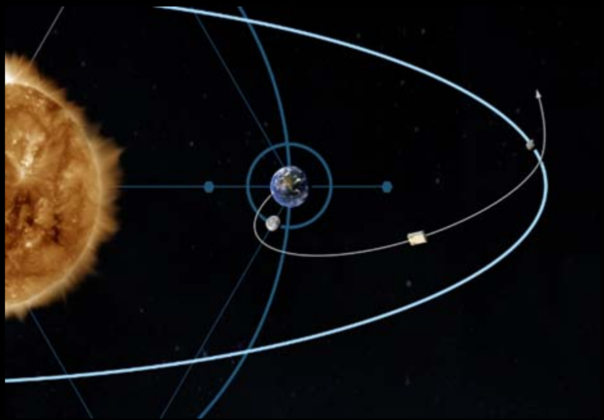
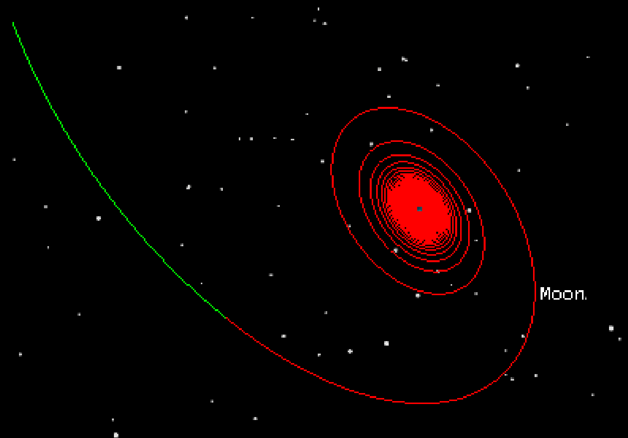
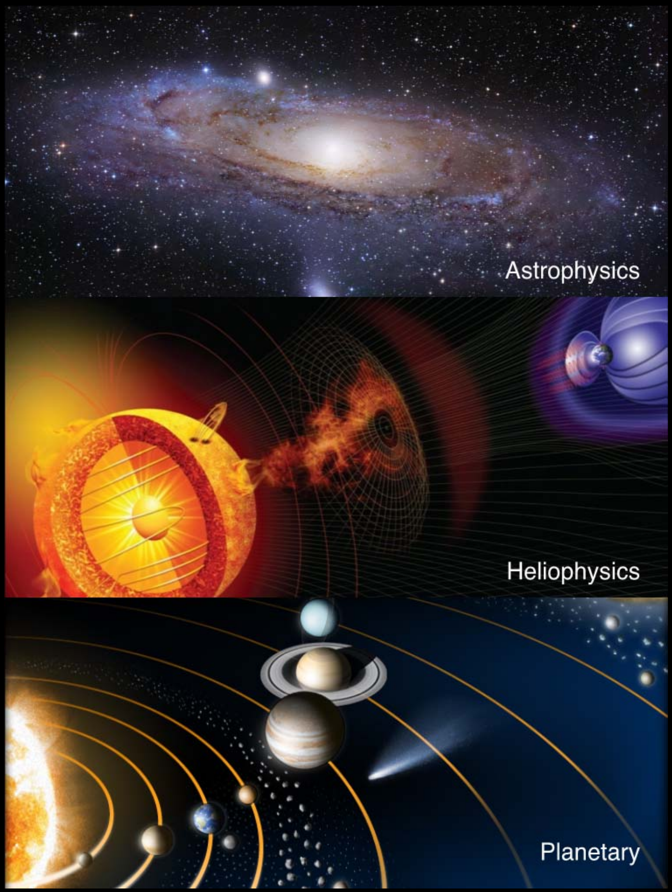
Integrated Vehicle and Trajectory Design of Small Spacecraft with Electric Propulsion for Earth and Interplanetary Missions

Small Satellite Conference 2015







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Ben Longmier, University of Michigan; Derek Dalle, NASA/Ames
Small Satellite Conference, August 2015, Logan, Utah

