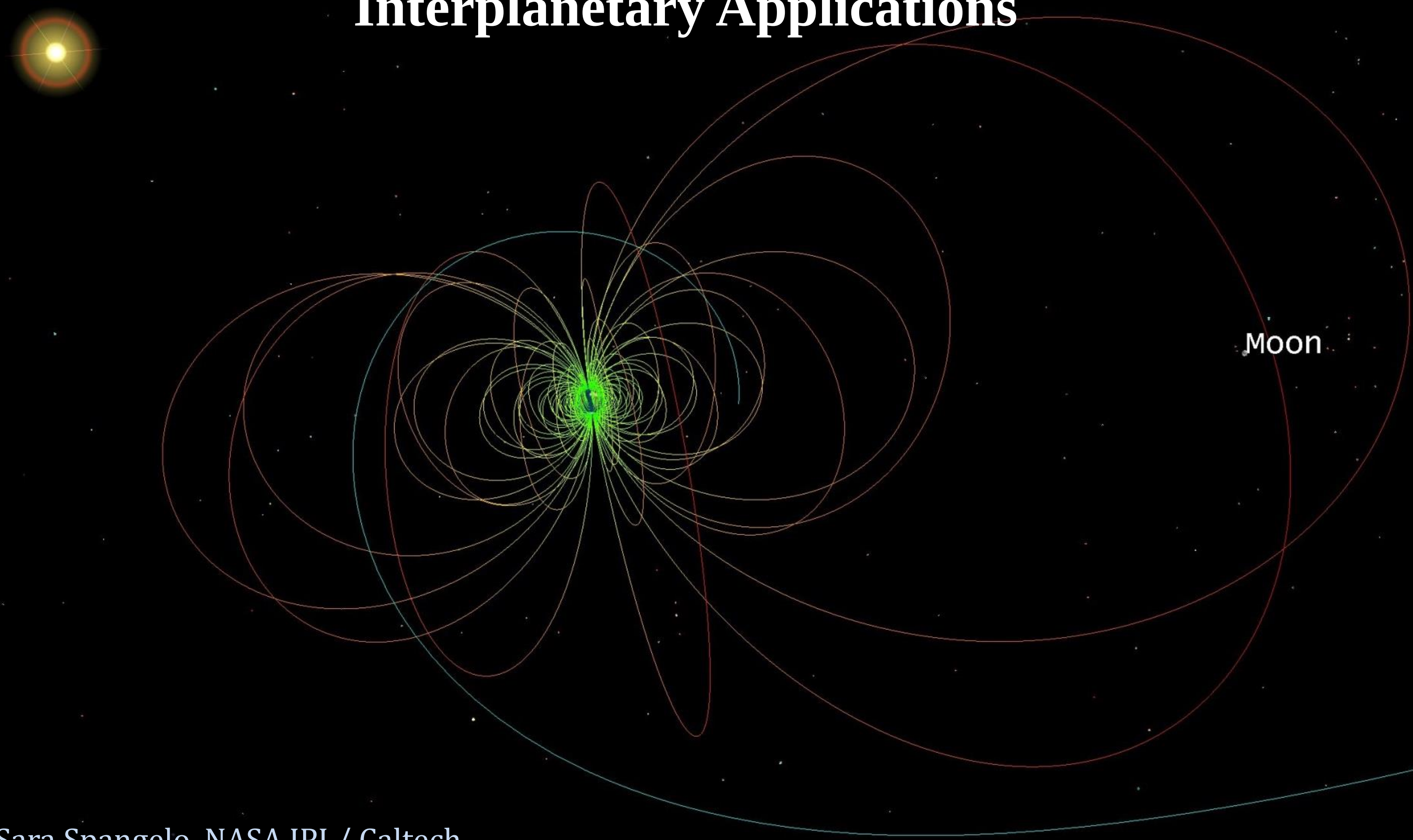

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Interplanetary Small Satellite Conference, April 2014

How Far Can CubeSats Go (Alone)?

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Motivation
Problem
Approach
Case 1
Case 2
Comparisons
Summary

- 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

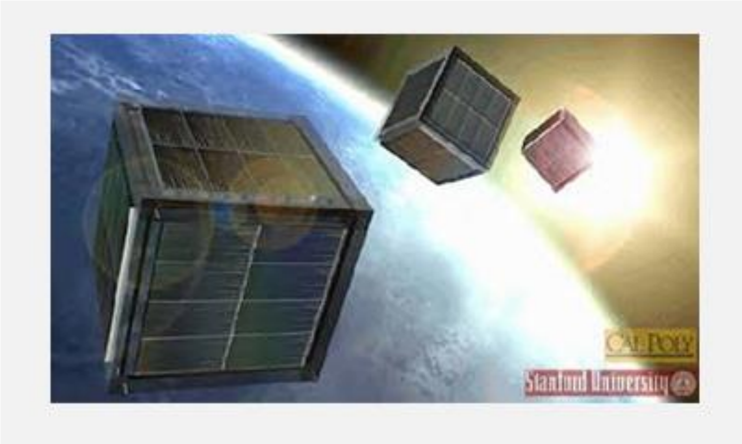


Photo Credit: NASA Website

Problem Objectives

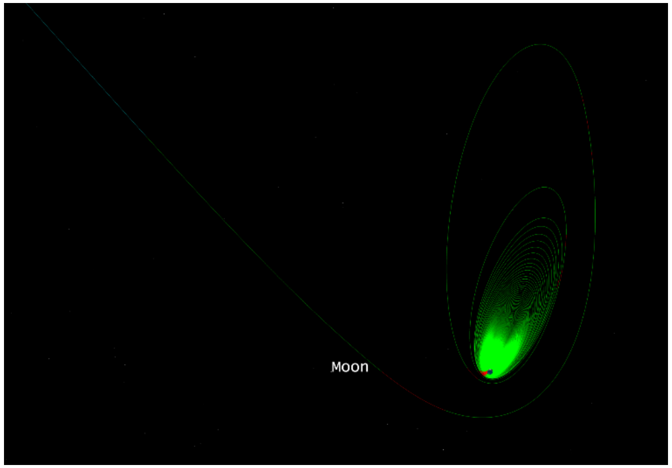
Last year we demonstrated the feasibility of escaping Earth orbit with the CubeSat Ambipolar Thruster (CAT) in a 3U CubeSat

This year we *optimize* trajectories and consider

- A variety of goals: minimizing time, fuel, volume, and radiation dosage
- Consider different maneuver schemes (i.e. spiral out, variable power/ time thrusts)
- Model energy balance (solar powered collection, eclipse)
- Model battery capacity (cycling, depth of discharge, degradation)

The goal of this work is to better understand the tradeoff between:

- The required fuel, batteries, and time to escape Earth orbit
- The risks with different schemes (i.e. due to battery cycling and radiation)



CAT: Large ΔV Engine Capability

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

Problem Assumptions

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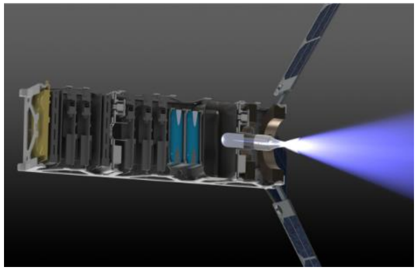
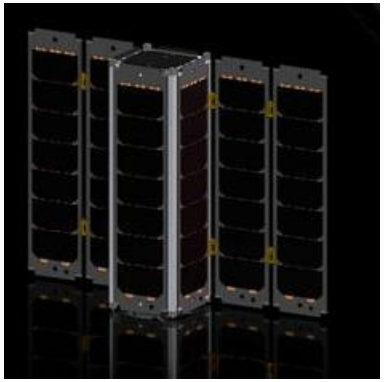
- CubeSat Ambipolar Thruster (CAT)
  - Mass: <0.5 kg, Volume: 0.1U
  - Iodine fuel (I_2), I_{sp}=1010 sec, Density=5 g/cm^3
  - Operating power levels: 3-300 W
  - 50-60% thruster efficiency
- 3U CubeSat Spacecraft Components
  - Blue Canyon XB1 Bus (GNC, C&DH, Telecom, Power, ACS)
    - Pointing: 7.2 arcsec accuracy, 1 arcsec stability, <2.5 kg, ~1 U, <2.5 W
  - Aluminum 3U CubeSat Structure
  - Deployable Solar Arrays (~30 W in sun)
  - Major subsystems (except fuel and batteries): ~3.3 kg, ~1.5 U (1.5 U remaining)
- Initial Orbit: 500 km circular, polar or near-polar
- Nominal operations (all but CAT): ~3 W

