Traveling the Solar System Using Ion Engines,
An Engineer's Perspective

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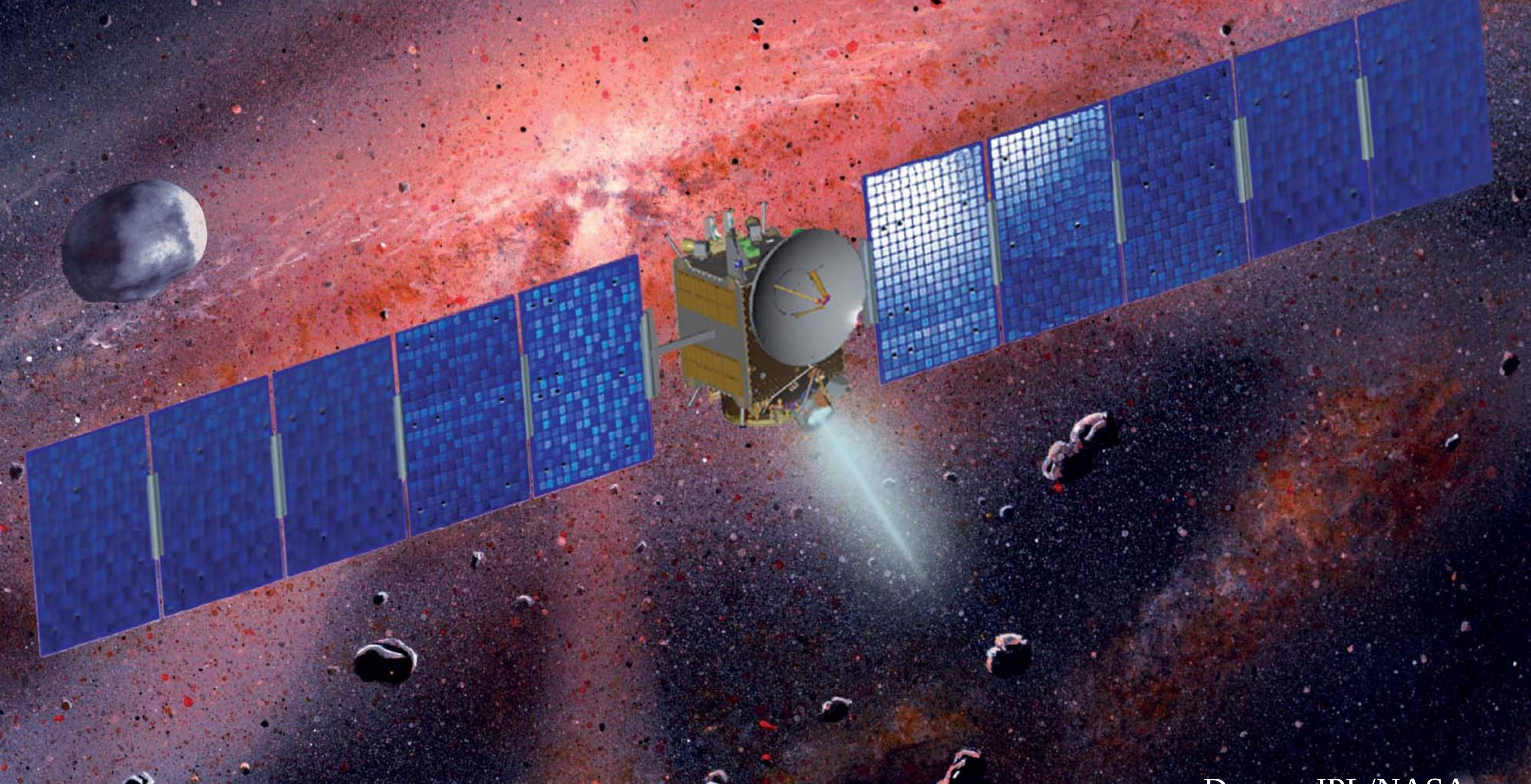


Dawn - JPL/NASA

What Is An Ion Engine?

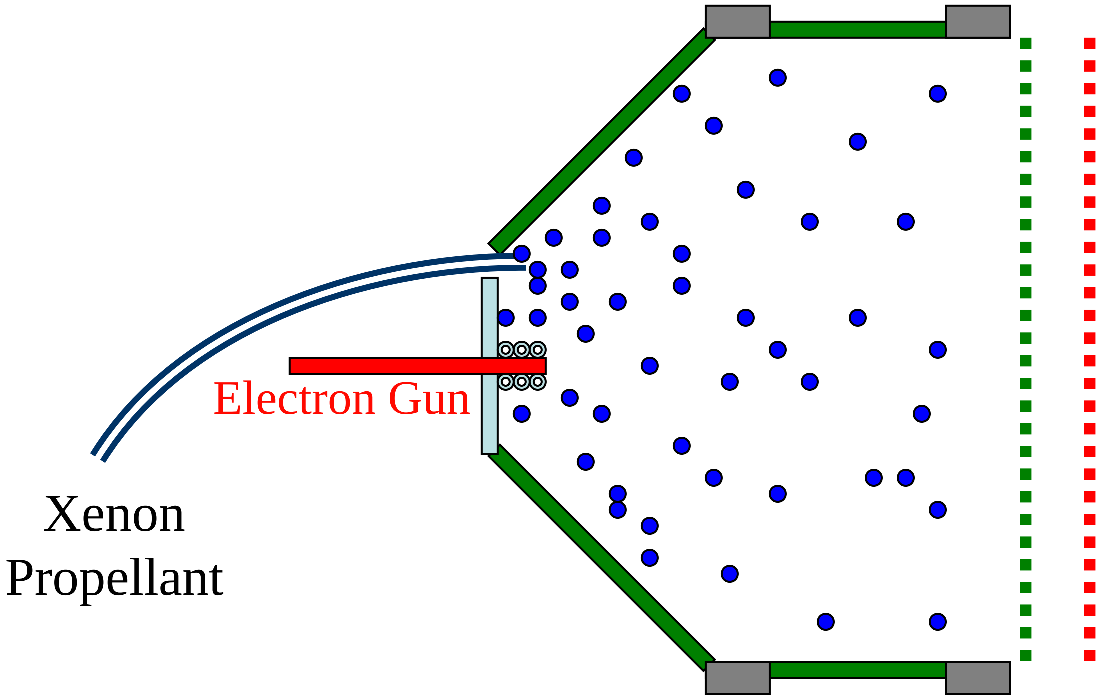
- A rocket engine accelerates propellant which causes a reaction that accelerates the spacecraft in the opposite direction.
- The ultimate capability of an engine is measured by the speed it can accelerate propellant - *exhaust velocity*
- Useful *chemical explosions* are limited to about 3.5 km/s or 8,000 mph.

What Is An Ion Engine?

- Traditional chemical rockets are near the peak theoretical capability
- To get the propellant exiting much faster, we need a non-thermal means of propellant acceleration.

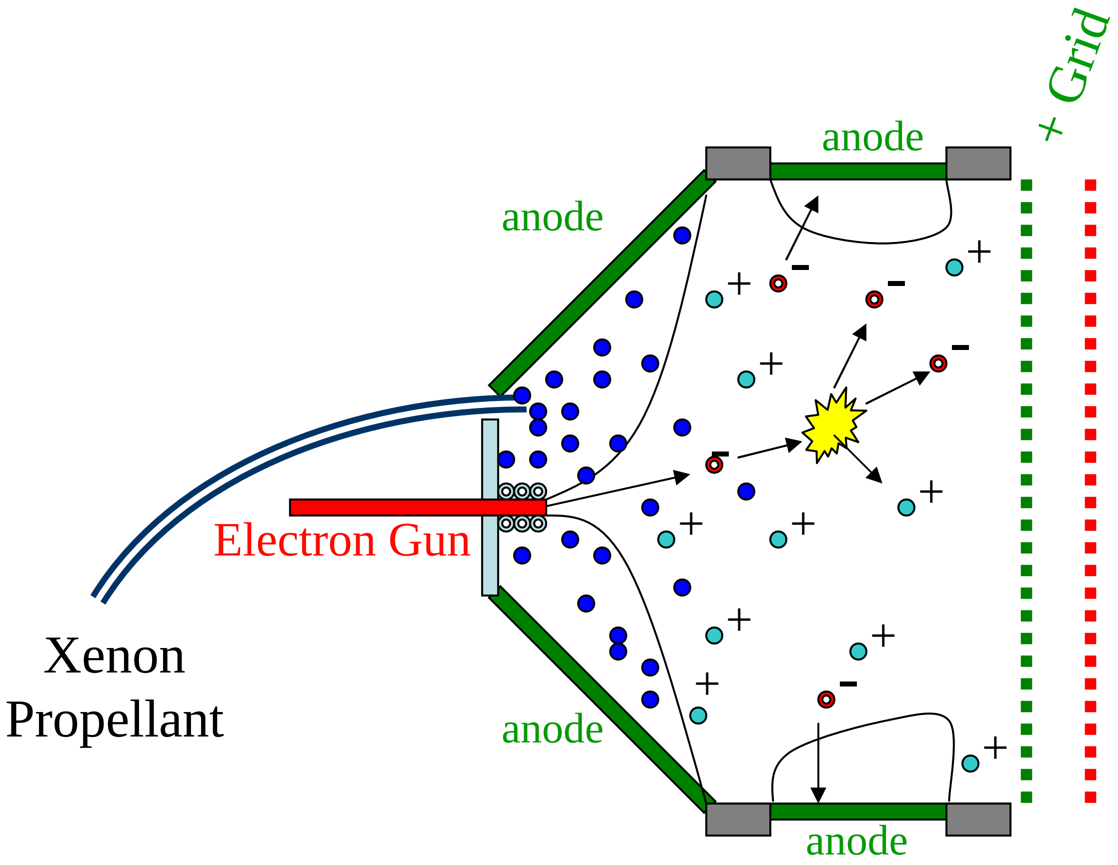
The ion engine is really a particle accelerator. Particle accelerators on Earth are limited only by the speed of light.

How an Ion Engine Works



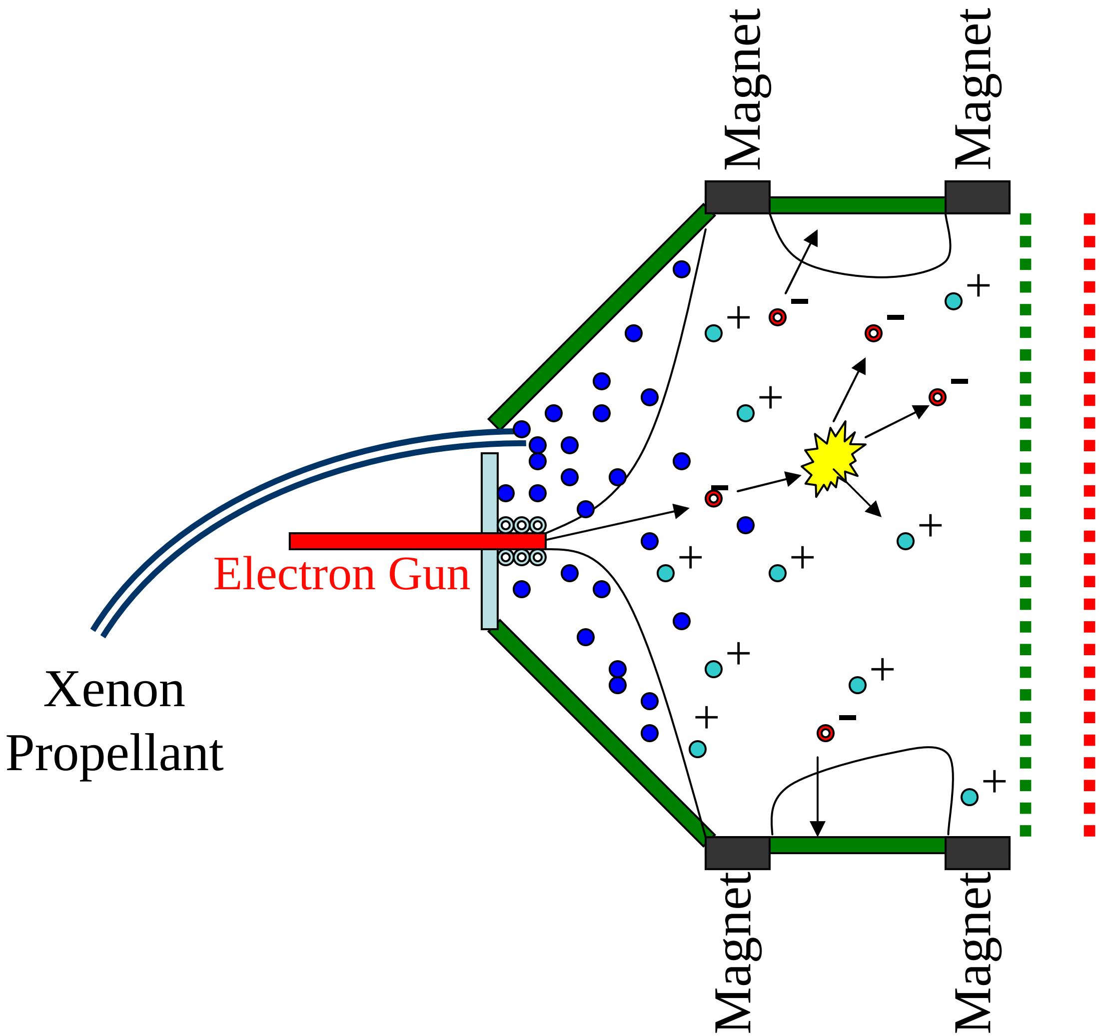
Xenon propellant is injected into the propulsion chamber

How an Ion Engine Works



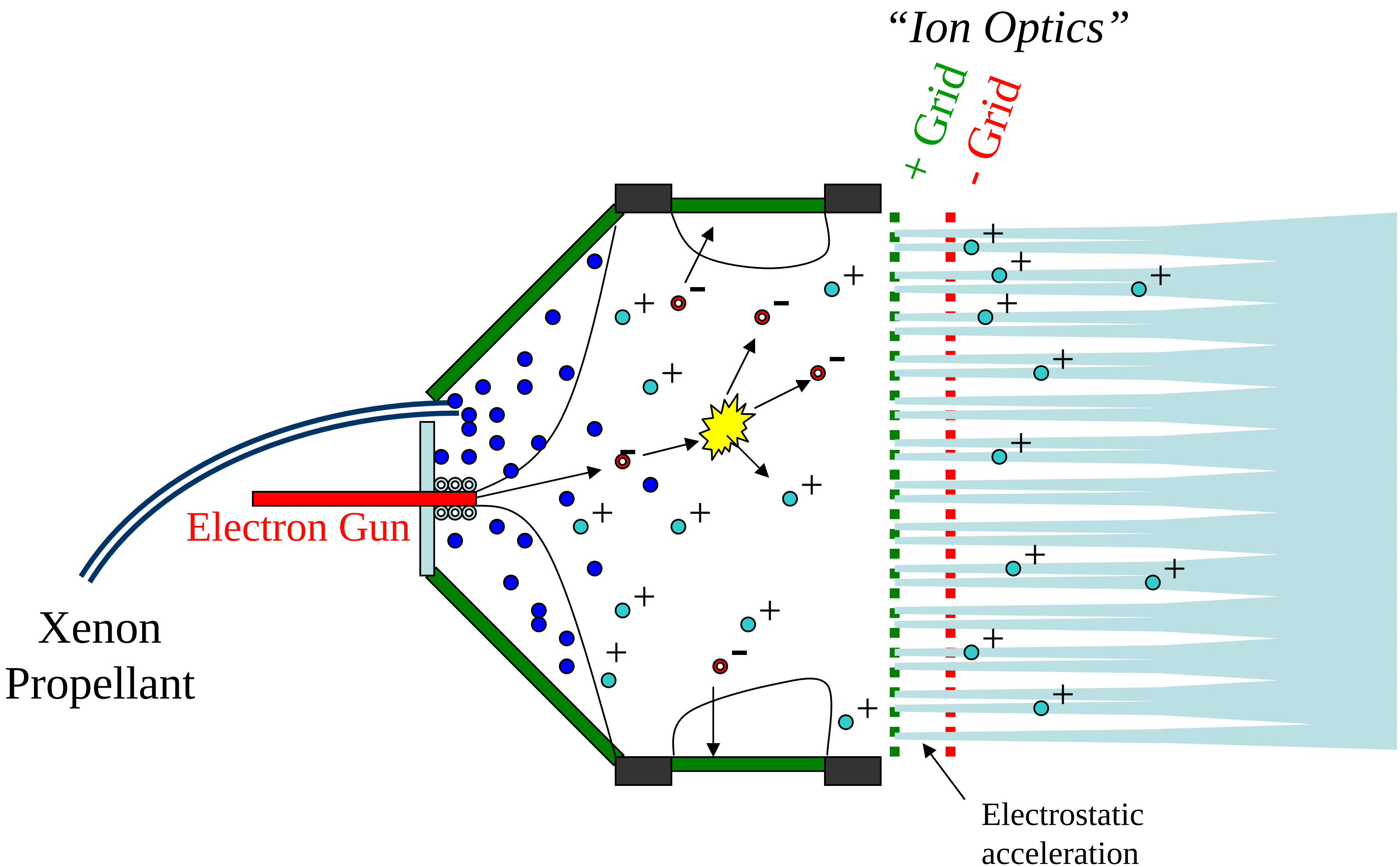
The propellant is ionized by electron bombardment, the + grid and thruster walls (anodes) absorb electrons.