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

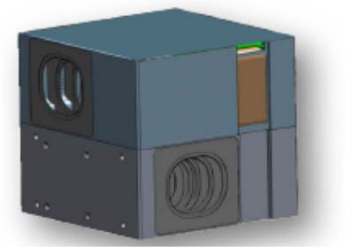
Future: Several emerging EP solutions for CubeSats
Game-changing and enabling/enhancing a broad class of missions:

- 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: Provide systems-level perspective of different small satellite electric propulsion technologies capabilities and key trade-offs

Heritage and Enabling Technology

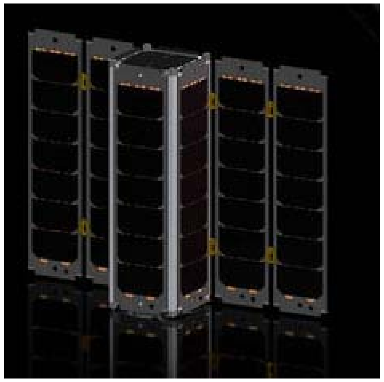
- Significant flight experience and heritage in LEO and high-TRL components
- Telecommunication and Navigation systems
  - High-rate X/Ka-Band radios (10+ Mbps in LEO)
  - Iris Transponder (JPL) and high gain antennas
- High-accuracy attitude control technology
  - Blue Canyon's XACT: 7.2 arcsec accuracy, 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

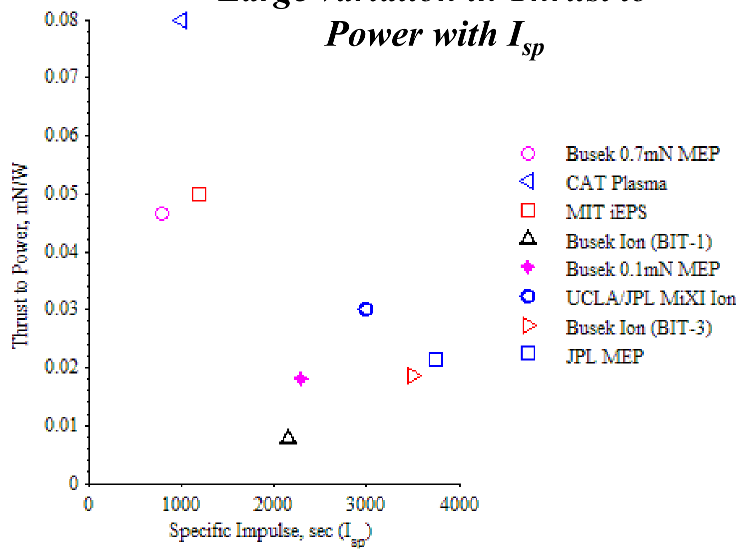
Image Credit: Clyde Space, ISIS, Blue Canyon, MMA

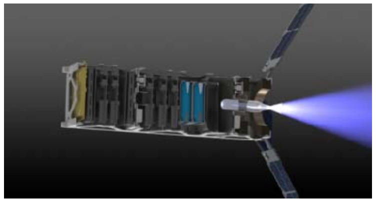
Overview of Emerging Small Spacecraft EP Systems

Thruster* (Point Design)	Technology	I_{sp}	Thrust	System Power
Units		sec	mN	W
Busek's 0.1 mN MEP	Electrospray	800	0.1	5.5
CAT Plasma	Magnetoplasma	1010	10	125
Busek's 0.1 mN MEP	Electrospray	2300	0.7	15
MIT iEPS	Electrospray	2000	0.1	2
Busek's Ion (BIT-1)	Ion	2150	0.1	13
MiXI Ion	MiXI Ion	3000	1.5	50
Busek's Ion (BIT-3)	Ion	3500	1.4	75
JPL's MEP	Electrospray	3744	0.16	8.2