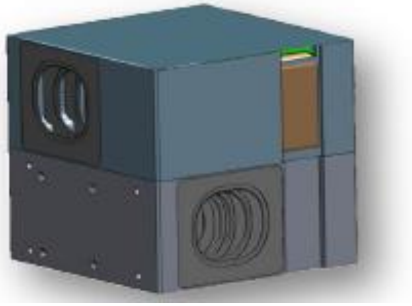
CAT engine with CubeSat subsystems





Clyde Space Double Deployed 2-Sided 30 W Solar Panels

ISIS 3U CubeSat Al Structure



XB1 Blue Canyon System

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

Multidisciplinary Approach

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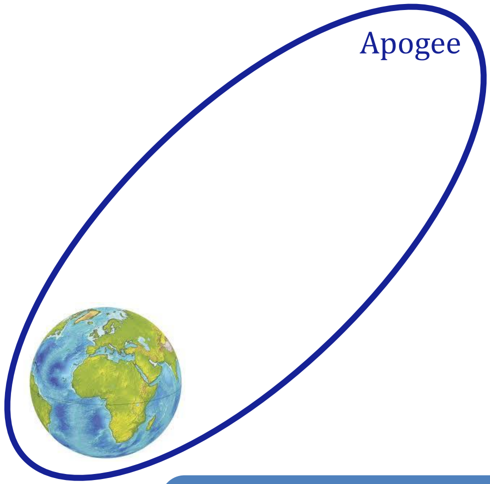
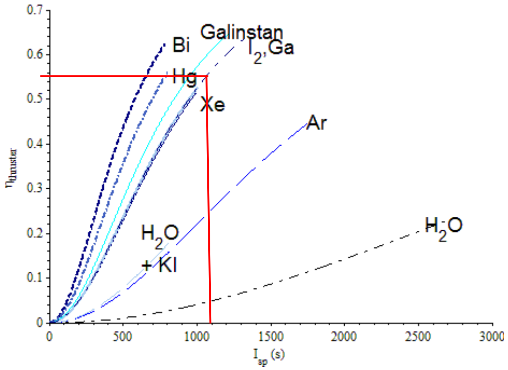
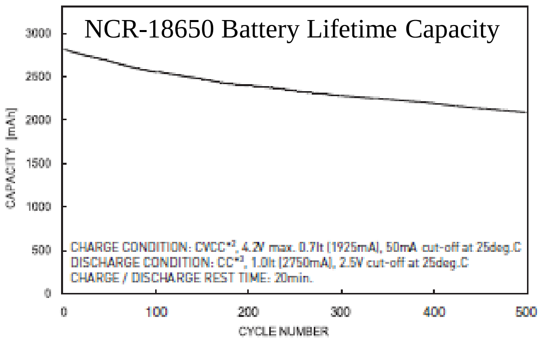


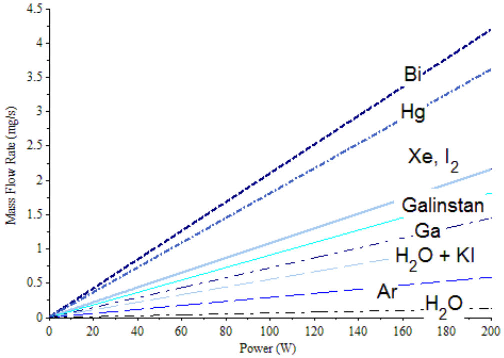
Orbit Design
- ΔV boosts orbit altitude
- Conservation of energy and angular momentum



Energy
- Battery Capacity
- Solar Power Collection
- Eclipse Management

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



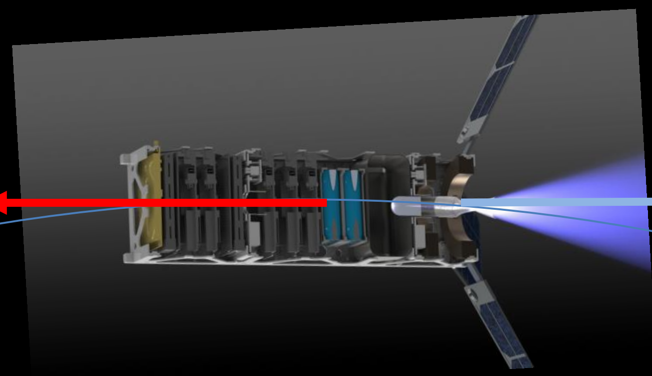


We also analyze attitude control and radiation, but not as part of the optimization problem

Case 1: Constant Thrusting in Velocity Direction

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

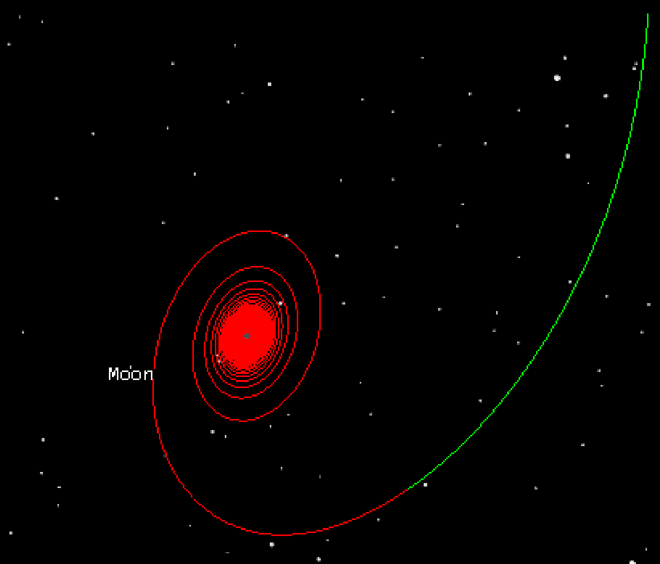
Velocity Vector
ΔV Direction



Thrust Vector

Orbit

Resulting spiral out trajectory



Red shows thrust/ Green shows cruise

Case 1: Constant Thrusting in Velocity Direction

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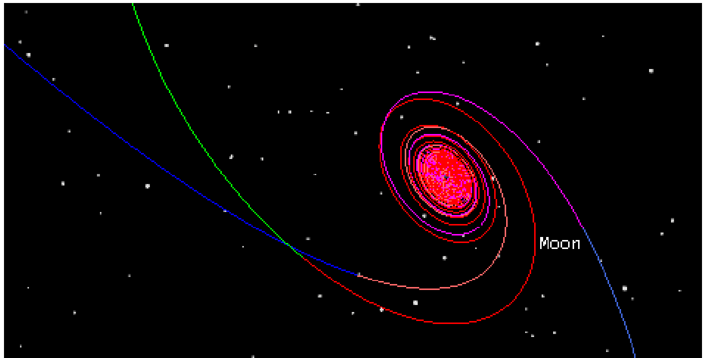
Results given for orbit starting in 500 W circular orbit until Earth-escape (925,000 km)

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

Case 1: Constant Thrusting for Sun Sync Orbits

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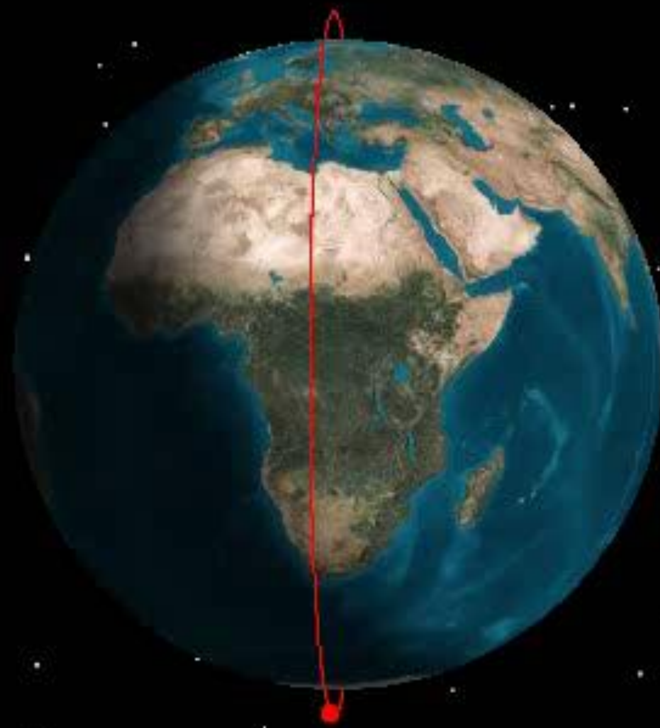
Starting from "worst-case" 500 km Sun Synchronous orbit ($\beta^{*}\approx 60^{o}$), with 25 W power setting, altitude is boosted such that there is no eclipse (>750 km) in <3 days!