How an Ion Engine Works



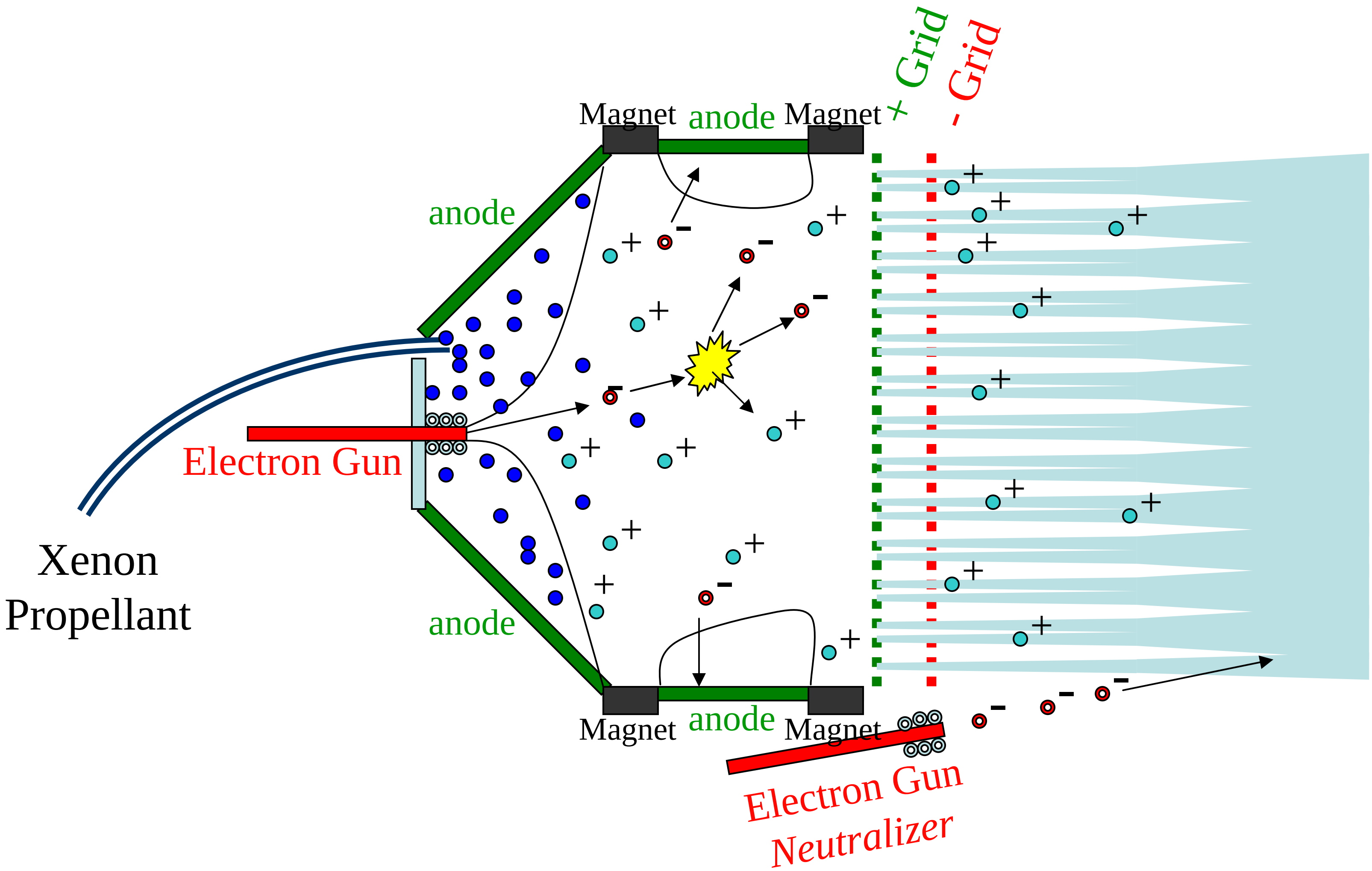
Ring magnets increase the electron residence time to improve ionization efficiency

How an Ion Engine Works

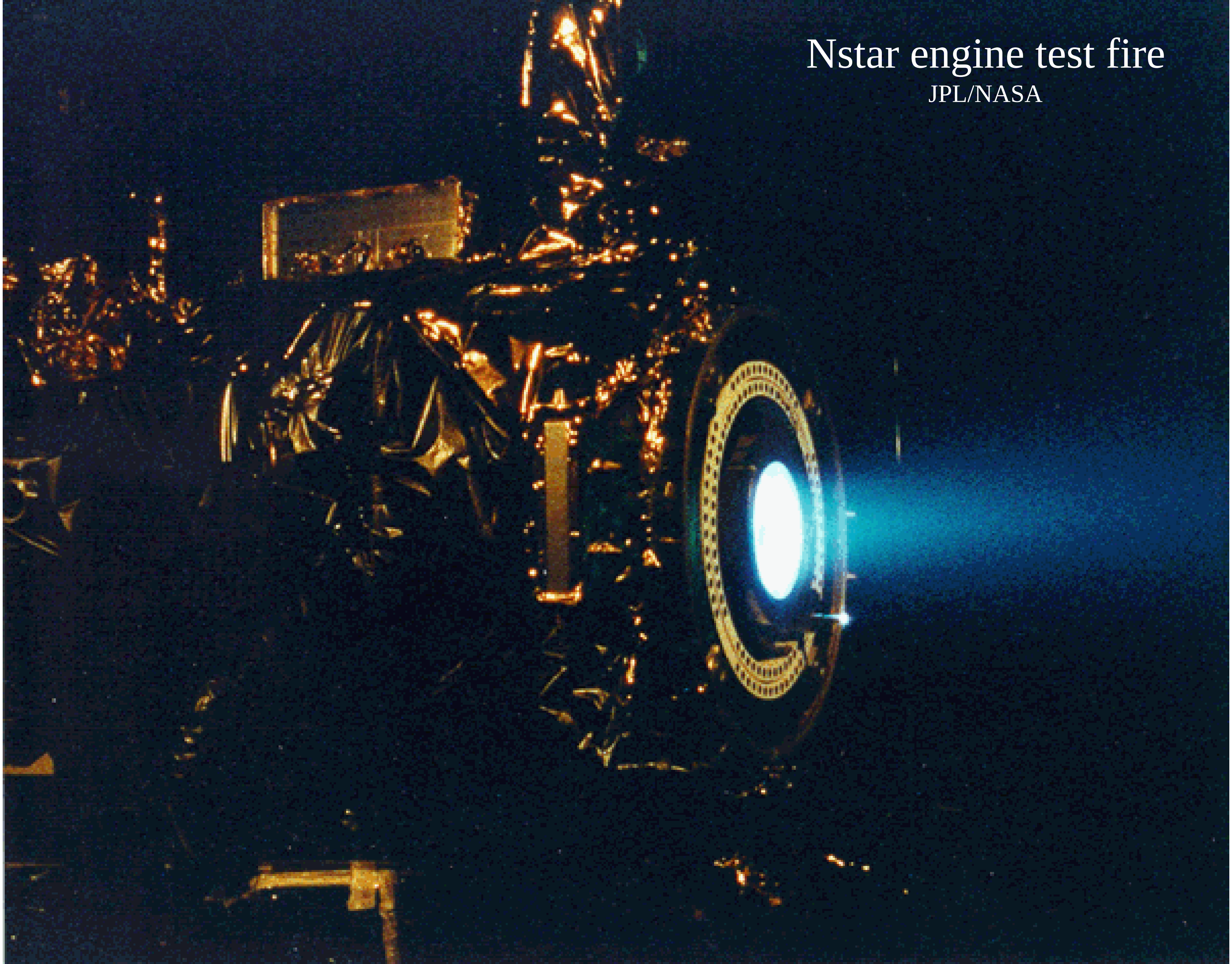


The ions diffuse towards the holes in the + grid, feel the - grid, are electrostatically accelerated to high speed, and focused through the holes on the - grid into space.

How an Ion Engine Works



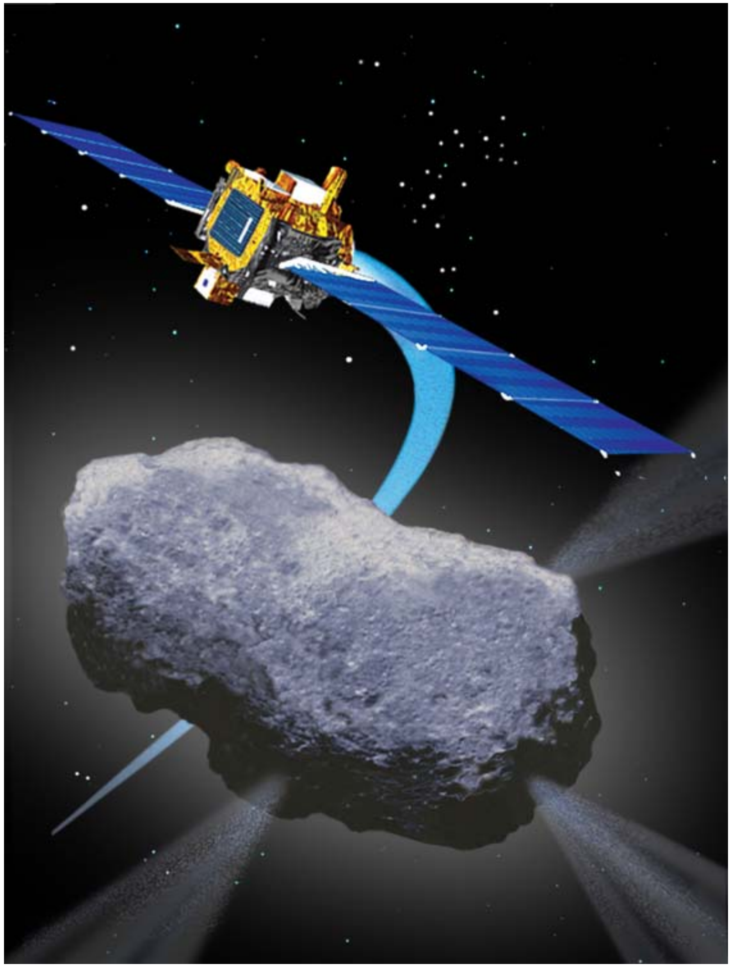
A neutralizer electron gun is required to inject electrons into
The ion beam to keep the spacecraft from building up charge


Nstar engine test fire
JPL/NASA

Some Background on Ion Engines

- Prototypes have been around for more than 40 years. A few have been used in Earth orbit.
- First deep space use was on NASA's *Deep Space One* spacecraft in the late nineties, and later on Japan's *Hayabusa* probe.
- Next deep space mission will be NASA's Dawn discovery mission
- Sometimes called "electric propulsion" - power from solar arrays can be used.

DS1 and Comet Borrelly



JPL/ NASA

- The exhaust velocities for NASA's flight qualified ion engines is about **10 times faster** than exhaust from traditional thrusters. Xenon ions are expelled at about 35 km/s or 77,000 mph from the Nstar thruster (flown on DS1 and to be flown on Dawn).

- Ultimately can get 10 times more push for the same propellant mass. *This enables missions that are impossible using traditional chemical engines.*

- However, ion engines can use only a very small amount of propellant at a time. This means the *force generated is small ("low-thrust") and the engine must operate for a long time.*

QuickTime™ and a
TIFF (LZW) decompressor
are needed to see this picture.

Navigating With Ion Engines …

- Many of the procedures and algorithms developed to guide traditional chemical propelled spacecraft through the Solar System **do not** extend to ion propelled spacecraft:

- Chemical engines are typically on for minutes and off (coasting) for years
- Ion engines are on for years (Dawn will likely operate its thrusters about 6 years!)

- Mission and trajectory design are much more difficult because of the near-continuous thruster operation.

Trajectory Design is an Optimization Problem

- Trajectory design is generally posed as an optimal control problem with a variety of (sometimes peculiar) constraints.