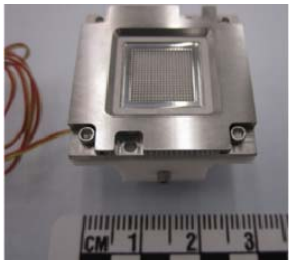
*Thruster specs based on publically available information

Large variation in Thrust to Power with I_{sp}





UMich/Aether's CubeSat Ambipolar Thruster (CAT)

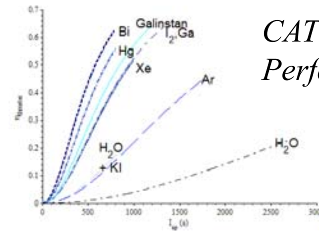


JPL's Indium MEP Thruster

Multidisciplinary Systems Modeling Approach

Propulsion Dynamics

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



CAT Thruster Performance

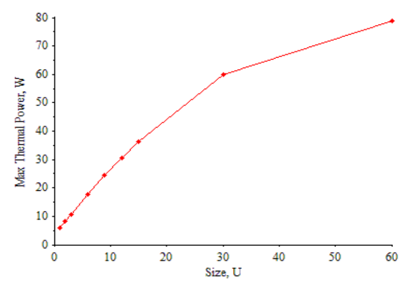
Orbit Dynamics

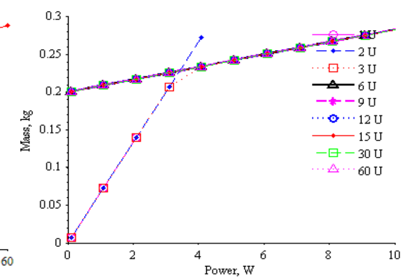
$$\dot{v}_r = \frac{v_\theta^2}{r} - \frac{\mu}{r^2} + a \sin\gamma,$$
$$\dot{v}_\theta = a\cos\gamma - v_r \theta,$$
$$\dot{\theta} = \frac{v_\theta}{r},$$
$$\gamma = \tan^{-1}\left(\frac{v_r}{v_\theta}\right),$$
$$a = -\frac{V_{ex}\dot{m}}{m}$$

$$v = \sqrt{v_r^2 + v_\theta^2},$$
$$h = v_\theta r,$$
$$e = \sqrt{\frac{(2\mu - rv^2) r v_r^2 + (\mu - rv^2)^2}{\mu}},$$
$$r_a = \frac{h^2}{\mu(1-e)},$$
$$r_p = \frac{h^2}{\mu(1+e)}.$$

Spacecraft Mass, Power, Volume Sizing

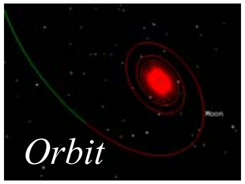
- Power
- Thermal
- Bus
- Telecom
- Payload



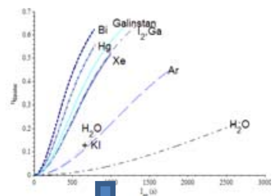


S. Spangelo and B. Longmier, "Optimization of CubeSat System-level Design and Propulsion Systems for Earth-Escape Missions", Journal of Spacecraft and Rockets, accepted December 2014.

Multidisciplinary Systems Modeling Approach



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

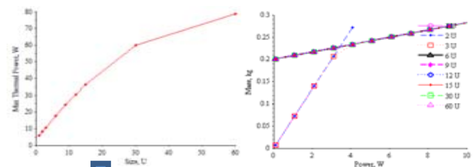


Trajectory
(ΔV, time, thrust)

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.