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

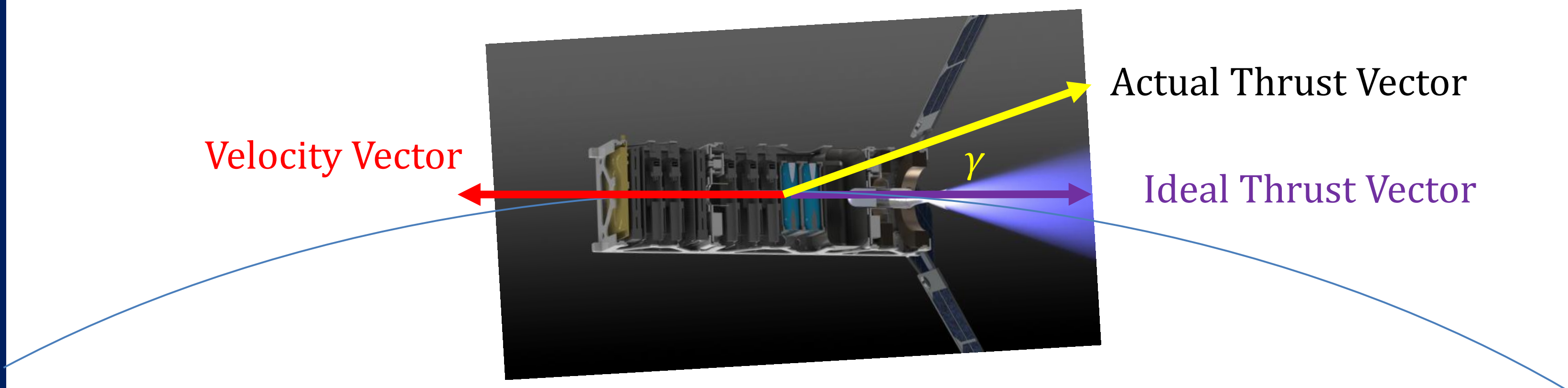
β^{*}: Angle between orbital plane and vector to Sun

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)

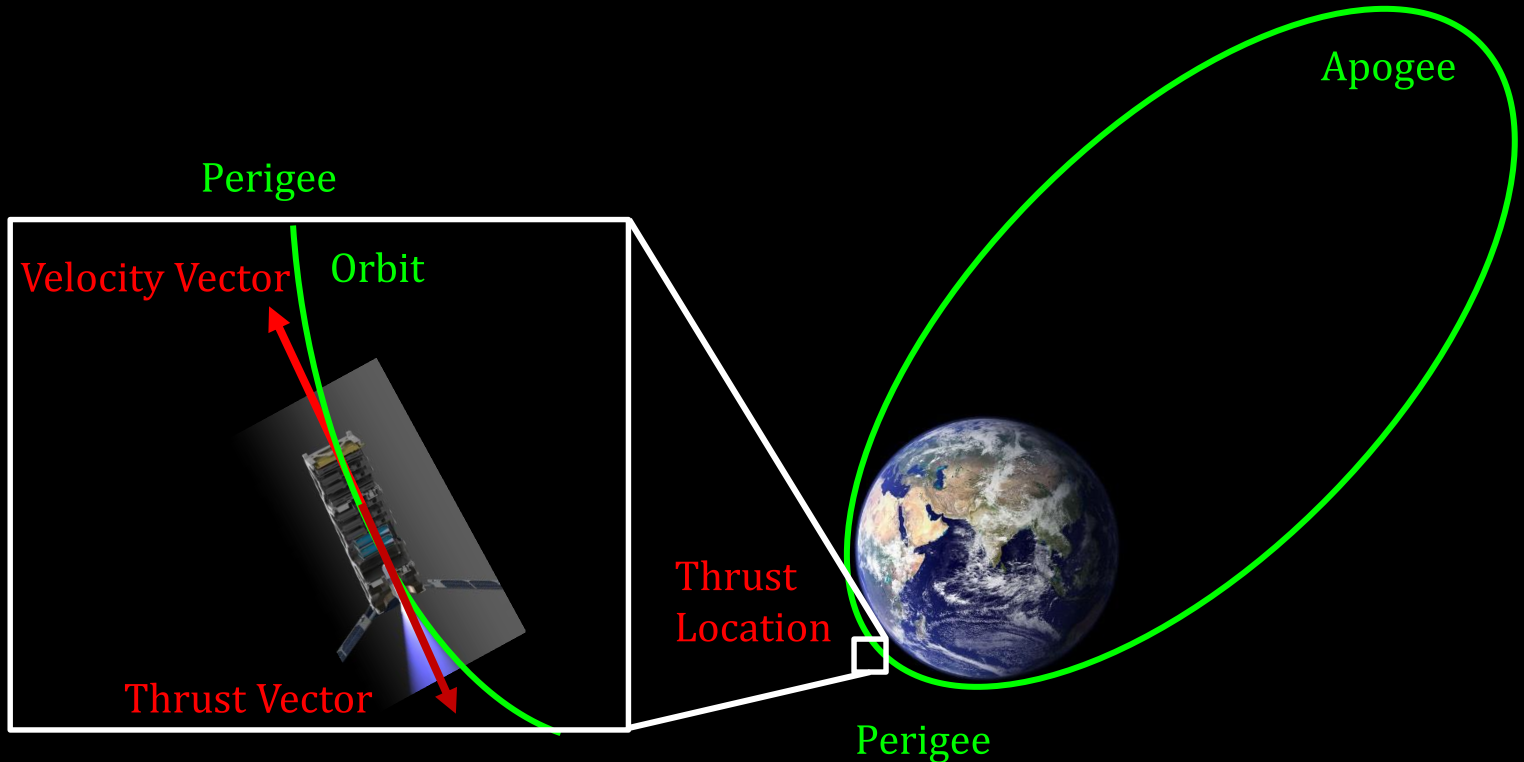
- Even with γ=20°, only requires an additional 13.1 days (10W)/ 5.2 days (25 W)
- Orbit shape and precession will also change with cross-track ΔV component

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%



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).

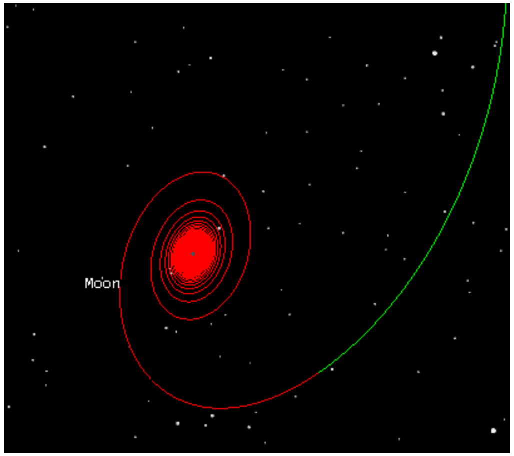
Case 2: Optimization Problem Formulation

Goal: Find the trajectory to minimize time to escape Earth's sphere of influence (radius: 925,000 km) by thrusting for a short time once per orbit, centered at perigee.