Get from point ***A*** *to point* ***B*** *delivering the* ***maximum*** *payload*

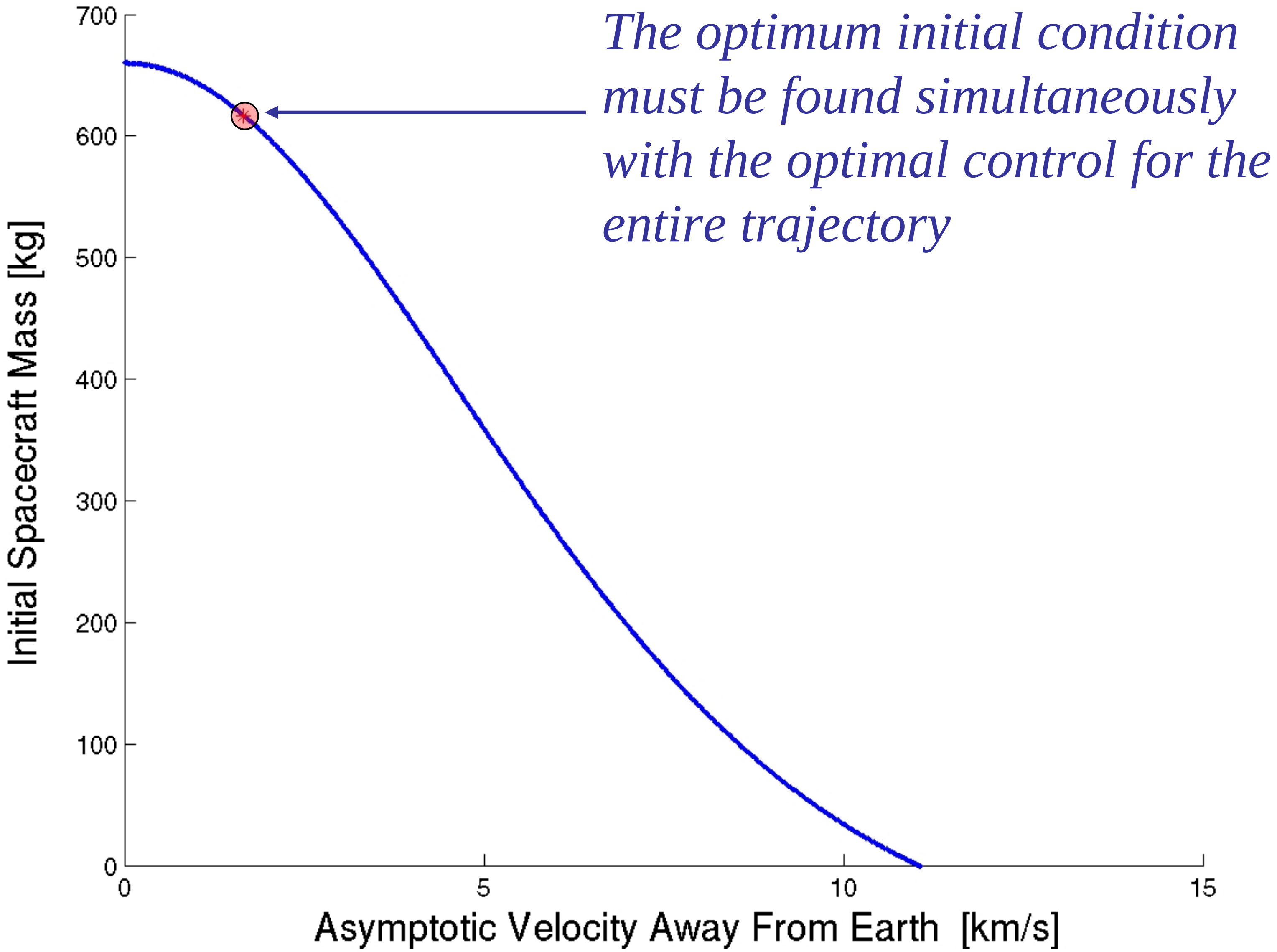
Subject to the laws of physics, engineering constraints, and programmatic constraints

The Point "A"

NASA - launch of DS1

- Point A is often (but not always) a launch from Earth. The Earth is not fixed! It is moving rapidly around the Sun - so launch time is an important variable.
- The initial mass and energy provided by the launch vehicle are important, related variables.
- The optimal use for the launch vehicle is coupled with the optimal use of the ion thrusters in space

Example Launch Vehicle Performance Curve

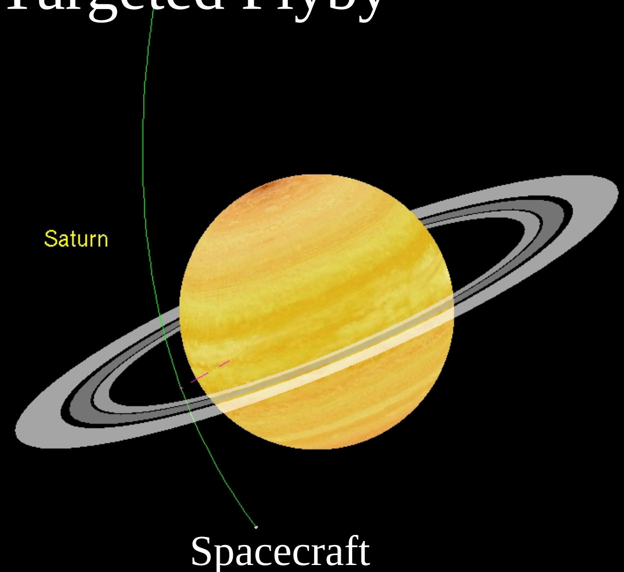


The Point "B" (and Maybe "C")

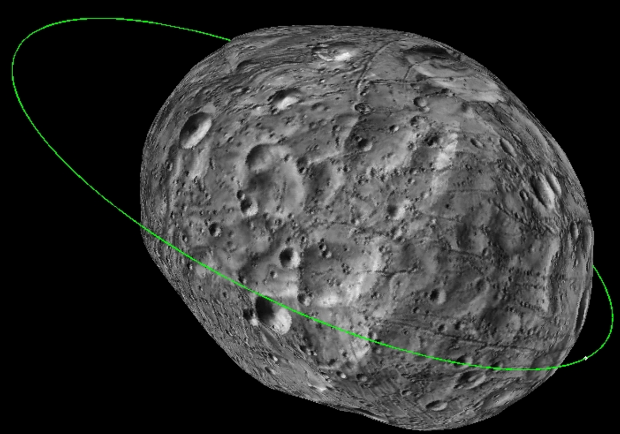
- The intermediate and final targets are usually based on science objectives: body flybys, science observation orbits, etc.

- Targets are always moving, and often only partially defined (leaving some variables available for optimization)

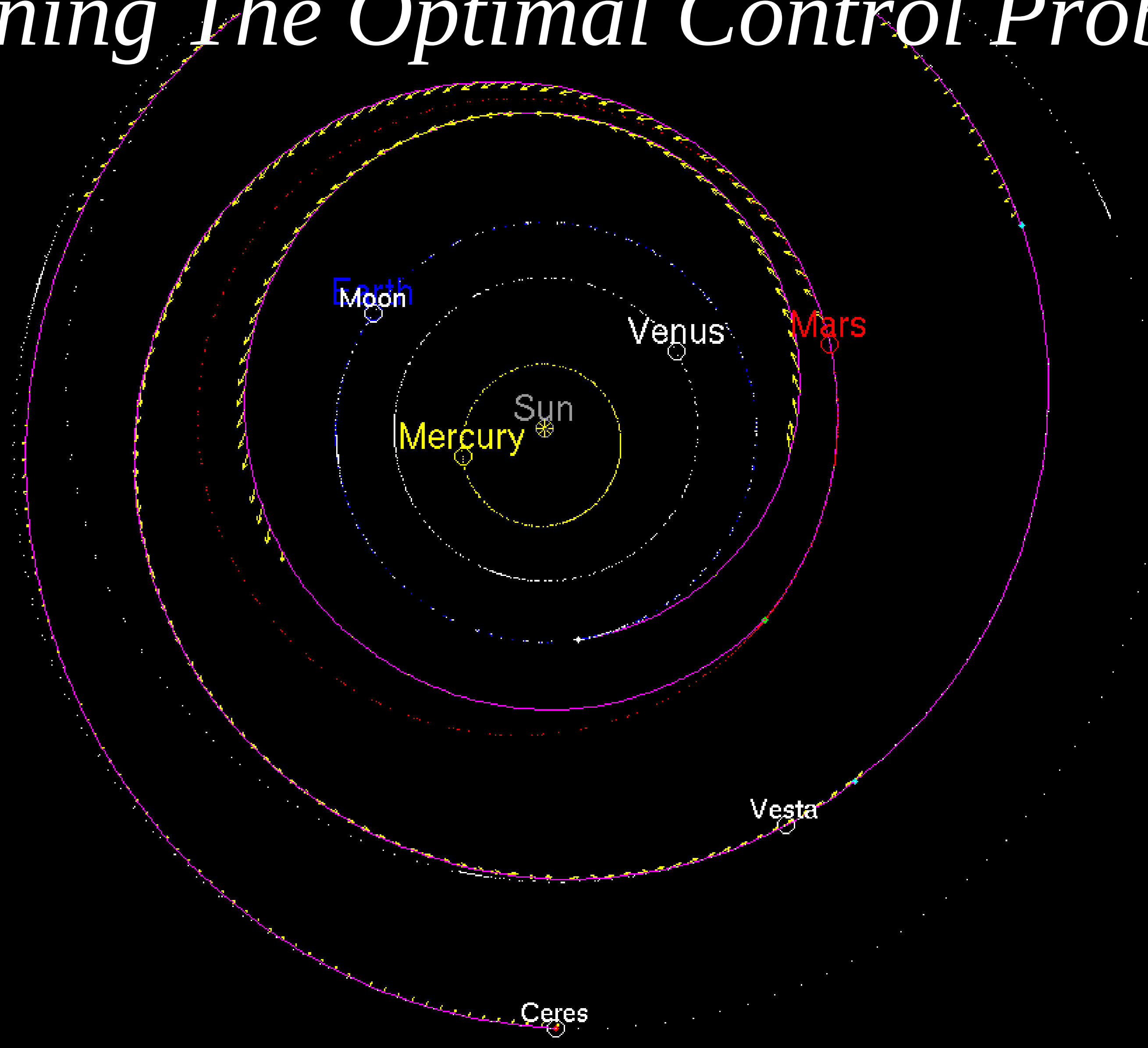
Targeted Flyby



Science Observation Orbit



Defining The Optimal Control Problem



State:

$$x(t) = \begin{bmatrix} x_1(t) \\ x_2(t) \\ x_3(t) \\ x_4(t) \\ x_5(t) \\ x_6(t) \\ x_7(t) \end{bmatrix} = \begin{bmatrix} x\ coordinate\ of\ spacecraft \\ y\ coordinate\ of\ spacecraft \\ z\ coordinate\ of\ spacecraft \\ x\ velocity\ of\ spacecraft \\ y\ velocity\ of\ spacecraft \\ z\ velocity\ of\ spacecraft \\ mass\ of\ the\ spacecraft. \end{bmatrix}$$

Dynamic control:

$$v(t) = \begin{bmatrix} v_1(t) \\ v_2(t) \\ v_3(t) \end{bmatrix} = \begin{bmatrix} x\ component\ of\ thrust \\ y\ component\ of\ thrust \\ z\ component\ of\ thrust. \end{bmatrix}$$

Parameters:

$$w = \begin{bmatrix} w_1 \\ w_2 \\ w_3 \\ w_4 \\ w_5 \\ w_6 \\ w_7 \\ w_8 \\ w_9 \\ w_{10} \\ w_{11} \end{bmatrix} = \begin{bmatrix} date\ of\ trajectory\ start \\ total\ flight\ time \\ longitude\ of\ the\ ascending\ node \\ argument\ of\ the\ periapsis \\ true\ anomaly \\ Orbital\ C_3 \\ Periapsis\ radius \\ inclination \\ initial\ mass \\ thuster\ specific\ impulse \\ solar\ array\ size \end{bmatrix}$$

The optimal Control problem:

Objective: $$maximize_{v(t),w}\ (Spacecraft\ final\ mass)$$

x(t): spacecraft state v(t): thrust vector w: parameters

The optimal Control problem:

Objective: $$maximize_{v(t),w} \; (Spacecraft\ final\ mass)$$

State equation: $$\frac{dx(t)}{dt} = T(x(t), v(t), w, t)$$ *Physics*

x(t): spacecraft state

v(t): thrust vector

w: parameters

The optimal Control problem:

Objective: $$maximize_{v(t),w} \ (Spacecraft\ final\ mass)$$

State equation: $$\frac{dx(t)}{dt} = T(x(t), v(t), w, t)$$

Initial Condition: $$x(t_0) = \Gamma(w)$$

x(t): spacecraft state v(t): thrust vector w: parameters

The optimal Control problem:

Objective: $$maximize_{v(t),w} \; (Spacecraft\ final\ mass)$$

State equation: $$\frac{dx(t)}{dt} = T(x(t), v(t), w, t)$$

Initial Condition: $$x(t_0) = \Gamma(w)$$

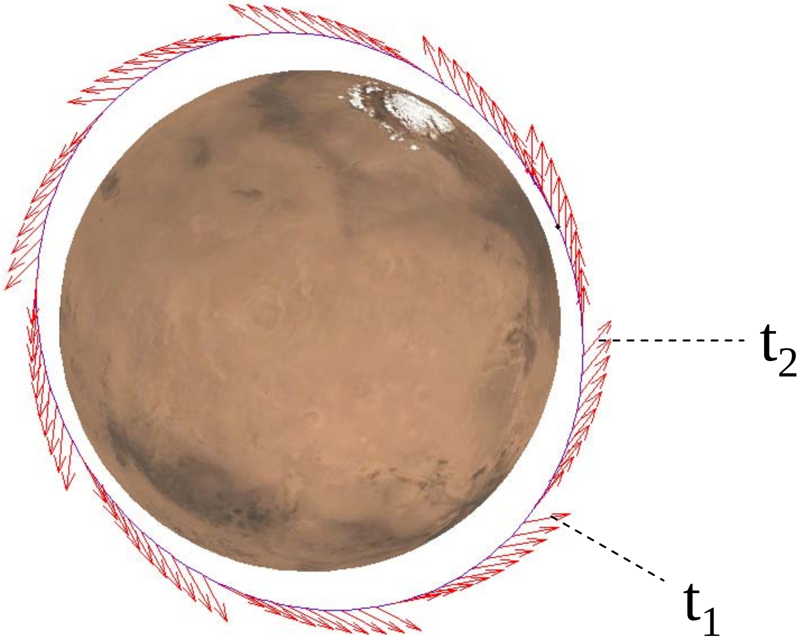
Target Final State and constraints: $$\Psi(x(t), v(t), w, t) = \; or \; \leq k_1$$

x(t): spacecraft state v(t): thrust vector w: parameters

The optimal Control problem:

Control Dynamics Limitation
(simplest example):

$$\frac{dv(t)}{dt} = 0 \qquad \forall\, t \in (t_1, t_2)$$



Dynamic limitations represent engineering constraints or operational requirements. For example, continuous, very slow, slewing of the spacecraft to change the thrust direction is not desirable for deep space operations. Instead the thrust direction is altered quickly at regular intervals.

x(t): spacecraft state   v(t): thrust vector   w: parameters

What is in the state equation:

Spacecraft:

$$\frac{dx}{dt} = \left[\begin{array}{c} \overrightarrow{x}_{4:6}(t) \\ \frac{\overrightarrow{v}(t)}{x_7(t)} + \frac{\overrightarrow{Rad}(x,t)}{x_7(t)} + \sum_{i=1}^{N_{bodies}} \overrightarrow{Grav}_i(x,t) \\ \dot{m}(v,x,t) \end{array} \right] \begin{array}{l} \leftarrow \text{Velocity} \\ \leftarrow \text{Acceleration} \\ \leftarrow \text{Mass change} \end{array}$$

x(t): spacecraft state

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Thrust

x(t): spacecraft state

v(t): thrust vector

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←Velocity

←Acceleration

←Mass change

Thrust

Radiation pressure

x(t): spacecraft state

v(t): thrust vector

w: parameters

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Thrust

Radiation pressure

- Ion thruster propellant rate

x(t): spacecraft state

v(t): thrust vector

w: parameters

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$$\frac{dx}{dt} = \begin{bmatrix} \vec{x}_{4:6}(t) \\ \frac{\vec{v}(t)}{x_7(t)} + \frac{\overrightarrow{Rad}(x,t)}{x_7(t)} + \sum_{i=1}^{N_{bodies}} \overrightarrow{Grav}_i(x,t) \\ \dot{m}(v,x,t) \end{bmatrix} \begin{array}{l} \leftarrow \text{Velocity} \\ \leftarrow \text{Acceleration} \\ \leftarrow \text{Mass change} \end{array}$$

Thrust

Radiation pressure

- Ion thruster propellant rate

Gravitational terms: N - body gravity, gravitational harmonics, and first order relativistic corrections

x(t): spacecraft state   v(t): thrust vector   w: parameters

An Example of the Initial Condition Equation:

$$\Gamma(w) = \begin{Bmatrix} initial\ position \\ initial\ velocity \\ initial\ mass \end{Bmatrix} = \begin{Bmatrix} \overrightarrow{X}\ (\Omega, \omega, \nu, C_3, R_p, i) \\ \overrightarrow{V}\ (\Omega, \omega, \nu, C_3, R_p, i) \\ mlv_c(c_3) + w_9 \end{Bmatrix}$$

An Example of the Initial Condition Equation:

Optimization parameters:
Launch vehicle final state

$$\Gamma(w) = \left\{ \begin{array}{l} initial\ position \\ initial\ velocity \\ initial\ mass \end{array} \right\} = \left\{ \begin{array}{l} \overrightarrow{X}\ (\Omega, \omega, \nu, C_3, R_p, i) \\ \overrightarrow{V}\ (\Omega, \omega, \nu, C_3, R_p, i) \\ mlv_c(c_3) + w_9 \end{array} \right\}$$

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Launch vehicle performance:
Delivered mass versus delivered energy

Models Required:

- Solar array performance as a function of temperature and illumination (distance from the Sun).
- Solar array degradation due to radiation damage
- Spacecraft subsystems (non-ion engine) power consumption
- Ion engine performance (generally non-linear!)
- Launch vehicle performance curves
- Mass distribution models for all gravitating bodies
- Spacecraft component reflectivities for radiation pressure
- Spacecraft attitude control system propellant usage

Constraints:

- Ion engine operational limits
- Launch vehicle ascent geometry limits
- Start, end, and total time of flight
- Propellant consumption (maximum tank size)
- Solar array thermal limits
- Ion engine thrust beam Sun relative direction constraints
- Periodic forced coasting for communications
- Targeted intermediate state conditions
- Targeted final state conditions

Characteristics of the OCP:

- Nonlinear objective, constraints, and state equation
- Non-convex
- System response is “Knife edged” (second order method is essential)
- Control is discontinuous in time
- State equation is discontinuous in time
- State equation is non-autonomous
- Large changes in physical scale occur over the problem’s time horizon

Tough problem!