Thermal and Power Model



Compute Power,
\# Thrusters,
Propellant Mass

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)

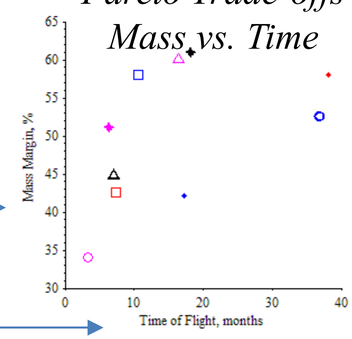
Size Solar Arrays
Check Feasibility of Thermal and Power Systems

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

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

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

Pareto Trade-offs
Mass vs. Time



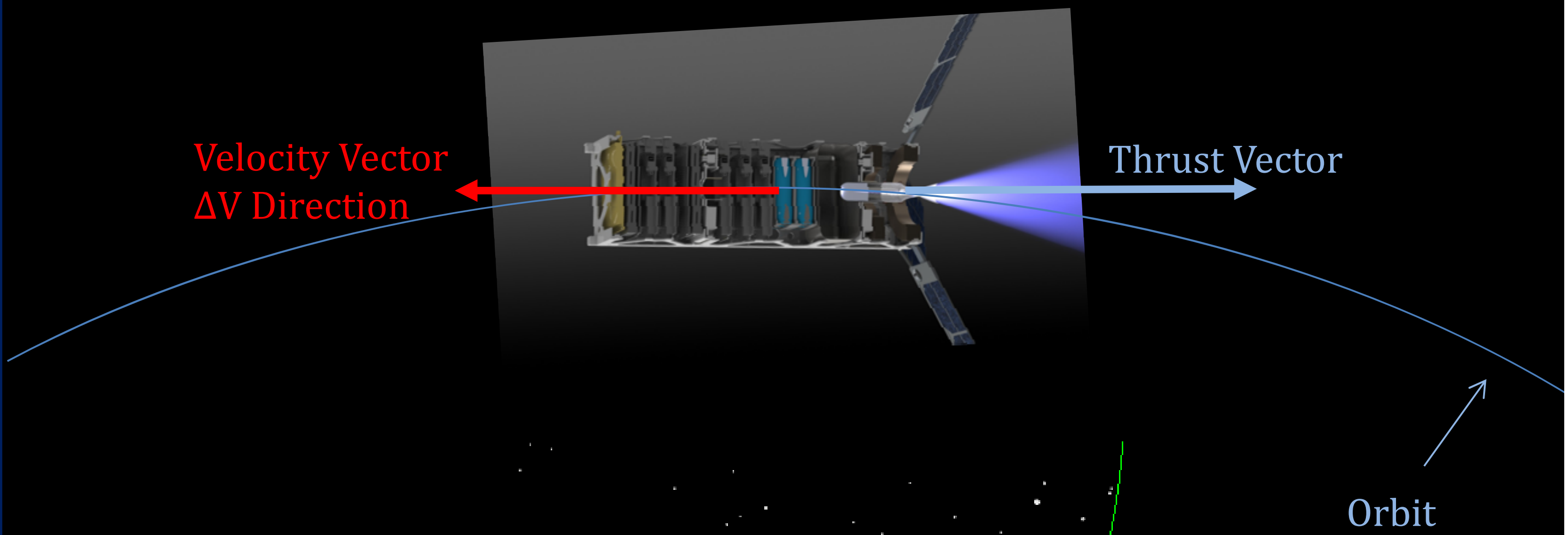
Check Total Mass Feasibility and Compute Margins

Thrust, I_{sp}, propellant

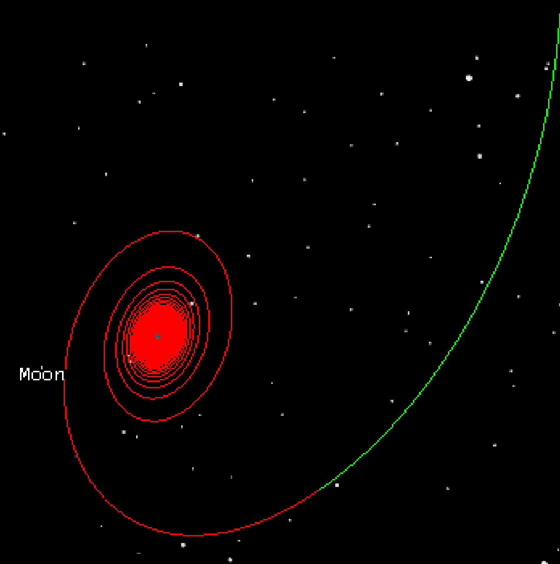
Modeling approach from: S. Spangelo, D. Landau, N. Aurora, S. Johnson, T. Randolph, “Defining the Optimal Requirements for the Micro Electric Propulsion Systems for Small Spacecraft Applications", Journal of Spacecraft and Rockets, Under Review.