Decisions: Power setting and duration during each maneuver (once per orbit at perigee)

Constraints:

- Thruster power settings (2-300 W)
- Battery capacity constrains available energy
  - Battery depth of discharge, battery degradation
- Positive energy balance throughout each orbit
- Maximum volume for fuel + batteries = 1.5U
  - Ideally want to minimize volume



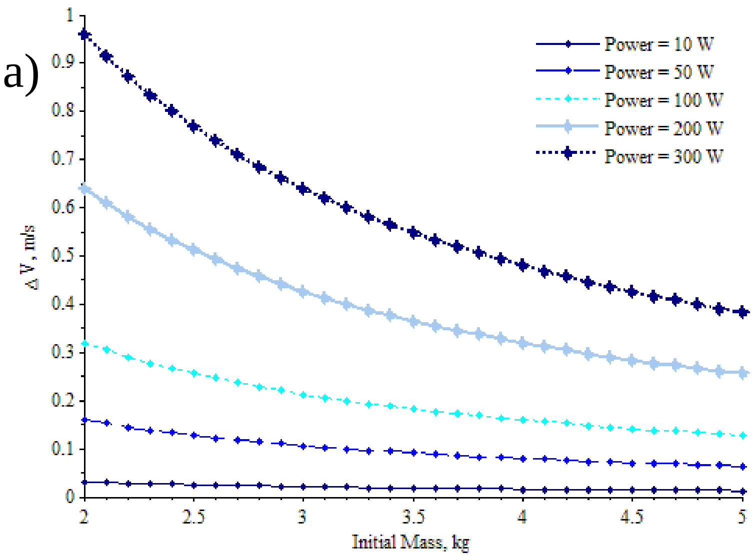
Dynamics:

- 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 apogee boost
- Energy: Model realistic collection and consumption (solar, nominal, thrust)

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

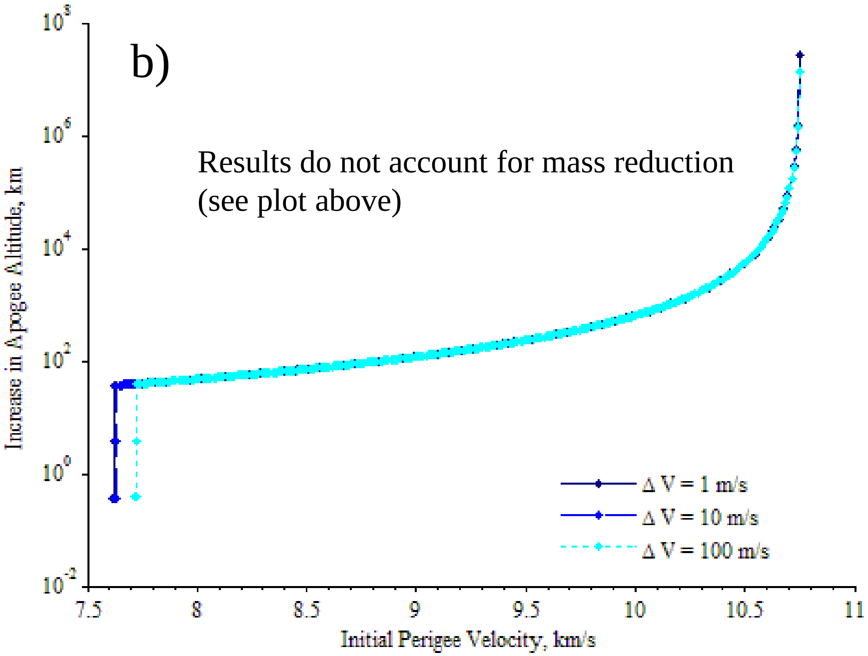
Case 2: Model Insights from Sensitivity Analysis

a)



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

b)



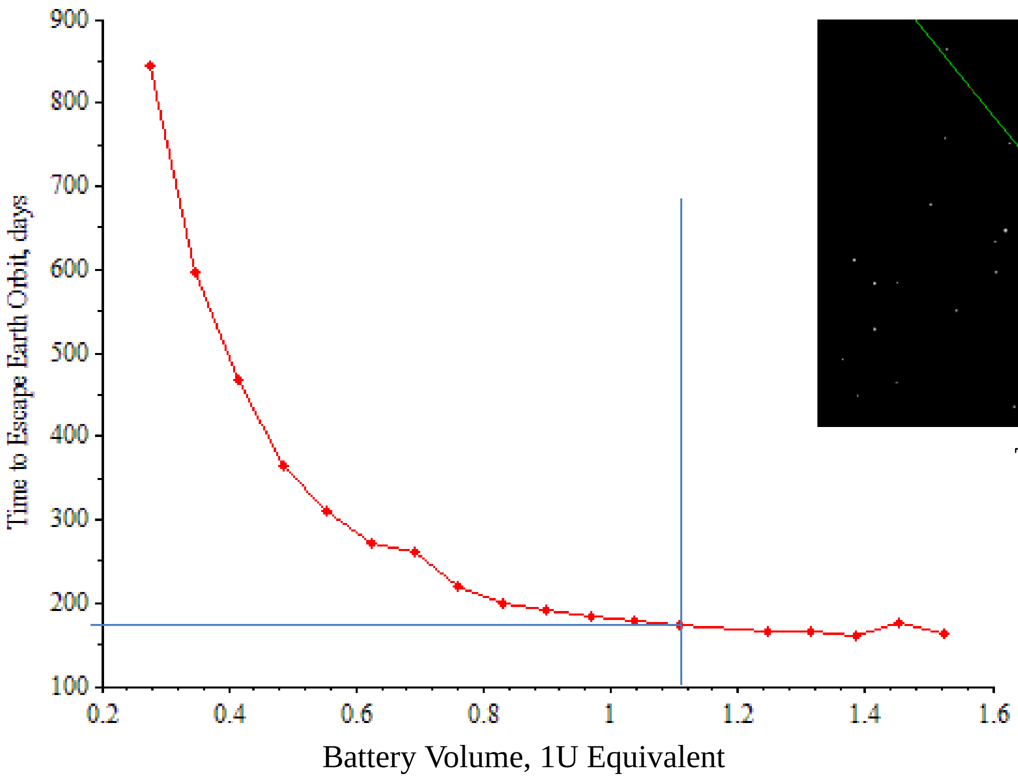
Results do not account for mass reduction (see plot above)

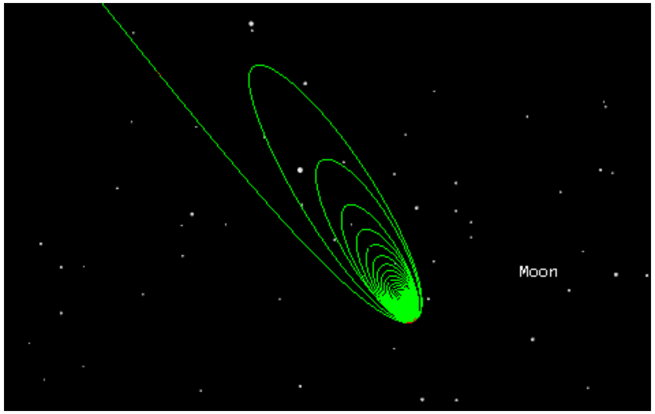
Greater apogee boost when perigee velocity is greater (several orders of magnitude!)