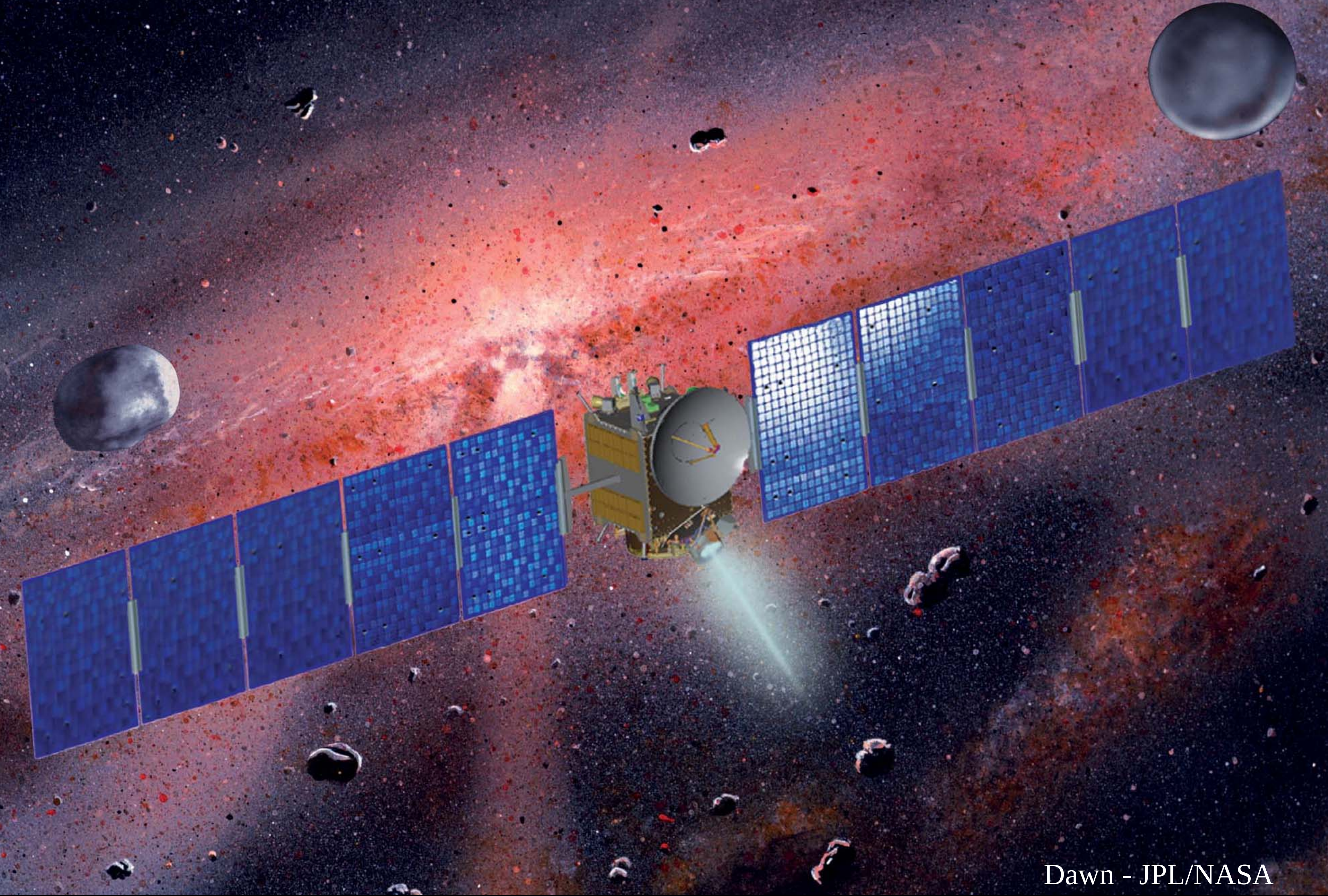
Method of Solution:

- Nonlinear optimal control based on Bellman’s principal of optimality (*Bellman*, *1957*), or more specifically the Hamilton, Jacobi, Bellman equation

- I developed an algorithm specifically to solve these types of problems:

“Static Dynamic Control” (*Whiffen*, *1999*)

Application: *The Dawn Discovery Mission*



Dawn - JPL/NASA

NASA's Dawn Discovery Mission

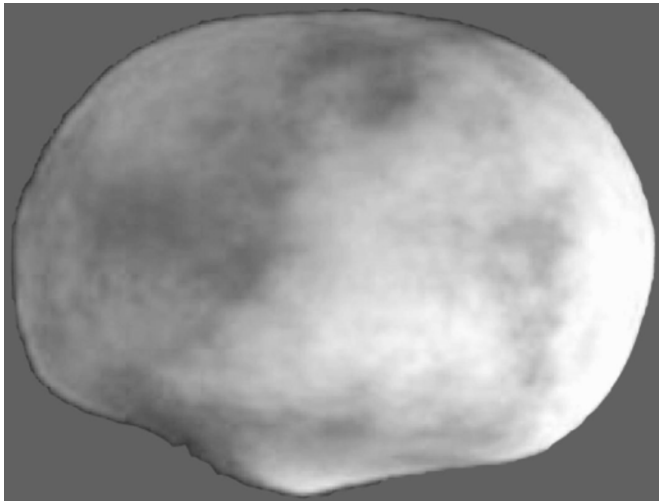
- Dawn will launch June or July of 2007.
- Destination: two largest asteroids or proto planets **Ceres** and **Vesta**. Both would be planets if not for the gravitation of Jupiter

Dawn will be the first mission to rendezvous with a main belt asteroid and the first mission to orbit two target bodies (this requires a 3-fold increase in propulsive capability over any previous mission)

- **Major participants**: University of California, Jet Propulsion Laboratory, Orbital Sciences Corporation, Italian space agency, German Aerospace Center - Mission cost $446 million

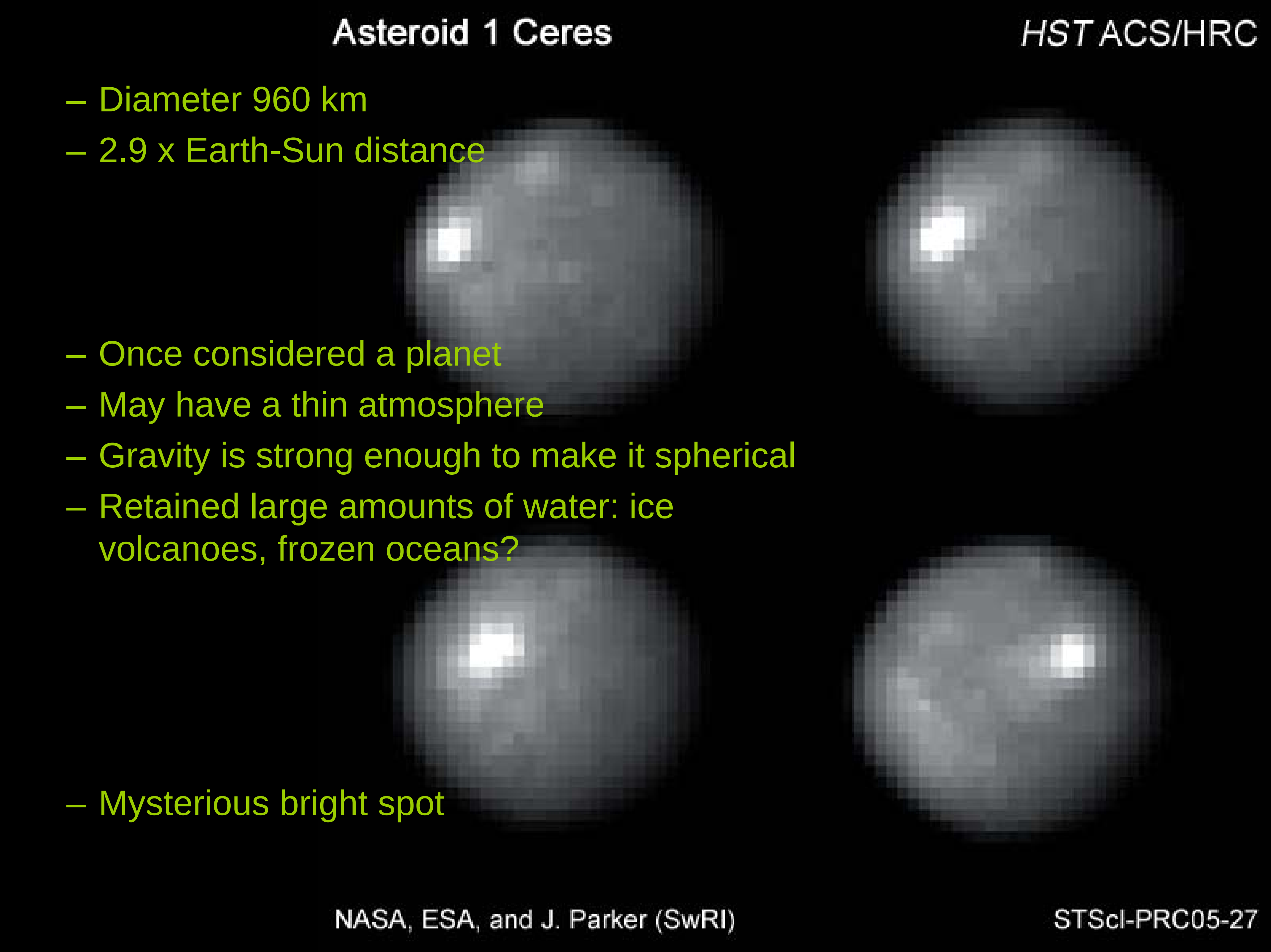
Science Motivation

- Dawn will compare Vesta and Ceres, giving insight into conditions at the formation of the solar system.
- **Vesta**
  - Diameter 530 km
  - Orbits Sun at 2.3 x Earth-Sun distance
  - Differentiated (melted, lava flows, dry)
  - South pole was “blown” away by a violent impact. This excavation will let us look into Vesta’s deep interior
  - Vesta is thought to be the source of Eucritic meterorites