Constant Thrusting in Velocity Direction

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



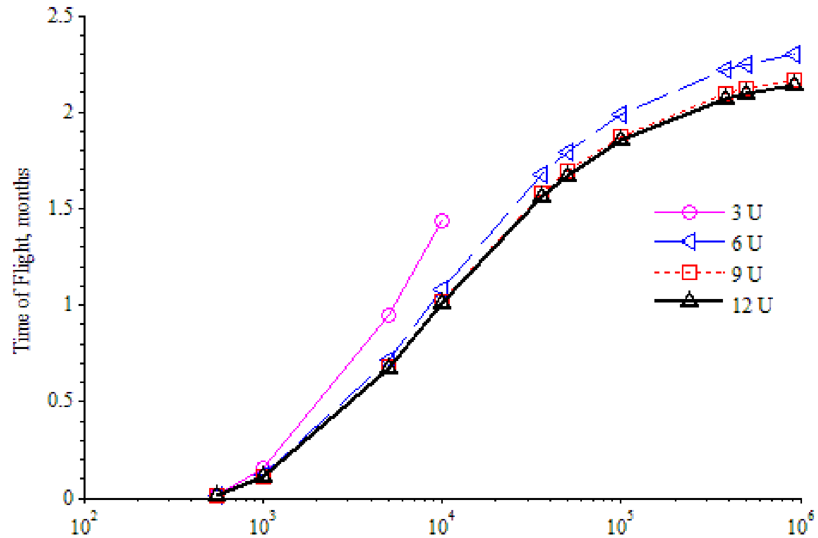
Resulting spiral out trajectory

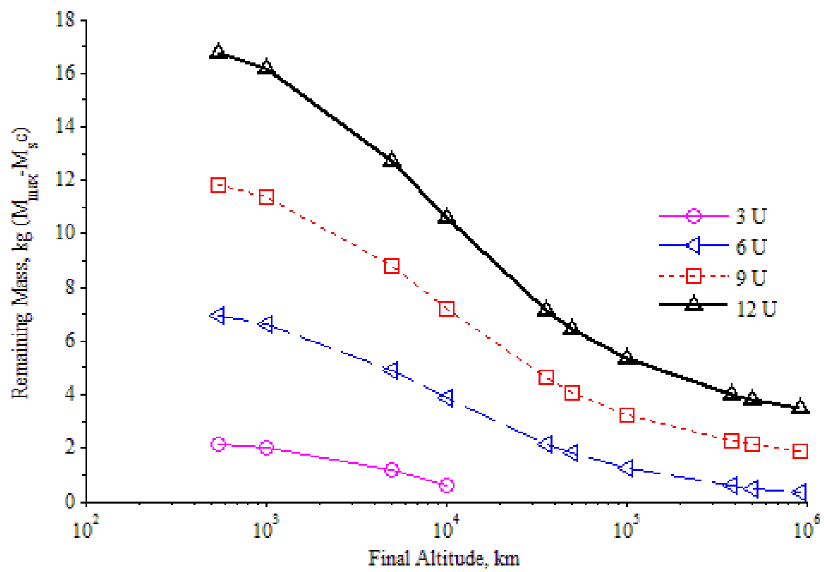


Red shows thrust/ Green shows cruise

Results: Altitude Orbital Transfers in LEO

Orbital transfers starting from 500 km circular orbits.