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

Case 2: Optimal Solutions

- Problem solved using MATLAB's *fmincon* and **feasible** initial conditions
- Minimum time to escape Earth orbit
    - Optimal ~174 days: require ~1.15 U volume (~30 batteries)
    - All solutions require 1.34 kg fuel ~ 0.27 U (~1.4 U total for fuel + batteries)
    - All solutions require 720-740 battery cycles
- Comparison to Case 1 (10 W): 35% reduction in time, and 46% reduction in fuel
- Comparison to Case 1 (25 W, sun sync): 62% longer, and 46% reduction in fuel





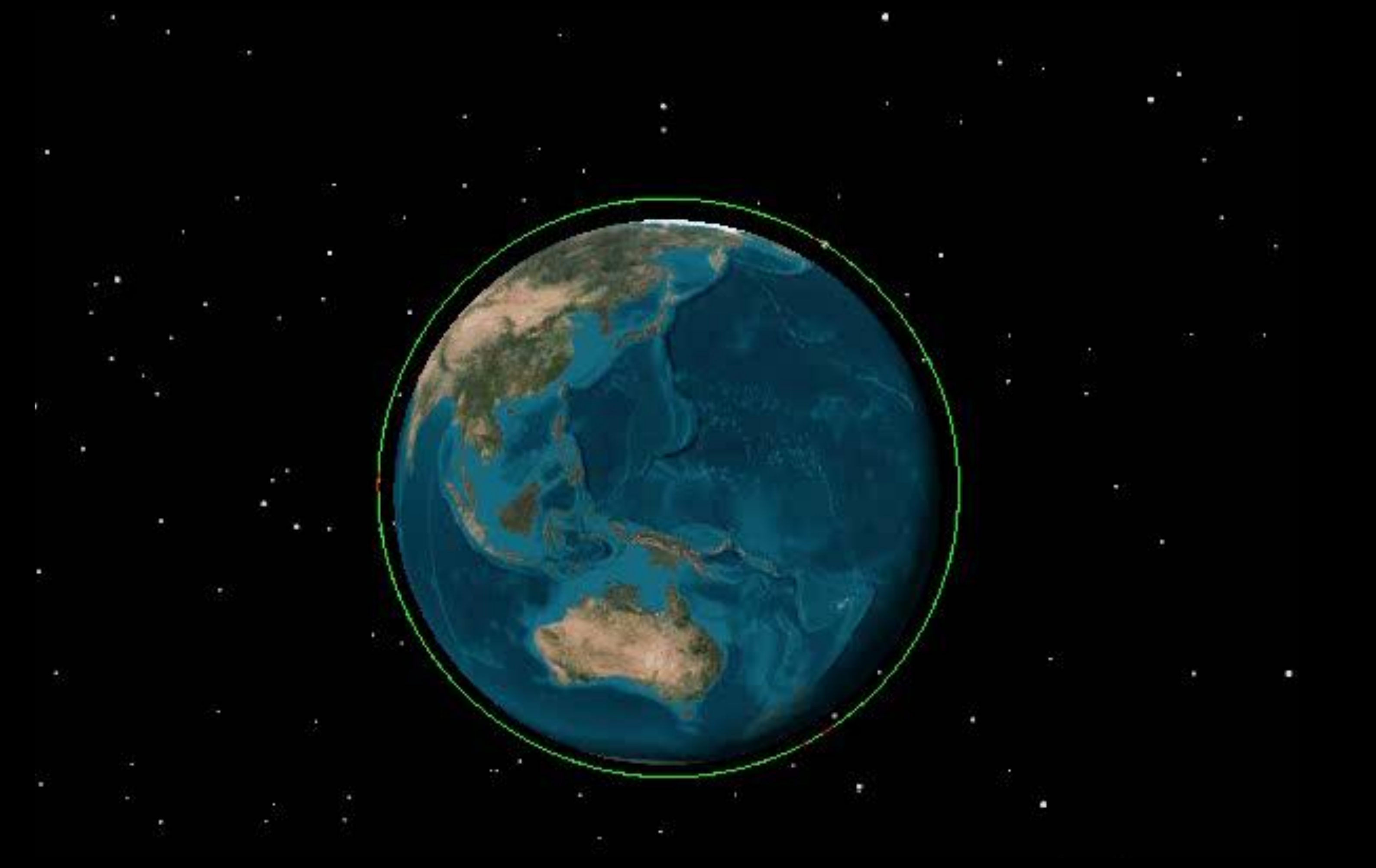
Trajectory Visualization

4x18650 Li-Ion batteries: 160 cm^3, 0.16 1U Equivalent

Case 2: Verification & Visualization

Solutions verified in System's Tool Kit (STK) with Astrogator Tool

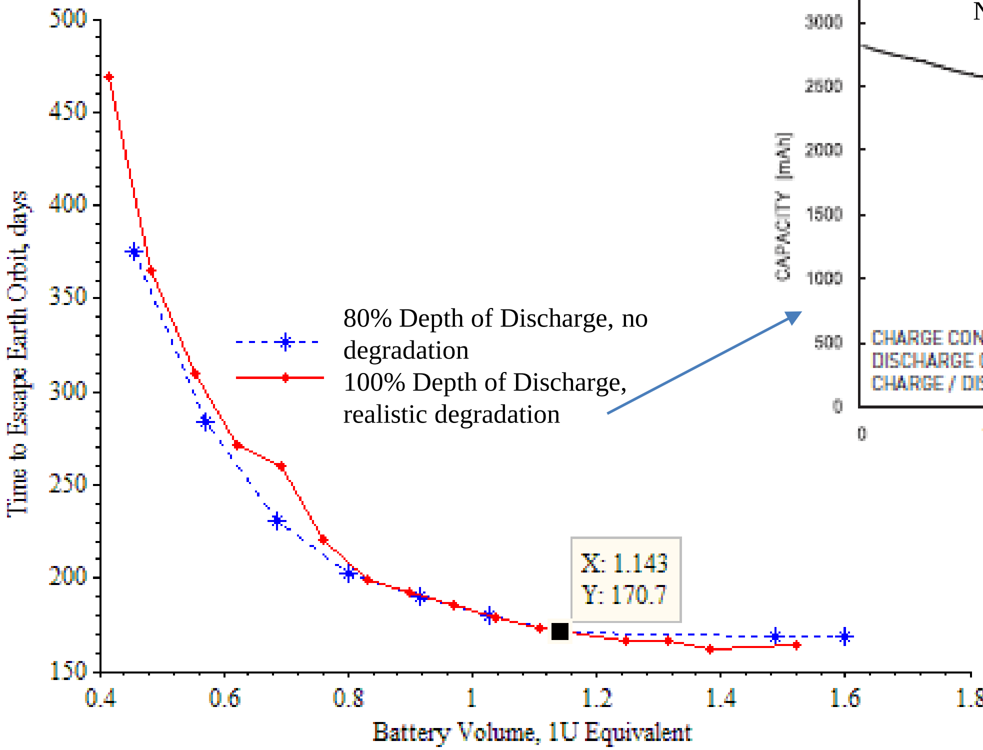
- Results similar for Case 1 (10 W): Earth-escape in ~178 days with 1.6 kg fuel

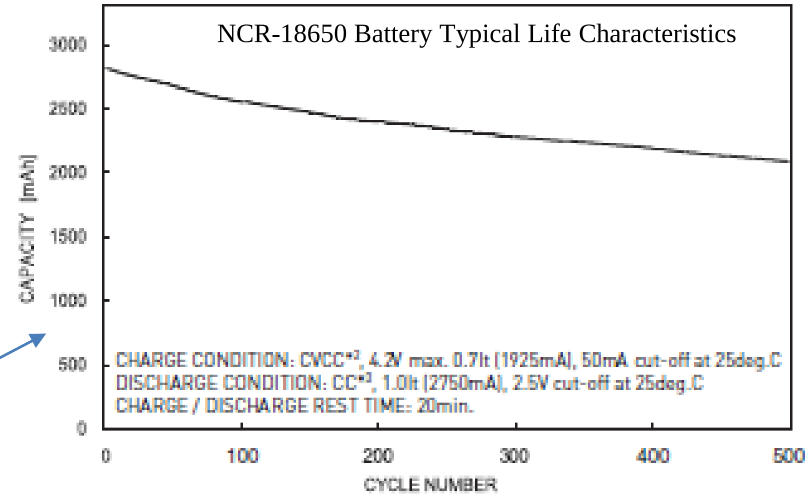


Red shows thrust/ Green shows cruise

Case 2: Realistic Battery Degradation

- In reality, 18650 Li-Ion batteries degrade with increased number of cycles
- Case 3: Same optimization approach as Case 2 with realistic battery degradation