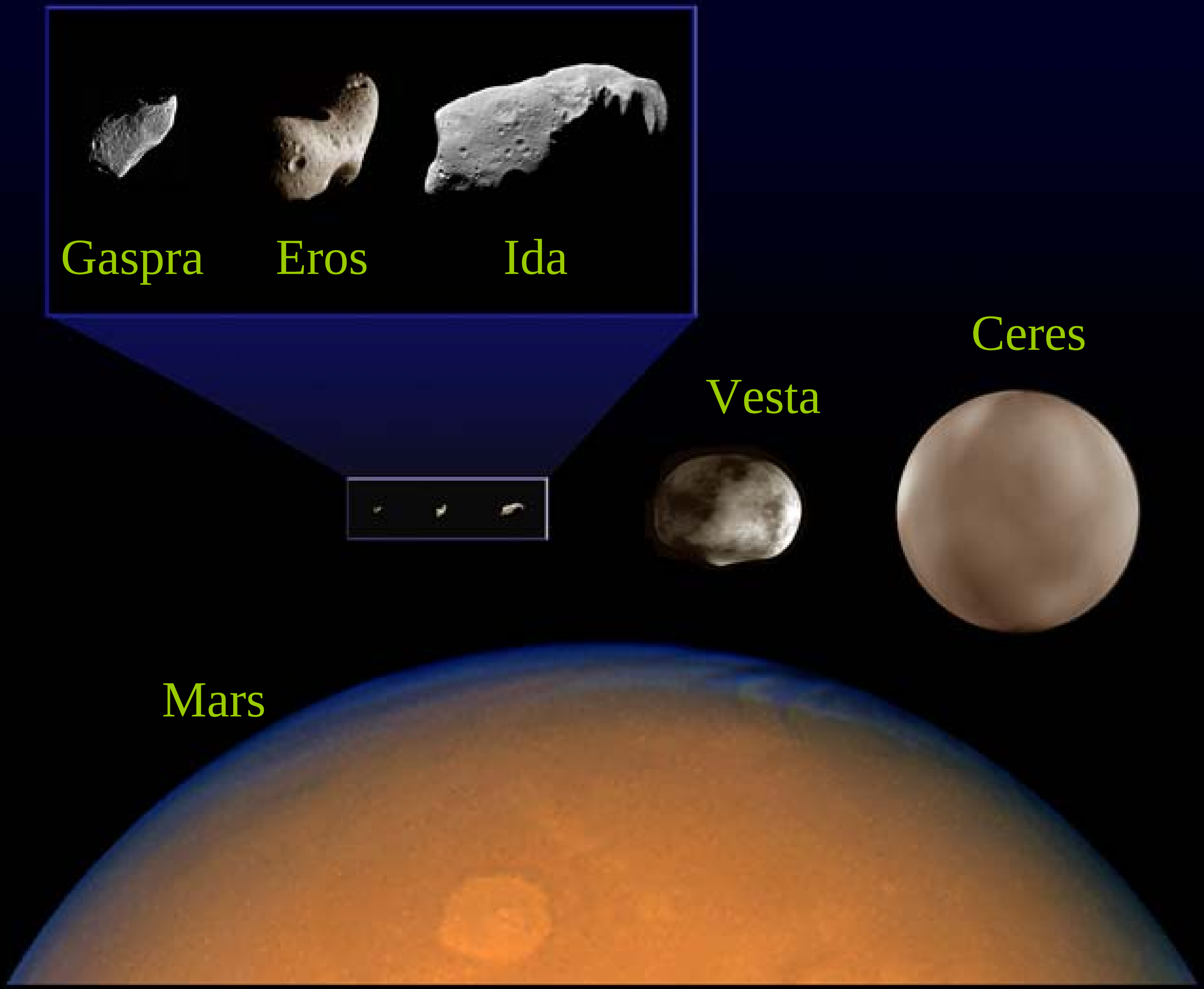
Smoothed HST image of Vesta

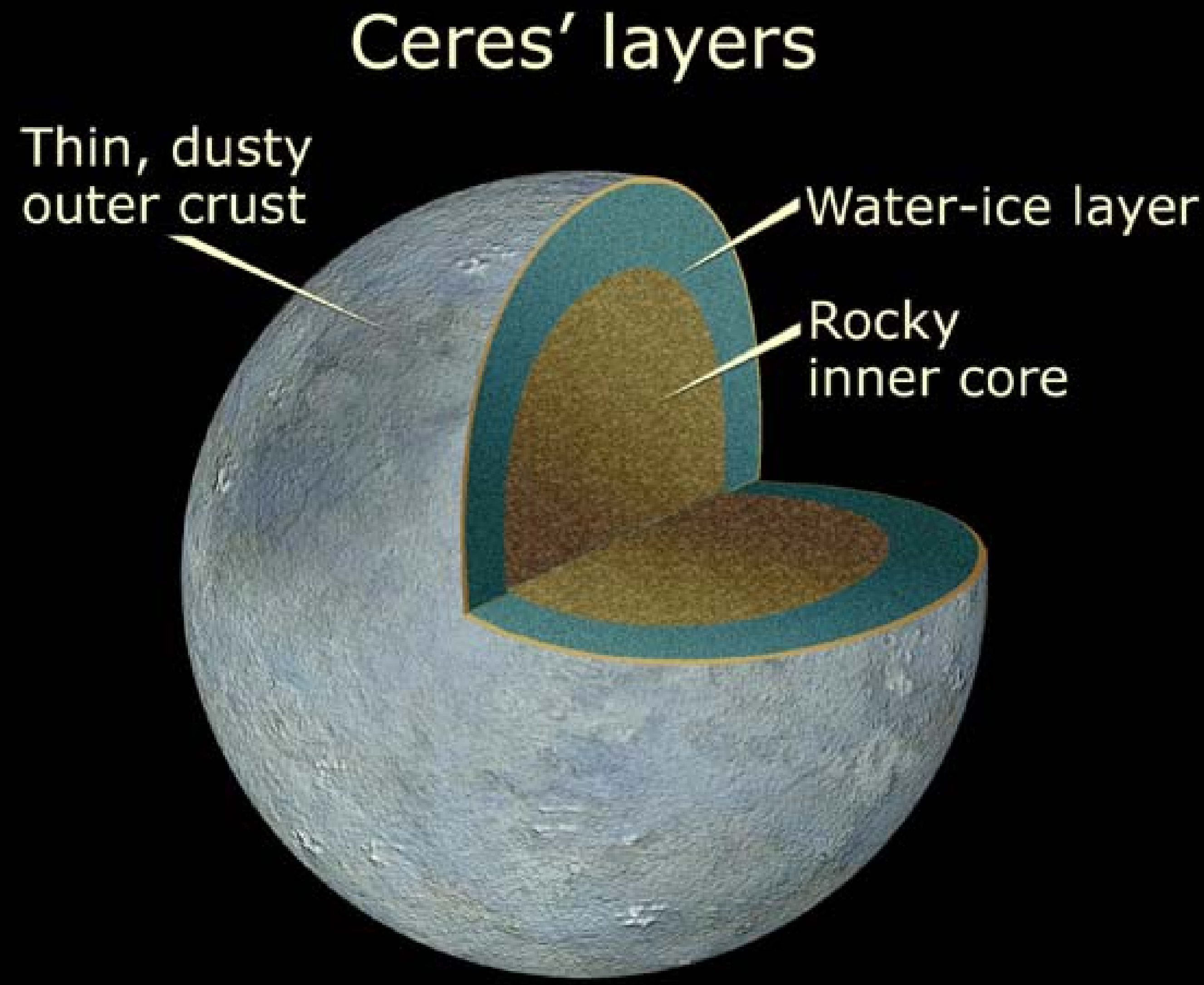
Eucritic meteorite



- Diameter 960 km
- 2.9 x Earth-Sun distance
- Once considered a planet
- May have a thin atmosphere
- Gravity is strong enough to make it spherical
- Retained large amounts of water: ice volcanoes, frozen oceans?
- Mysterious bright spot

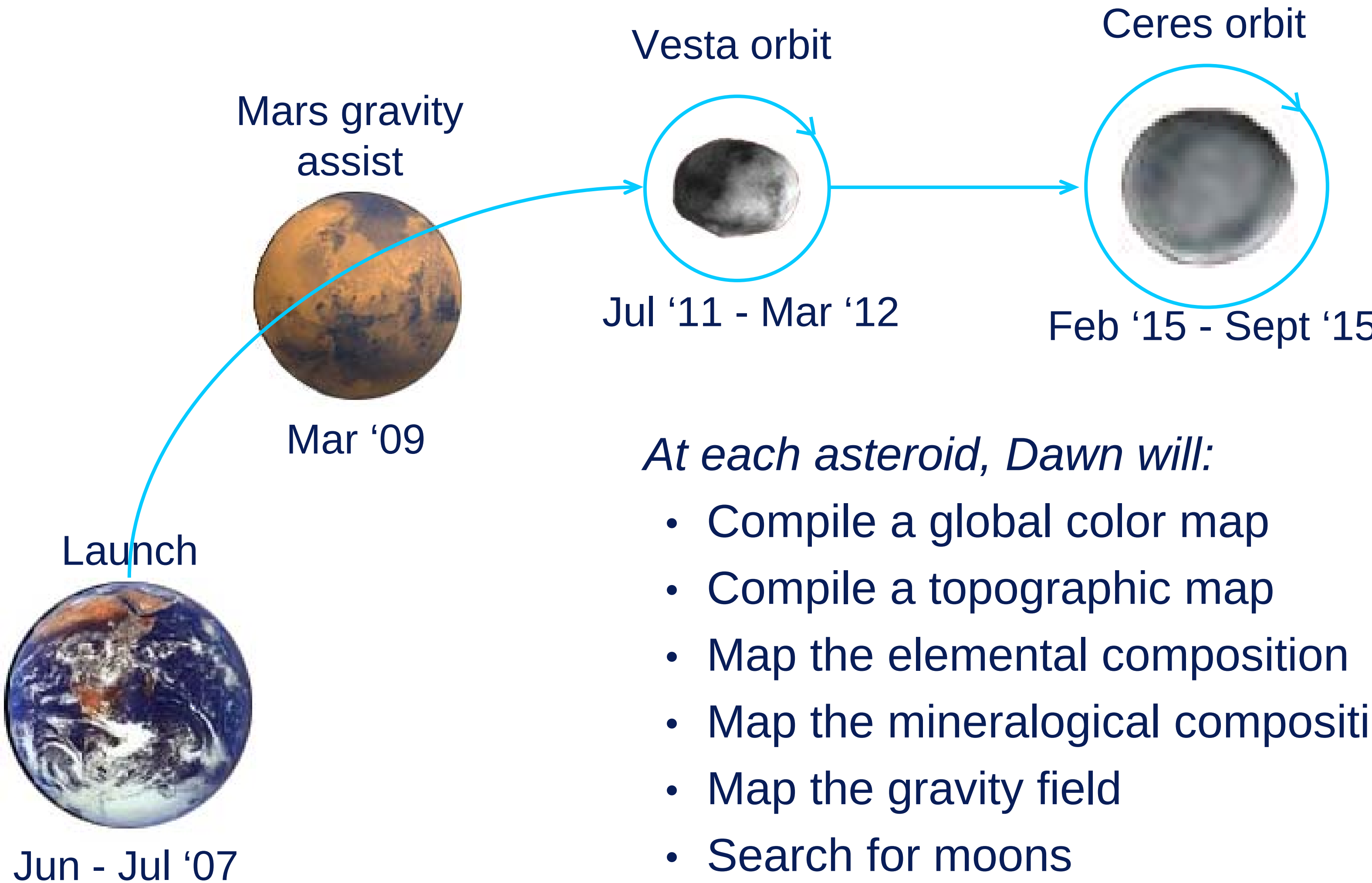
Relative Sizes





NASA, ESA, and A. Field STScI

Mission Itinerary

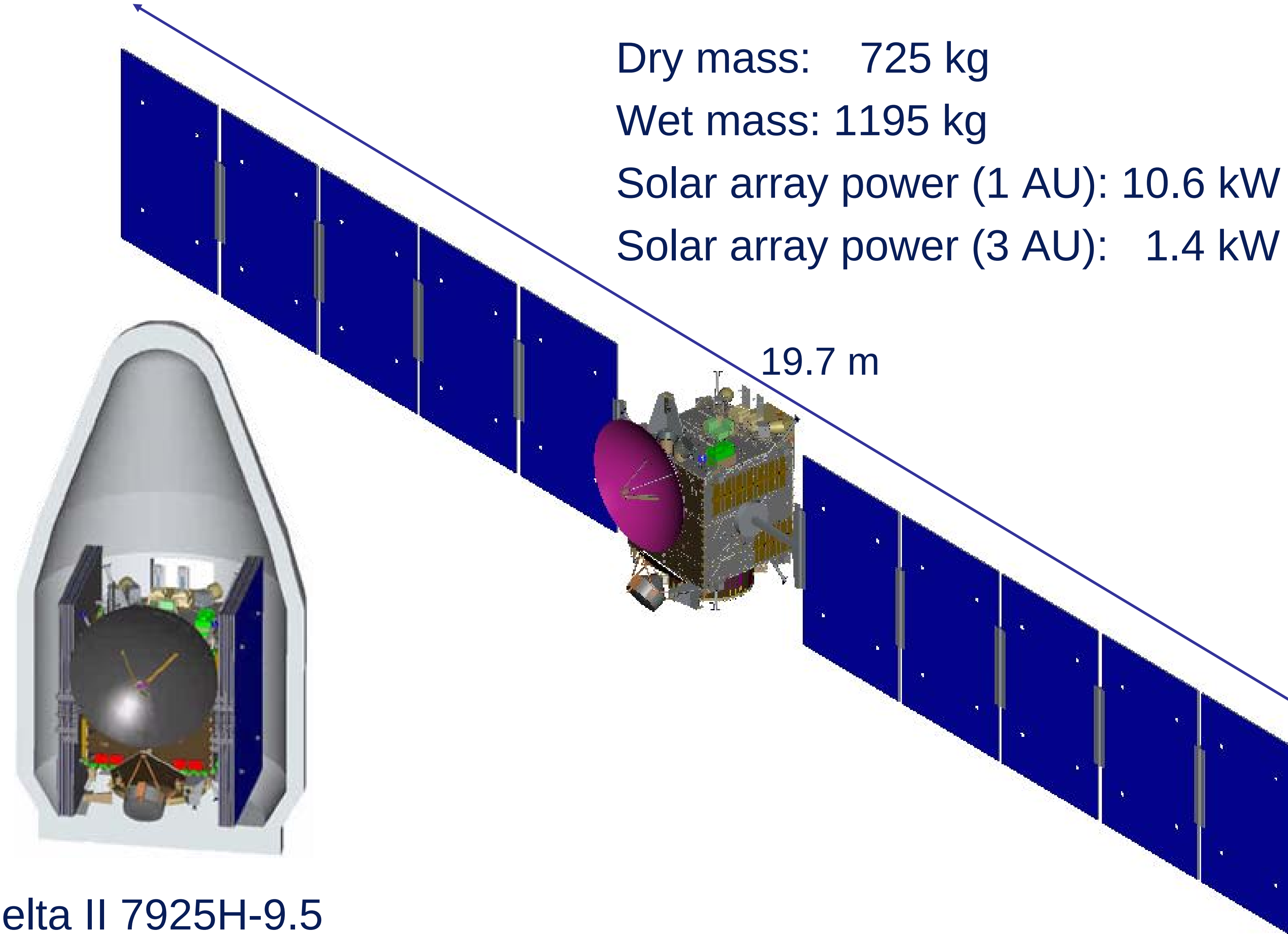


At each asteroid, Dawn will:

- Compile a global color map
- Compile a topographic map
- Map the elemental composition
- Map the mineralogical composition
- Map the gravity field
- Search for moons

Slide credit: Dr. Marc Rayman

Spacecraft



Delta II 7925H-9.5

Dry mass: 725 kg

Wet mass: 1195 kg

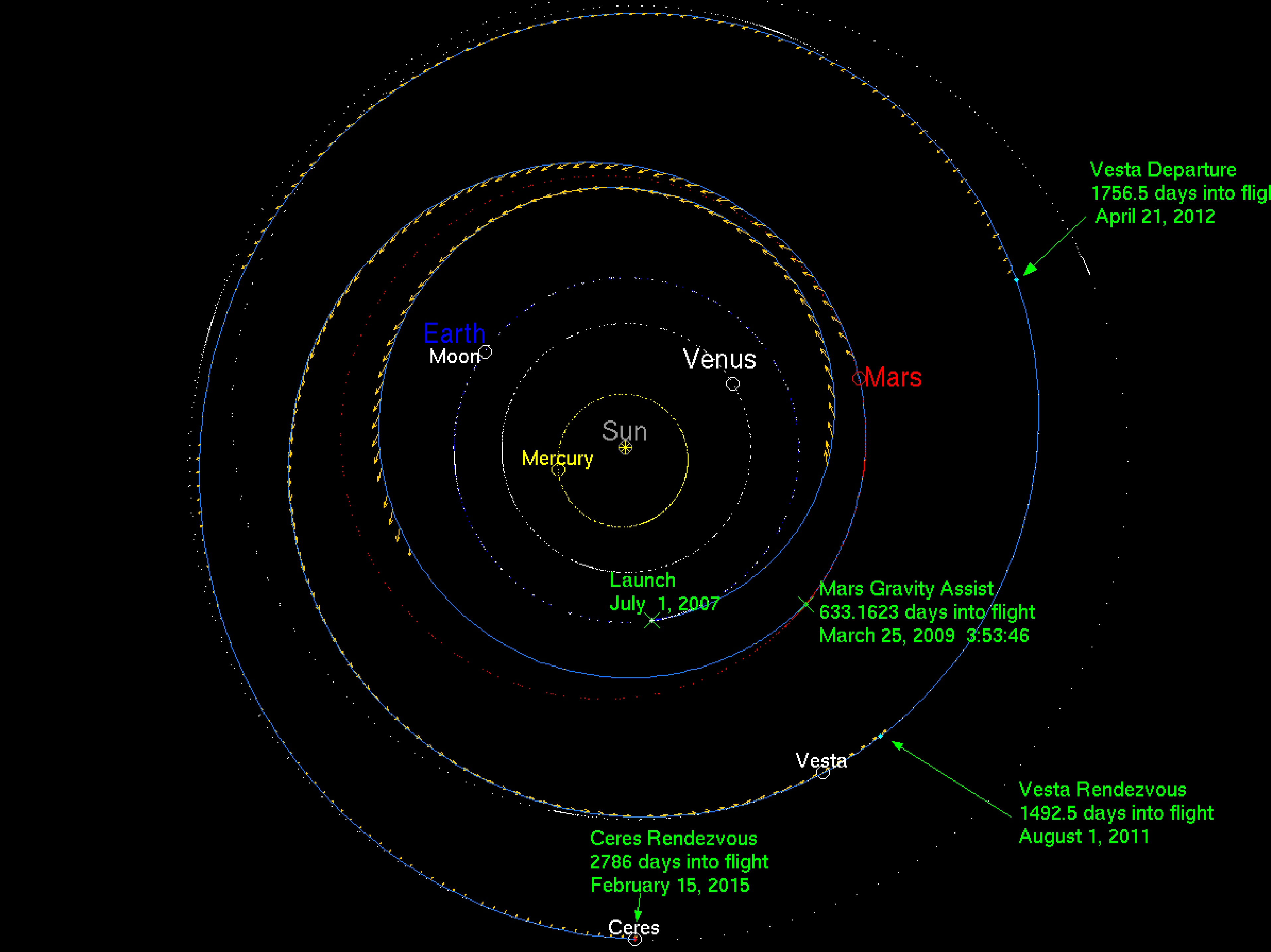
Solar array power (1 AU): 10.6 kW

Solar array power (3 AU): 1.4 kW

Slide credit: Dr. Marc Rayman

Dawn’s Optimal Trajectory

- Fully optimized trajectory for launch on July 1, 2007 through Ceres arrival in February 15, 2015
- Optimize use of launch vehicle simultaneously with the use of the ion propulsion system through the Mars flyby, Vesta rendezvous and finally the Ceres rendezvous
- Ion thruster can only start up 60 days after launch and is not allowed to operate near Mars