Parameter	Maximum LEO	GEO	Mean Moon	Earth's SOI
A (km)	2,000	35,700	384,000	919,000
ΔV (km/s)	0.6	4.4	6.5	6.9

*SOI: Sphere of Influence

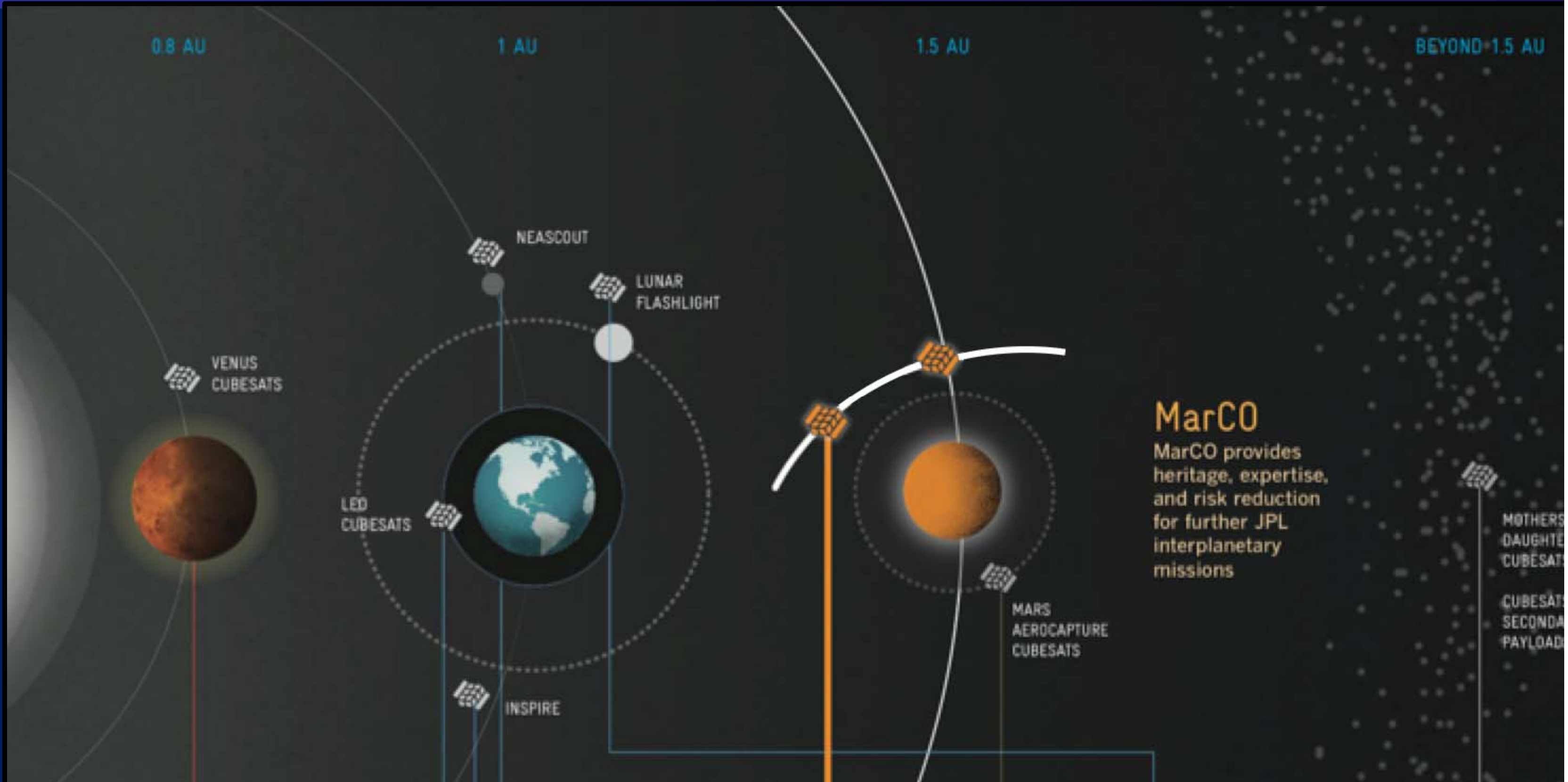
Flight times improve with spacecraft size as more thrusters can be accommodated (power, thermal)