- Optimal solution: ~171 days, ~30 batteries (~1.15 U), ~1.36 kg fuel required, 171
  - Nearly identical to Case 1 with constant thrust
- Solutions don’t deplete 100% battery capacity every cycle (degradation conservative)

Reference: Panasonic Spec Sheet

Case 1 and 2: Radiation Effects

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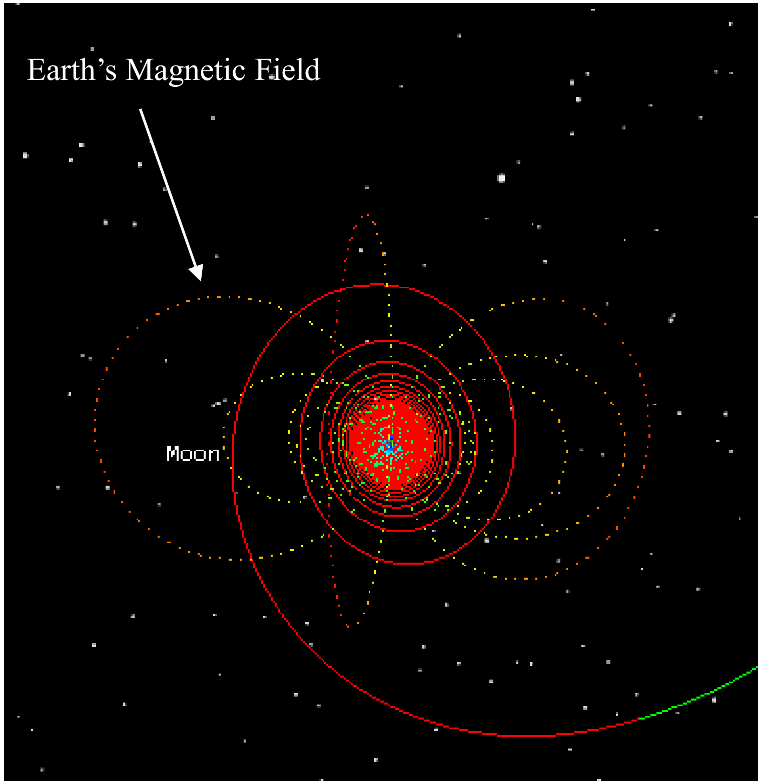
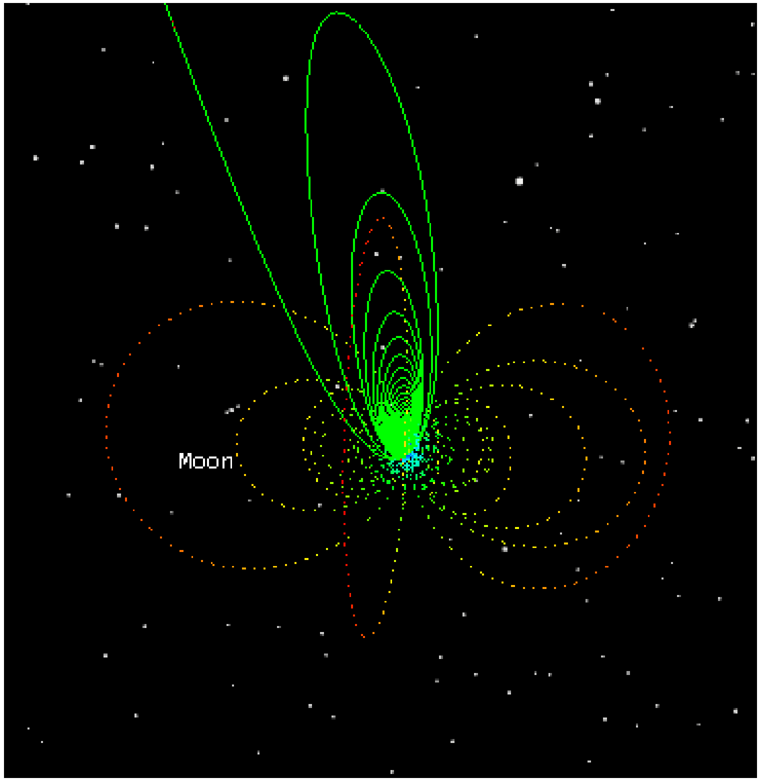
- Total Accumulated Radiation Dosage measured with STK's SEET*
- Case 1 spends more time in radiation belts than Case 1



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



Case 2: Optimized Variable Thrust/Time at Perigee

Red shows thrust/ Green shows cruise

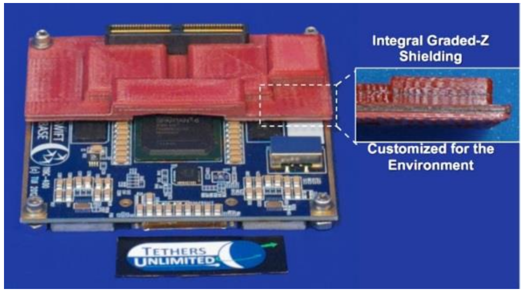
*SEET: Space Environment and Effects Tool

Case 1 and 2: Radiation Effects and Mitigation

Accumulated radiation dosage after one year mission (starting 500 km orbit)

Aluminum shielding thickness	**Case and Description**		
	Case 1: Constant Thrust (10 W)	Case 1: Constant Thrust (25 W)	Case 2: Variable Thrust at Perigee
82.5 Mils (2.1 mm)	30.0 krad	12.1 krad	8.9 krad
232.5 Mils (5.9 mm)	3.9 krad	2.0 krad	1.0 krad
457.5 Mils (11.6 mm)	2.2 krad	1.1 krad	0.6 krad
Predicted Al thickness for Accumulated Dosage <5 krad*	5.5 mm	4.5 mm	3.5 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 Unlimited's Radiation Shielding

ISIS 3U CubeSat Al Structure

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

Comparison of All Cases

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Solutions represent “best” of each case that is energetically feasible, ~1.15 U of batteries

Parameter	Case 1	Case 2
Maneuver Approach	Thrust continuously in velocity direction	Variable thrust magnitude and duration at perigee
Battery Modeling	80% DoD, no degradation	80% DoD, no degradation/ 100% DoD, realistic degradation per cycle
Thrust Power Level	10 W (sun sync)/ 25 W	Variable (50-300 W)
Thrust Time	Constant	Variable (1-60 min)
Earth Escape Fuel	2.50 kg	1.34 kg/ 1.36 kg
Fuel & Battery* Volume (1.5 U available)	2.5 kg, 0.5 U	2.84 kg, 1.4 U
Earth Escape Time	269 days/ 108 days	175 days/ 171 days
Number of Orbits	1322/ 545	720/ 701
Total Accumulated Ionizing Dose with 232.5 Mils (5.9 mm) after 1 year	3.90 krad/ 1.96 krad	1.03 krad

*Additive relative to those in XB1 CubeSat Bus (25 Whr)

Optimal Solutions for Different Goals

Cases start in 500 km circular orbit until they escape Earth's SOI (975,000 km)

Optimization Goal	If you can achieve sun sync orbit	If you can't achieve sun sync orbit
Minimize Time	Case 1: Thrust continuously in velocity direction (25 W)- **108 days**	Case 2: Variable thrust magnitude and duration at perigee- **175 days**
Minimize Fuel	Case 2: Variable thrust magnitude and duration at perigee- **1.34 kg**	
Minimize Fuel & Battery Mass/ Volume	Case 1: Thrust continuously in velocity direction (11 W)- **2.5 kg/ 0.5 U**	Case 1: Thrust continuously in velocity direction (25 W)- **2.5 kg/ 0.5 U**
Minimize Radiation	Case 2: Variable thrust magnitude and duration at perigee- **1.03 krad**	

Summary:

- Case 2 is appealing for time-optimal solutions (when not sun-sync)
- Case 2 is attractive to reduce fuel and reduce radiation exposure
- Case 1 always requires less total battery mass and volume

Additional Challenges and Future Work

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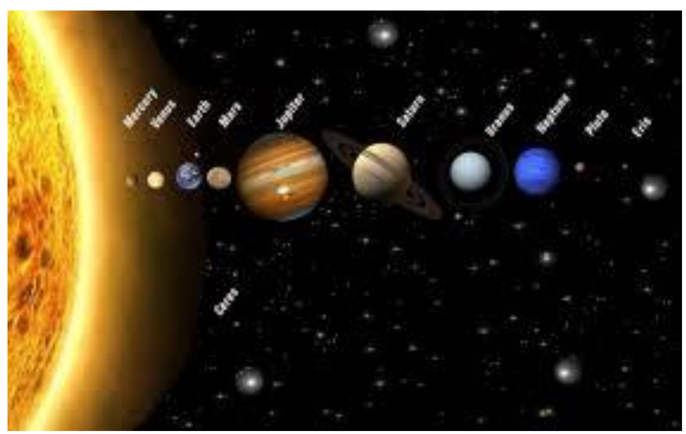
Additional Mission Design Challenges

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

Future Work

- Model power, thermal, radiation, and attitude control in optimization problem
- 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?