Vesta Departure
1756.5 days into fligh
April 21, 2012
Earth
Moon
Venus
Mars
Sun
Mercury
Launch
July 1, 2007
Mars Gravity Assist
633.1623 days into flight
March 25, 2009 3:53:46
Vesta
Vesta Rendezvous
1492.5 days into flight
August 1, 2011
Ceres Rendezvous
2786 days into flight
February 15, 2015
Ceres

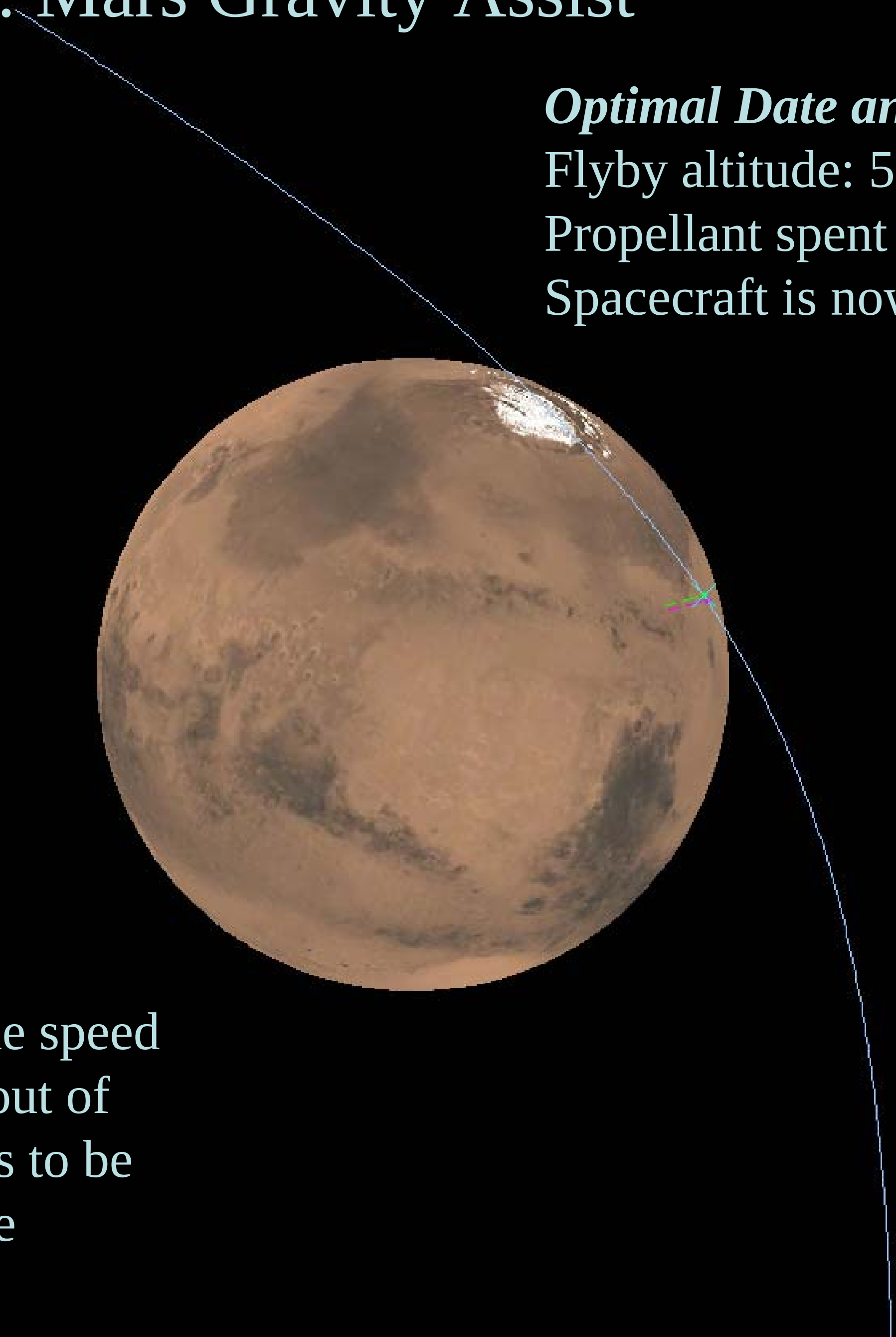
July 1, 2007: Launch

Launch vehicle
Third stage burn out

Optimal Departure:
Declination: -22.0277°
Right ascention: 10.190°
Mass: 1261.31 kg
Energy/mass: 4.829 km^2/s^2

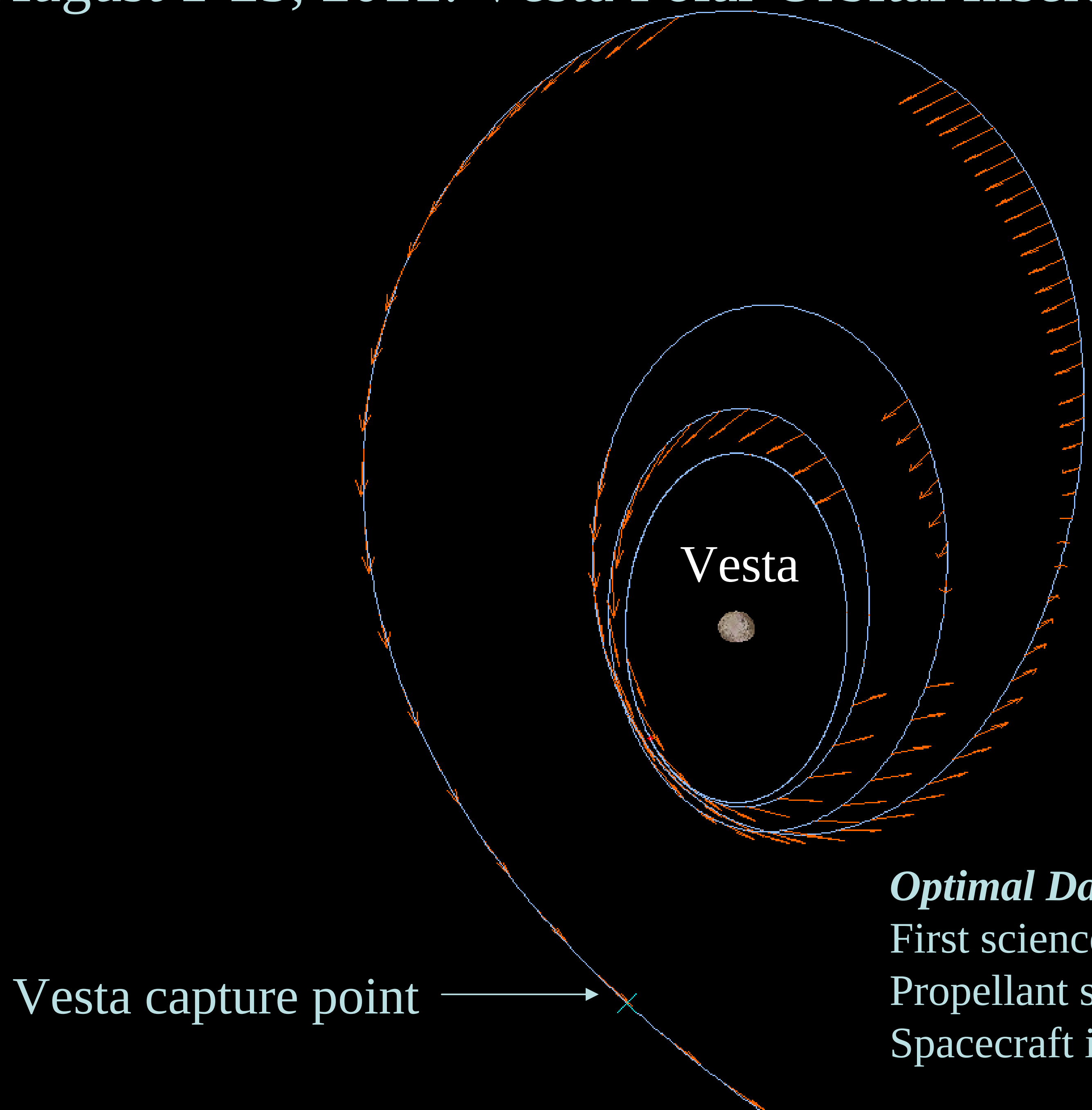
March 23, 2009: Mars Gravity Assist

***Optimal Date and Geometry*.**
Flyby altitude: 501.7 km
Propellant spent so far: 94.5 kg
Spacecraft is now 1164.3 kg



Mars both increases the speed of Dawn and sends it out of the plane of the planets to be in Vesta's orbital plane