SHOW feasible instruments

Total payload masses improve with spacecraft size as spacecraft bus grow smaller than max size

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

Interplanetary Targets for Future Missions



Destination	Venus	Moon	Mercury	Mars
Distance	0.72 AU	384 K km	0.39 AU	0.52 AU
Available Solar Power at Target (relative to 1 AU)	193%	100%	670%	44%

Approach: Interplanetary Transfers and Flybys

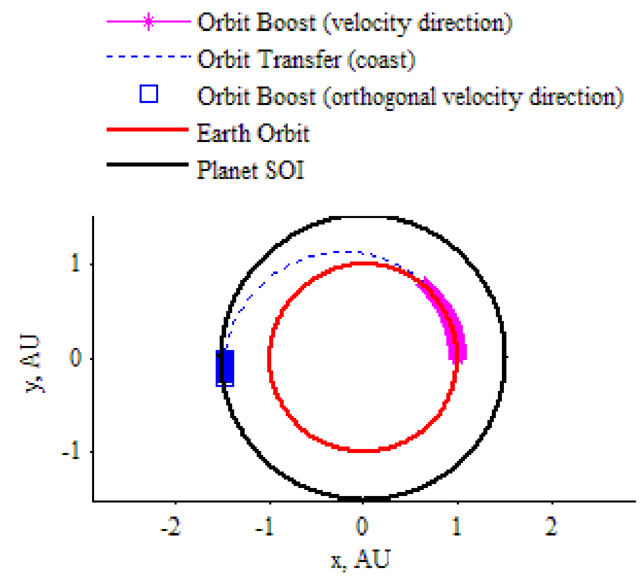
Phases to achieve flyby:

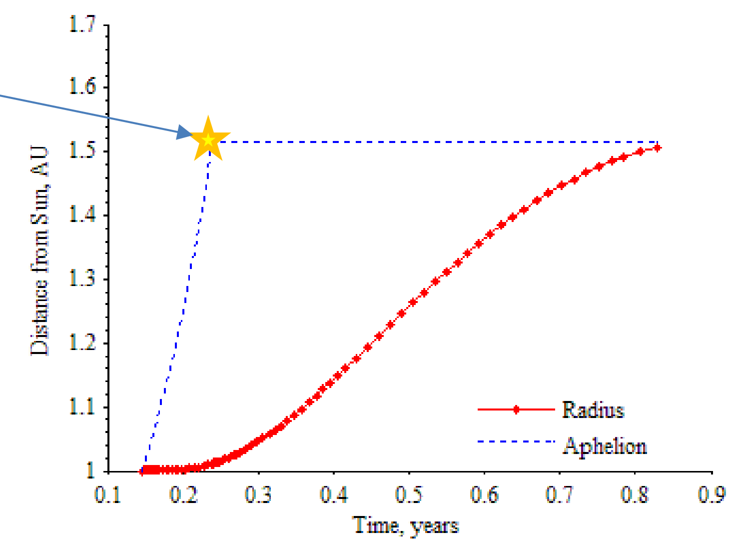
1. *Initalization*: Start trajectory in circular GEO.

2. *Earth-Escape*: Thrust in velocity direction until reach Moon/ escape Earth's SOI*.

3. *Orbit Boost*: Thrust in velocity direction until aphelion is equal to the distance to the plant from the Sun.

4. *Cruise Phase*: No thrusting until performs flyby….