Work through Rocket Equation for I_2 fuel and CAT

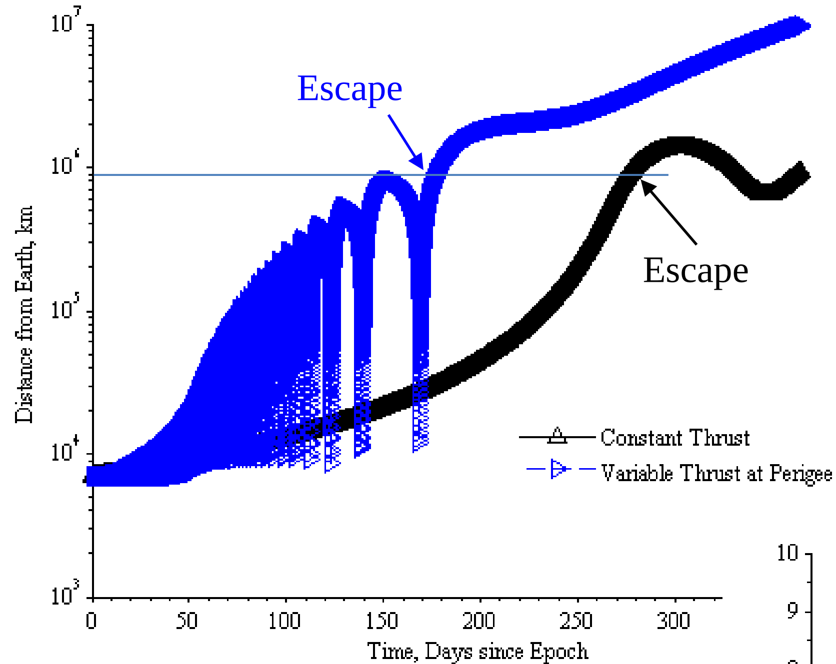
Parameter	Symbol	Input/ Equation	Value	Units
Specific Impulse	I_{sp}	Input	1010	sec
Exhaust Velocity	V_{exh}	$V_{exh} = I_{sp} \cdot \mathrm{g}$	9908	km/sec
Dry Spacecraft Mass	m_s	Input	2.5	kg
Propellant Mass	m_p	Input	2.5	kg
Initial Mass	m_i	$m_i = m_s + m_p$	5.0	kg
Final Mass	m_f	$m_f = m_s$	2.5	kg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

Ideal Rocket Equation

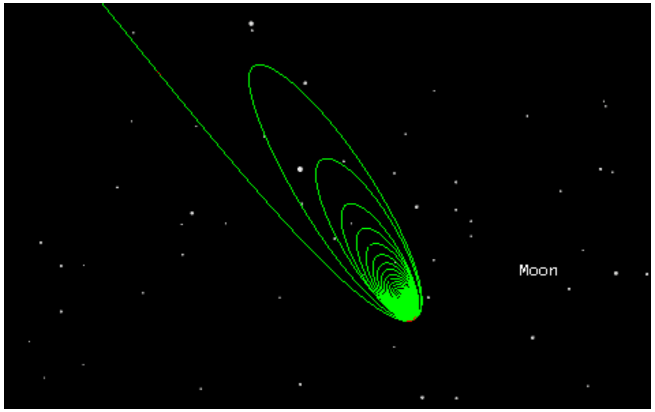
We can escape Earth's Sphere of Influence (ΔV~ 7km/sec) with ~2.5 kg of fuel!

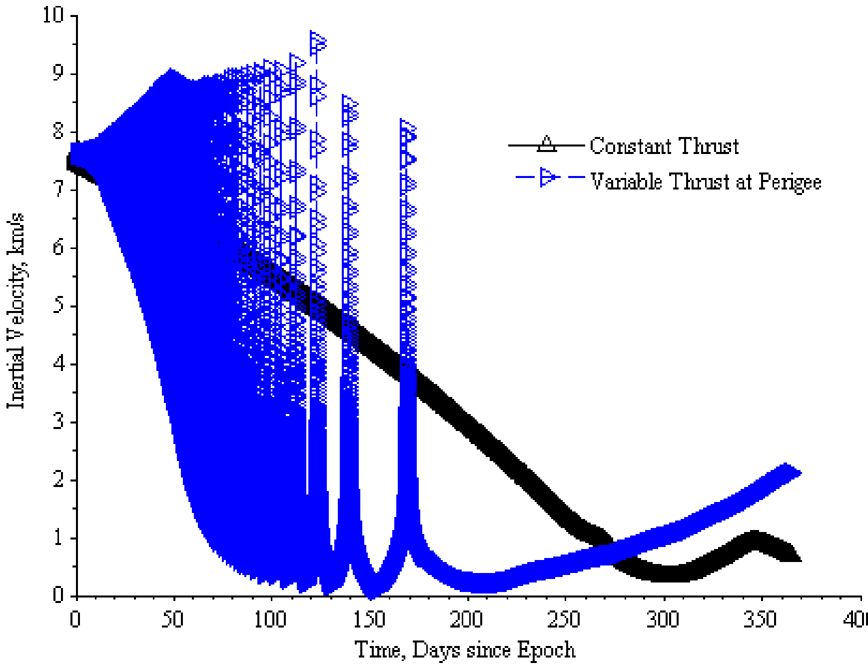
g: gravity constant = 9.81 m/sec^2

Case 1 and 2: Properties of Solutions



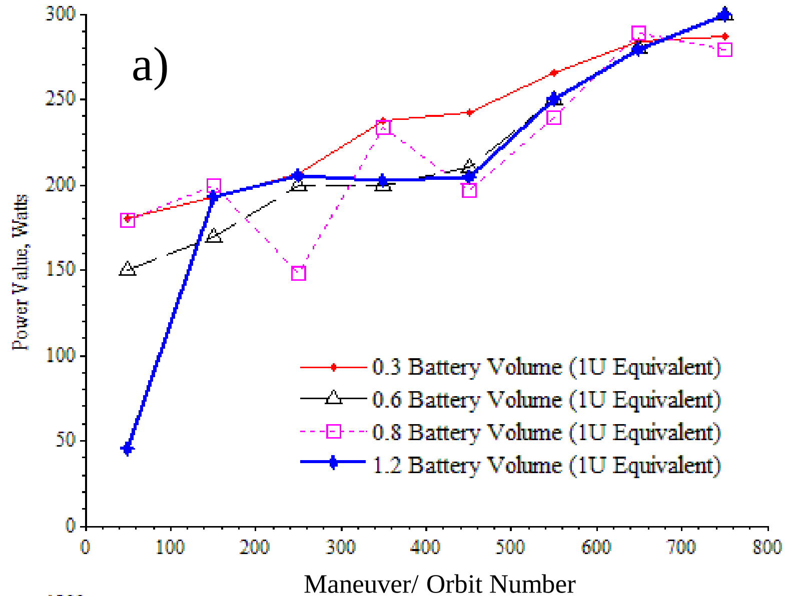
Different approaches for achieving Earth escape (apogee raise vs. apogee and perigee raise), which is more efficient (fuel + time + radiation exposure)