August 1-25, 2011: Vesta Polar Orbital Insertion



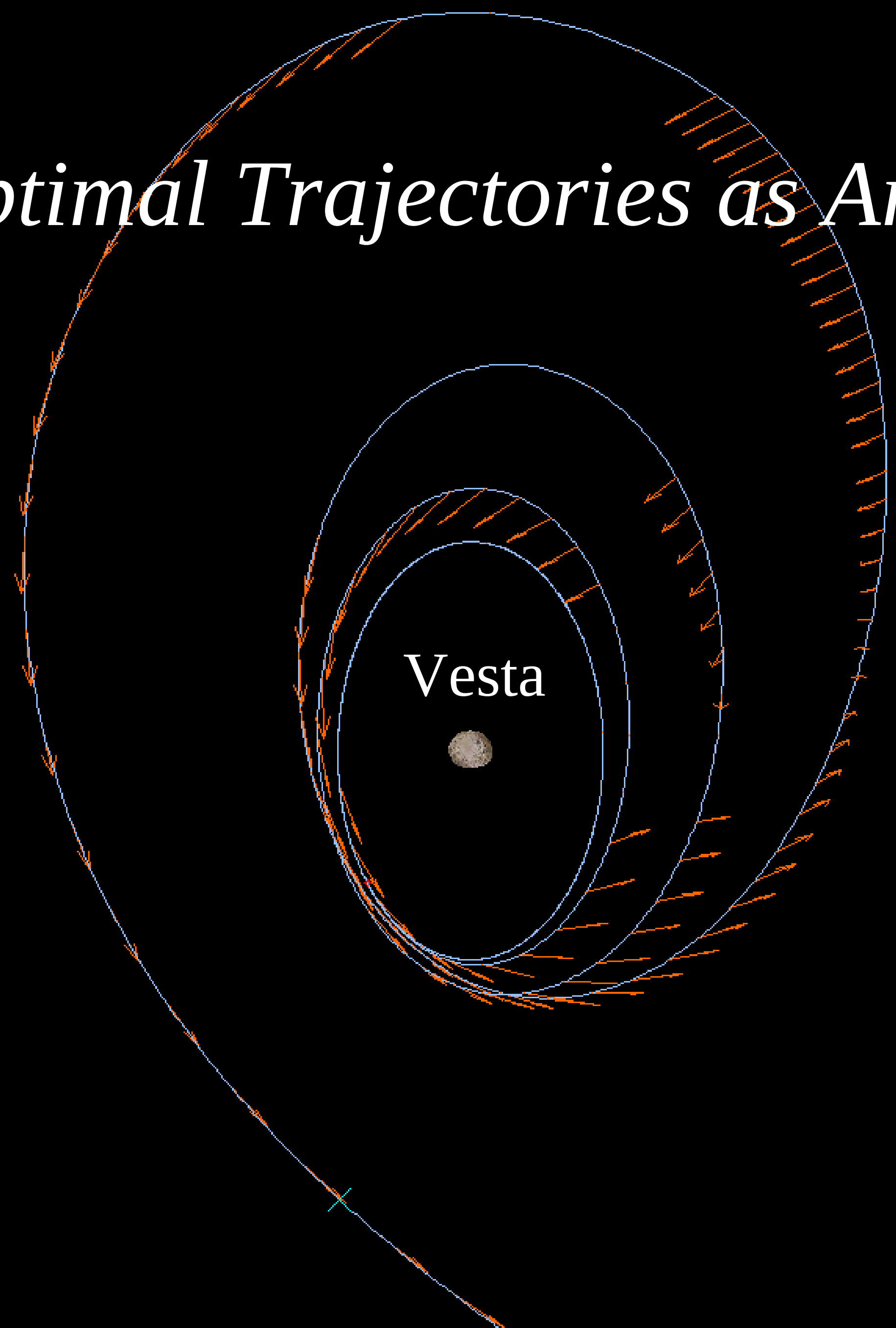
***Optimal Date and Geometry*.**

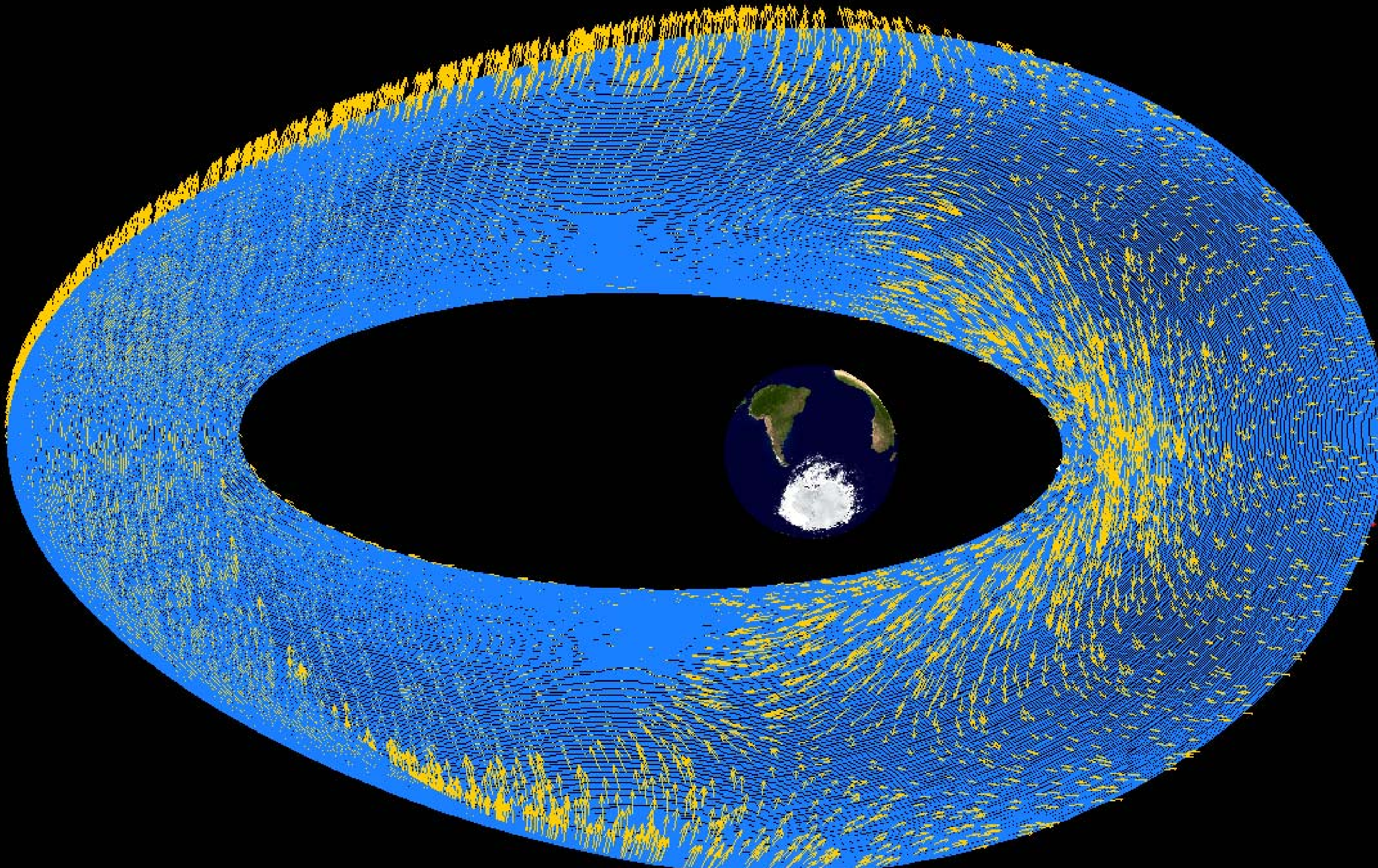
First science orbit: 2700 km

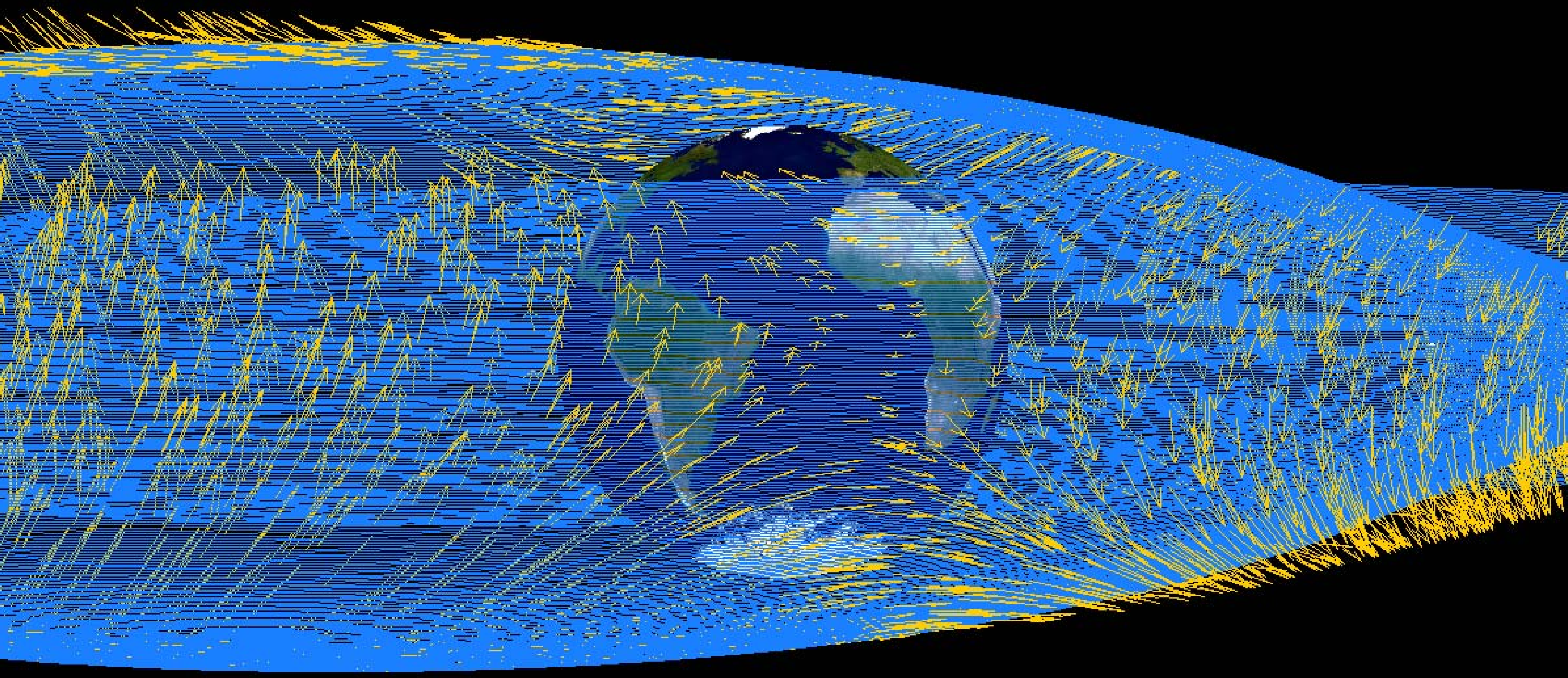
Propellant spent so far: 267.5 kg

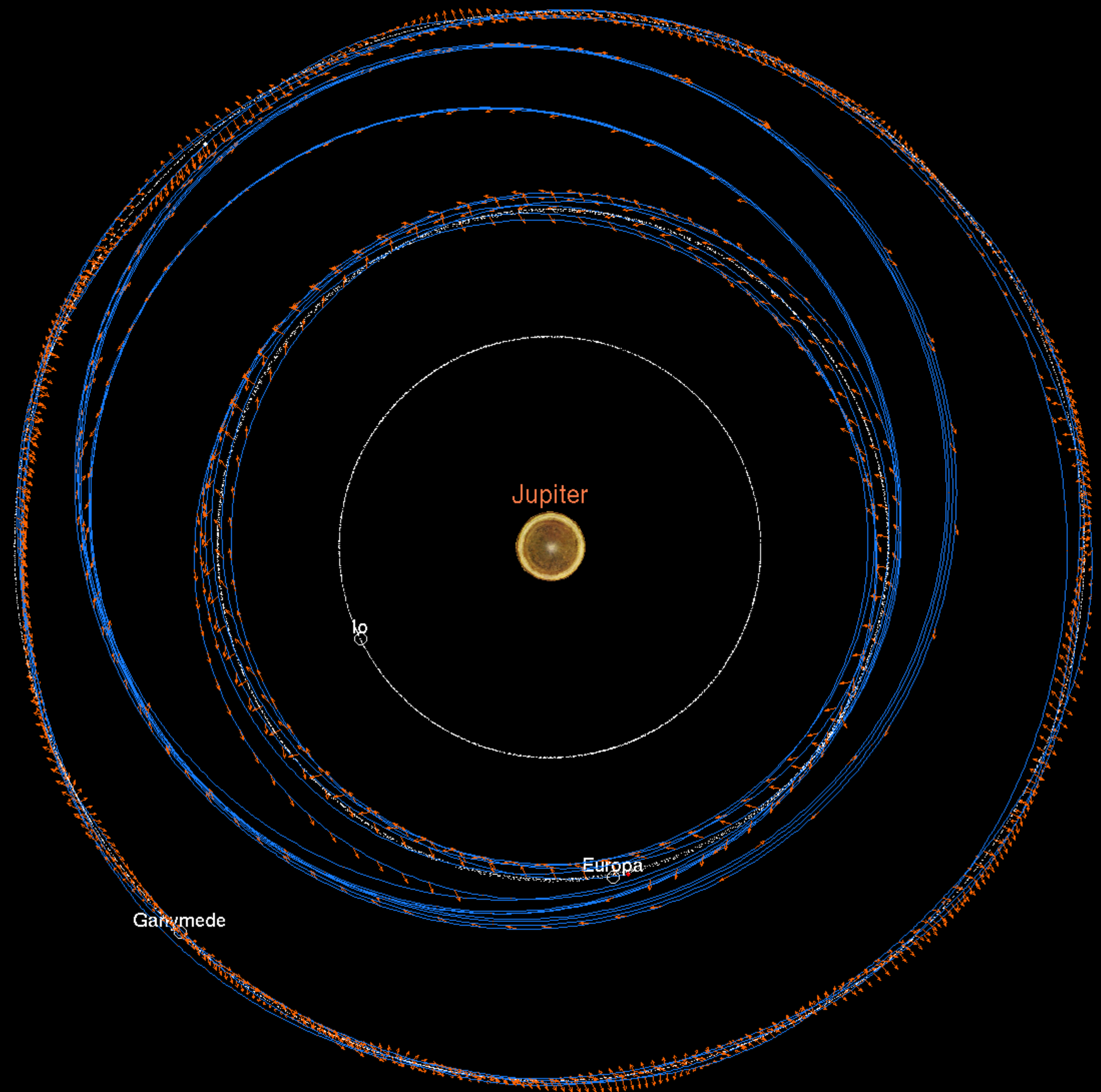