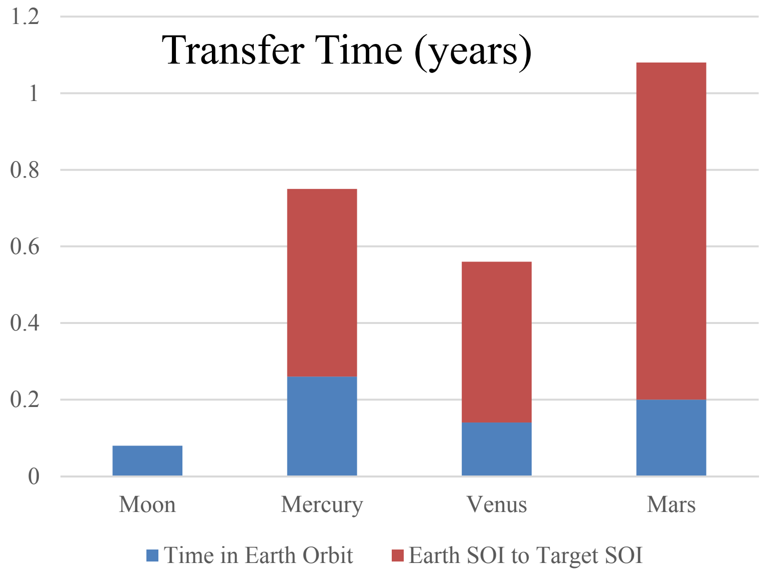


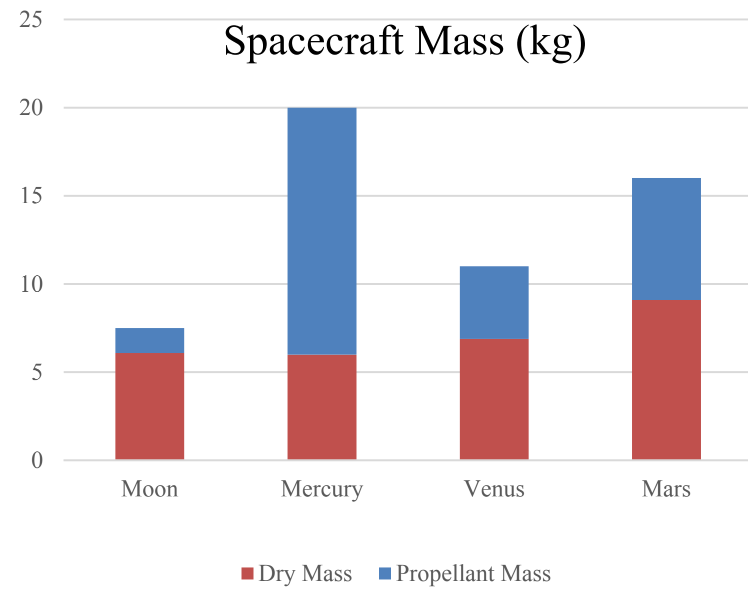
*SOI: Sphere of Influence (for Earth, radius: 925,000 km)

Example Mars Transfer (16 kg, 100 W)

Results: Interplanetary Flybys

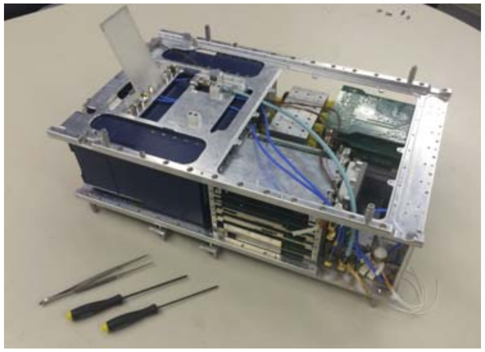
Flybys from GEO to all planets in less than one year in a <20 kg CubeSat!





Vehicle design based on other JPL deep space CubeSats (INSPIRE, MarCO, etc.)

- Deployable solar arrays, batteries
- BCT XB1 Bus (C&DH, ADCS, EPS, etc.)
- Structure, reaction wheels scale with size
- Iris transponder (tracking & communication)
- Total 6U dry mass ~ 6 kg



MarCO 6U CubeSat (JPL)

Summary & Future Work

Summary

- Systems-level framework for evaluating diverse thruster technologies
- Integrated trajectory and design decisions, inputs, constraints, objectives
- Showed trade-offs/ sensitivities for performance metrics (mass, volume, time) for Earth orbit altitude changes
- Designed feasible vehicles and trajectories for interplanetary flybys/ captures

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.