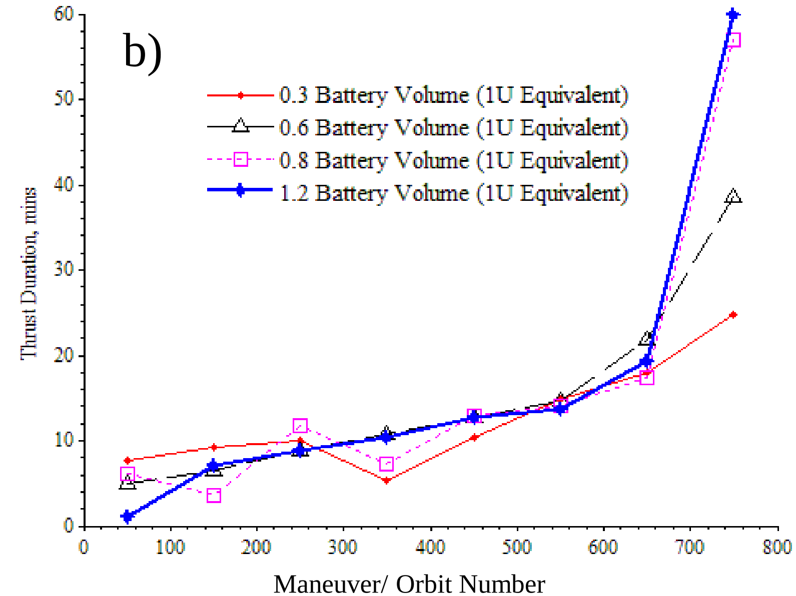


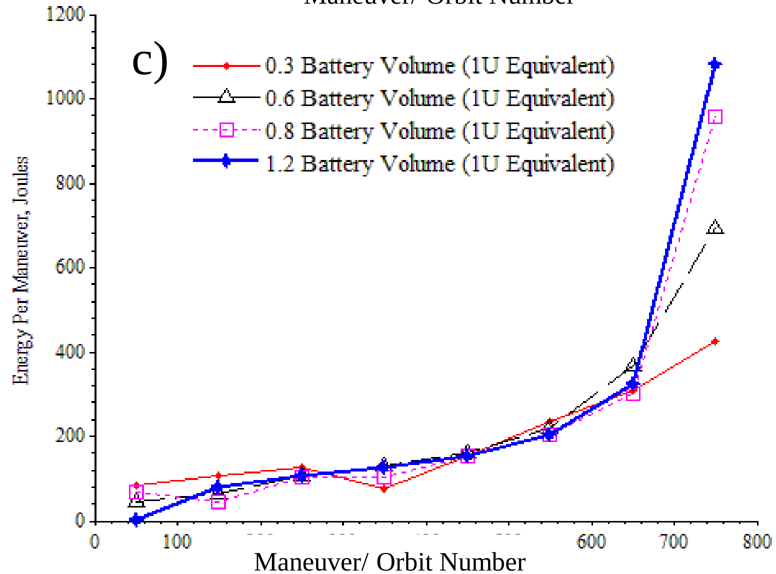


Case 2: Properties of Optimal Solutions

Representative power values and thrust durations from optimal solutions.

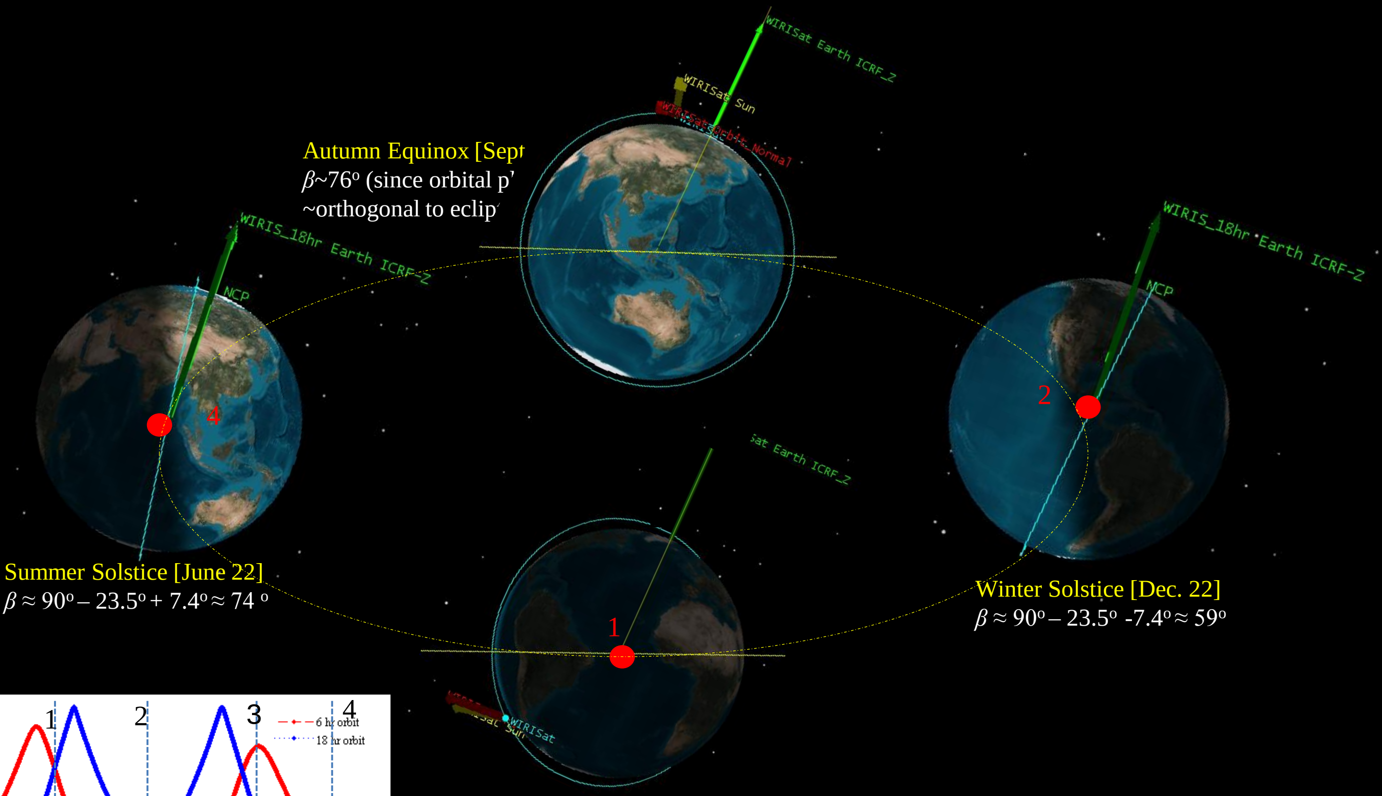






As maneuver/ orbit number increases→
orbit apogee increases →
eclipse fraction decreases →
higher energy maneuvers possible →
greater apogee boosts to approach escape

Annual Solar Variation

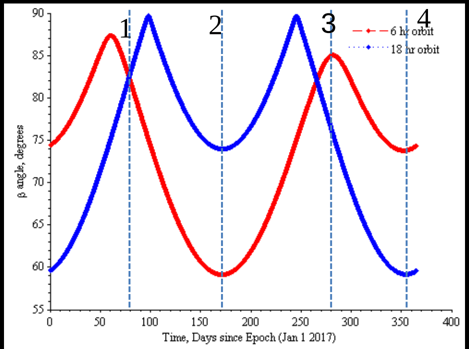


Autumn Equinox [Sept
β~76º (since orbital p
~orthogonal to eclip

Summer Solstice [June 22]
$\beta \approx 90^o - 23.5^o + 7.4^o \approx 74^o$

Winter Solstice [Dec. 22]
$\beta \approx 90^o - 23.5^o - 7.4^o \approx 59^o$

Vernal Equinox [March 21]
β~83º (since orbital plane
~orthogonal to ecliptic)



Local Time of
the Descending
Node (LTDN)
= 18 hrs