Spacecraft is now 987.3 kg

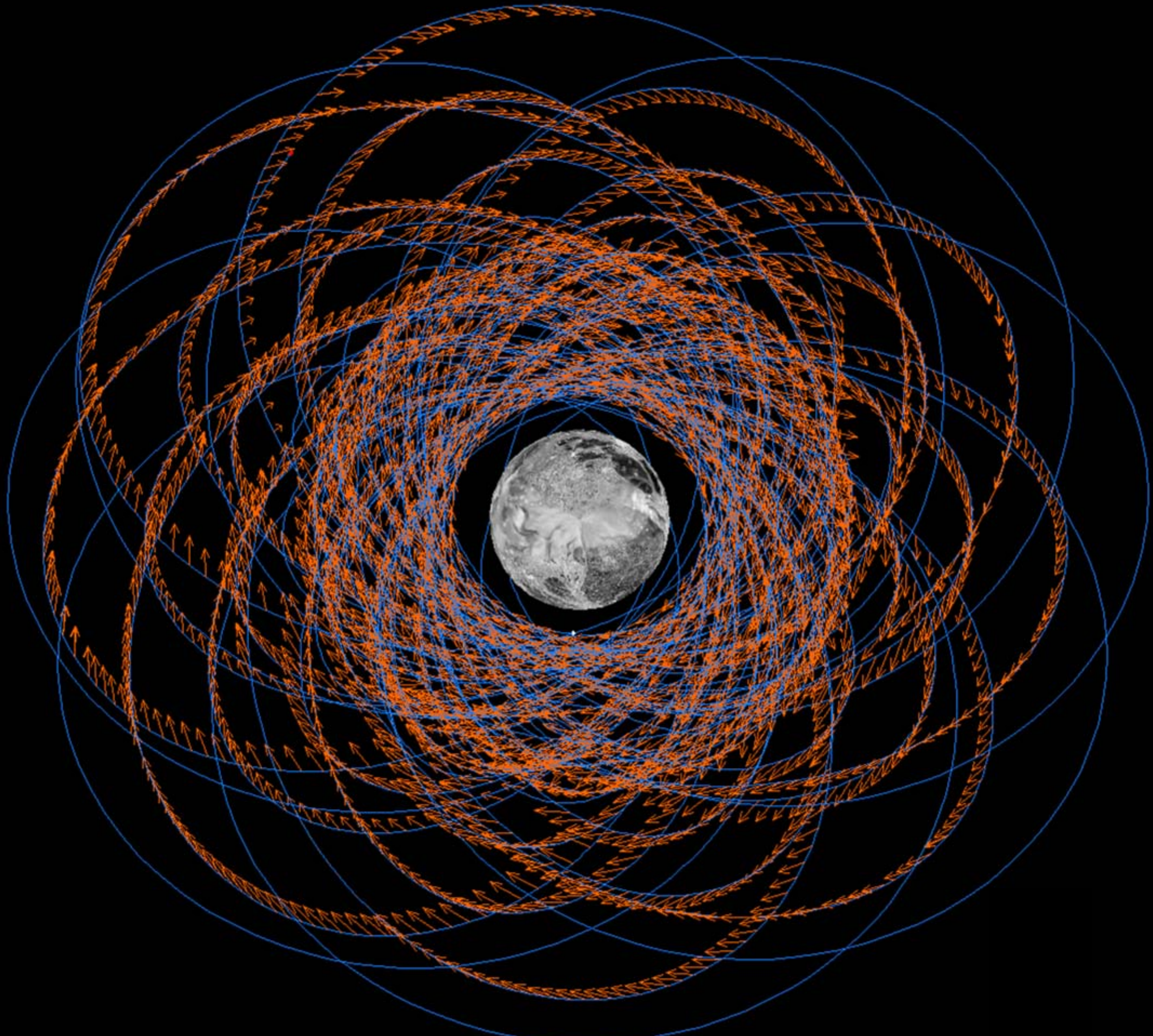
Optimal Trajectories as Art …

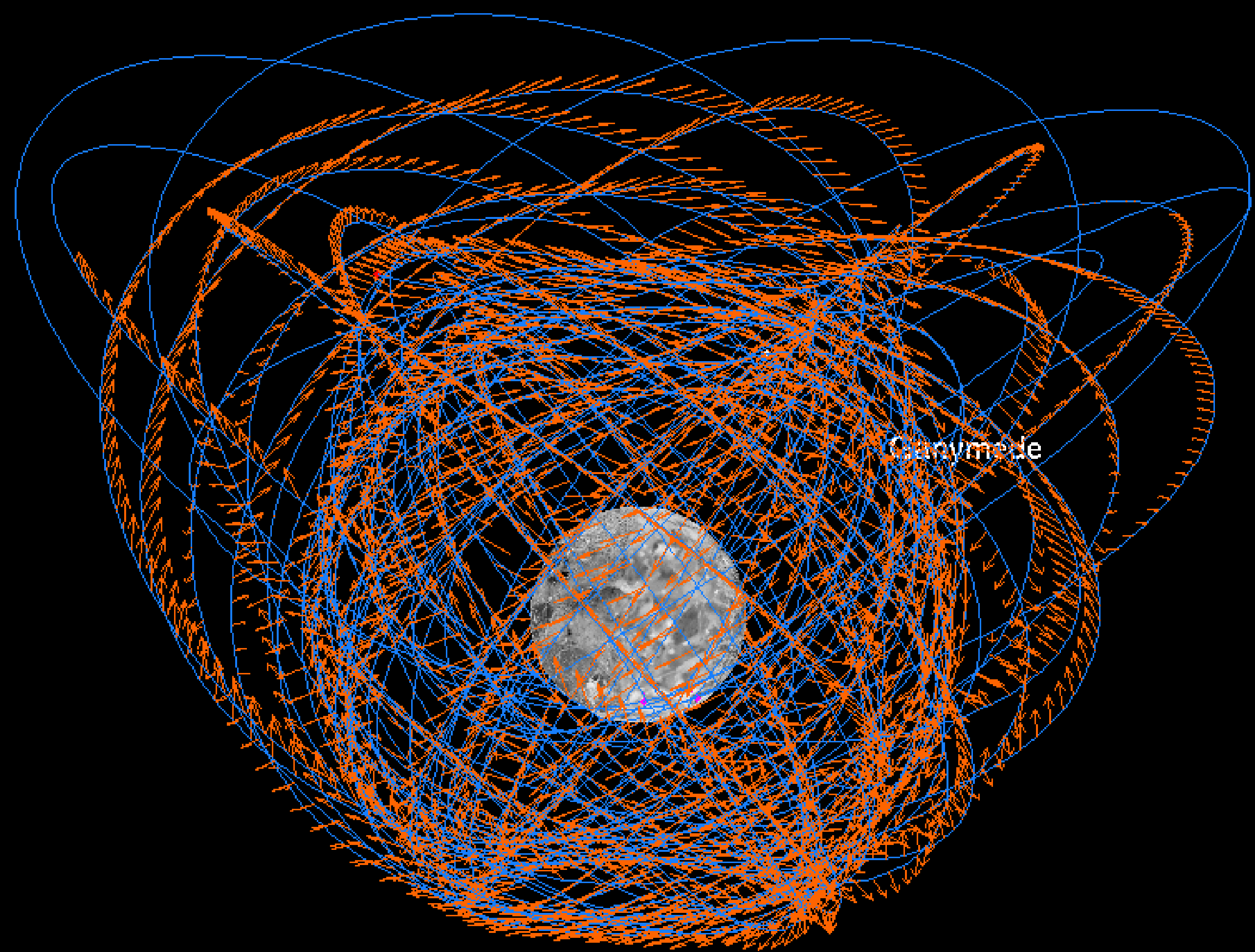


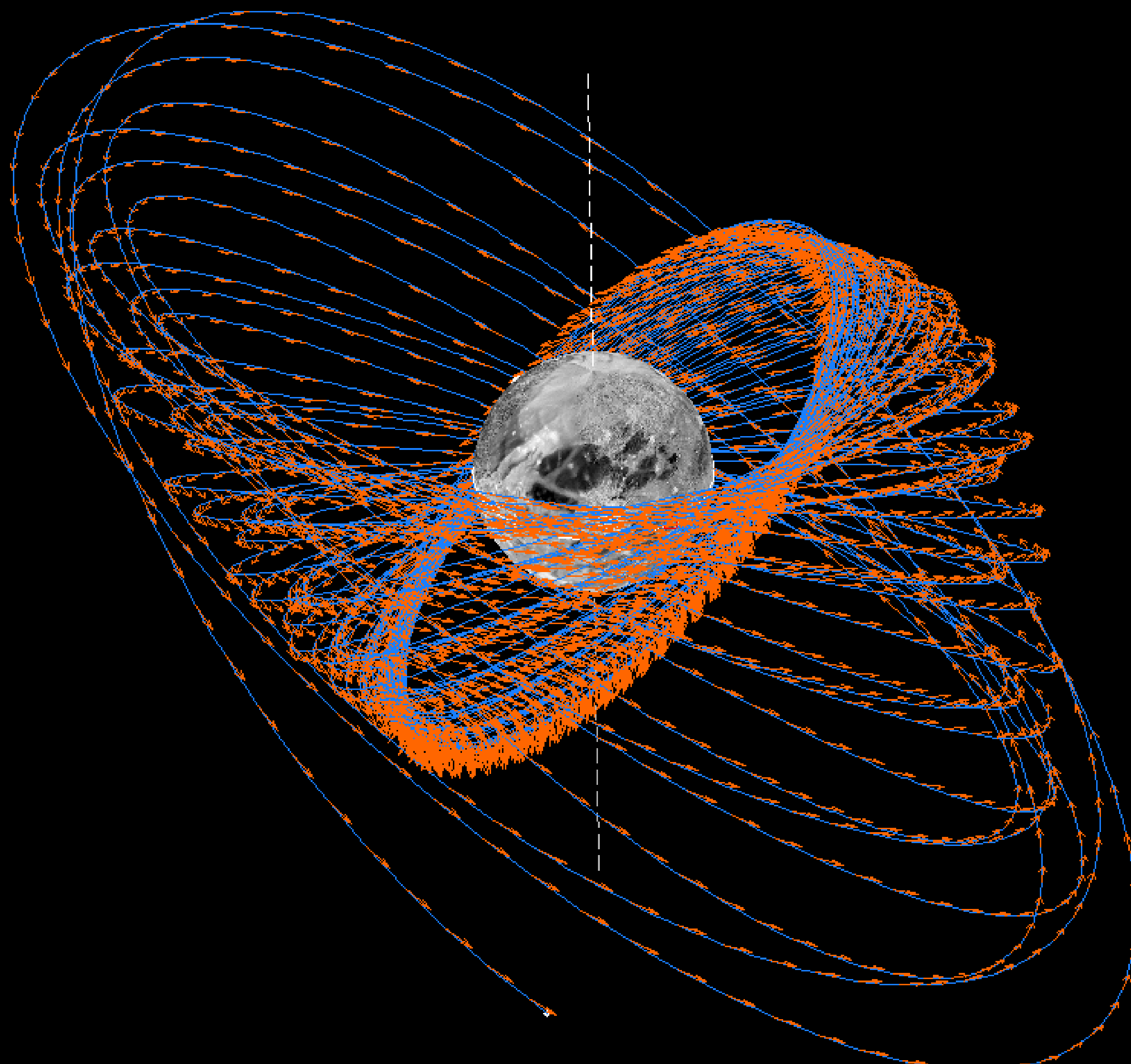


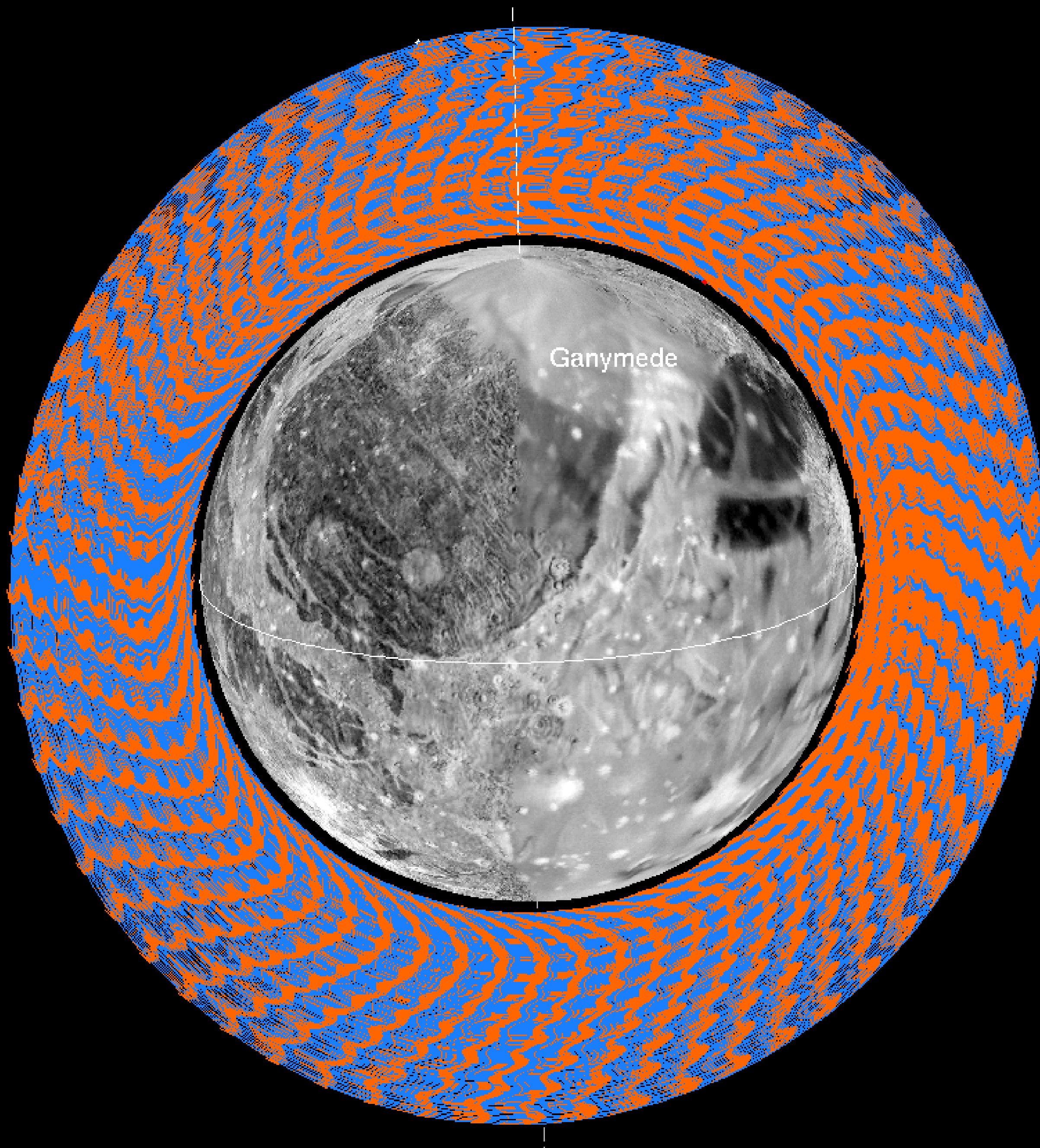


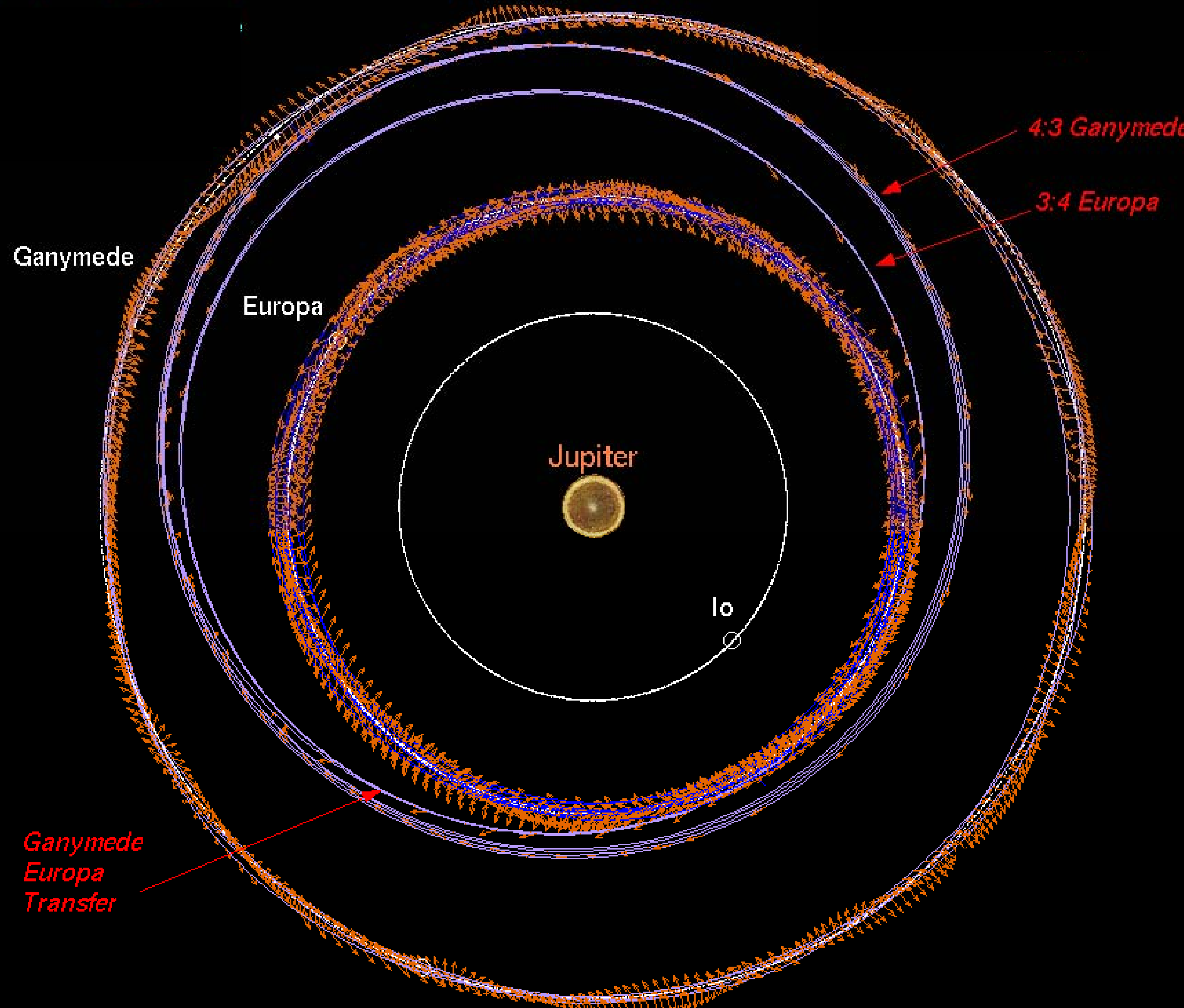

Jupiter
Io
Europa
Ganymede

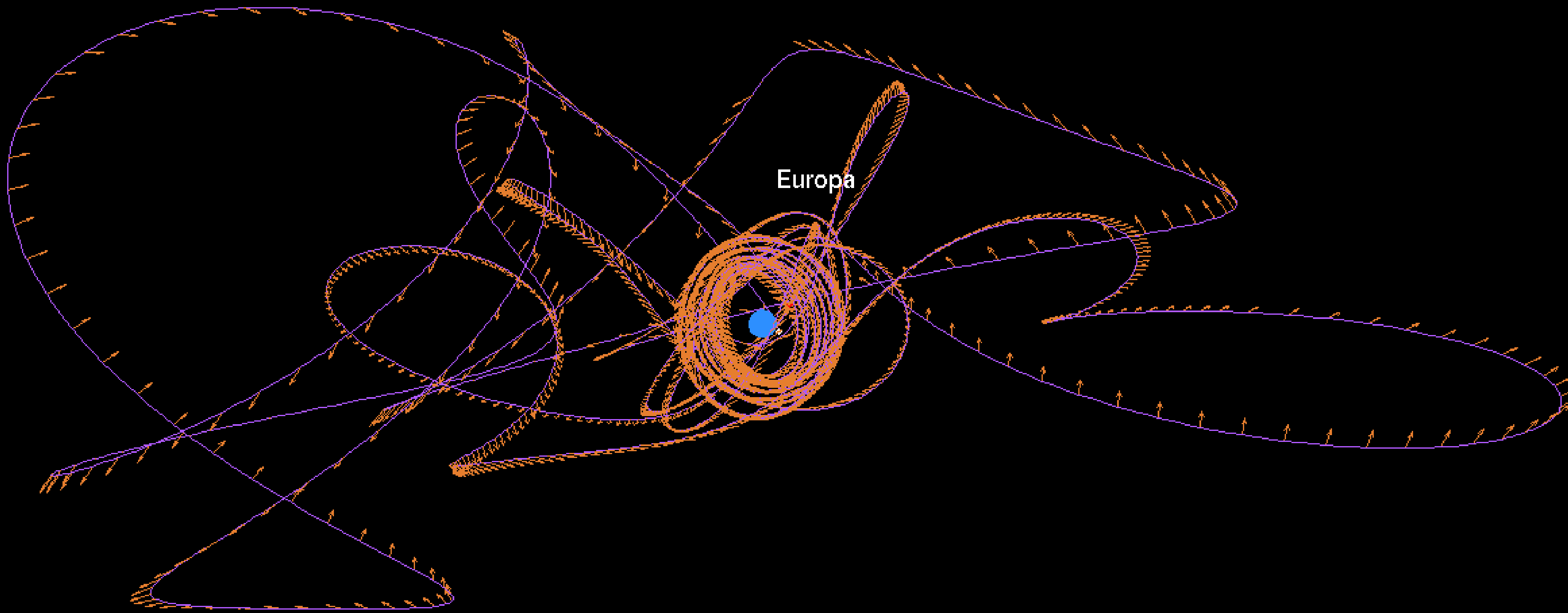


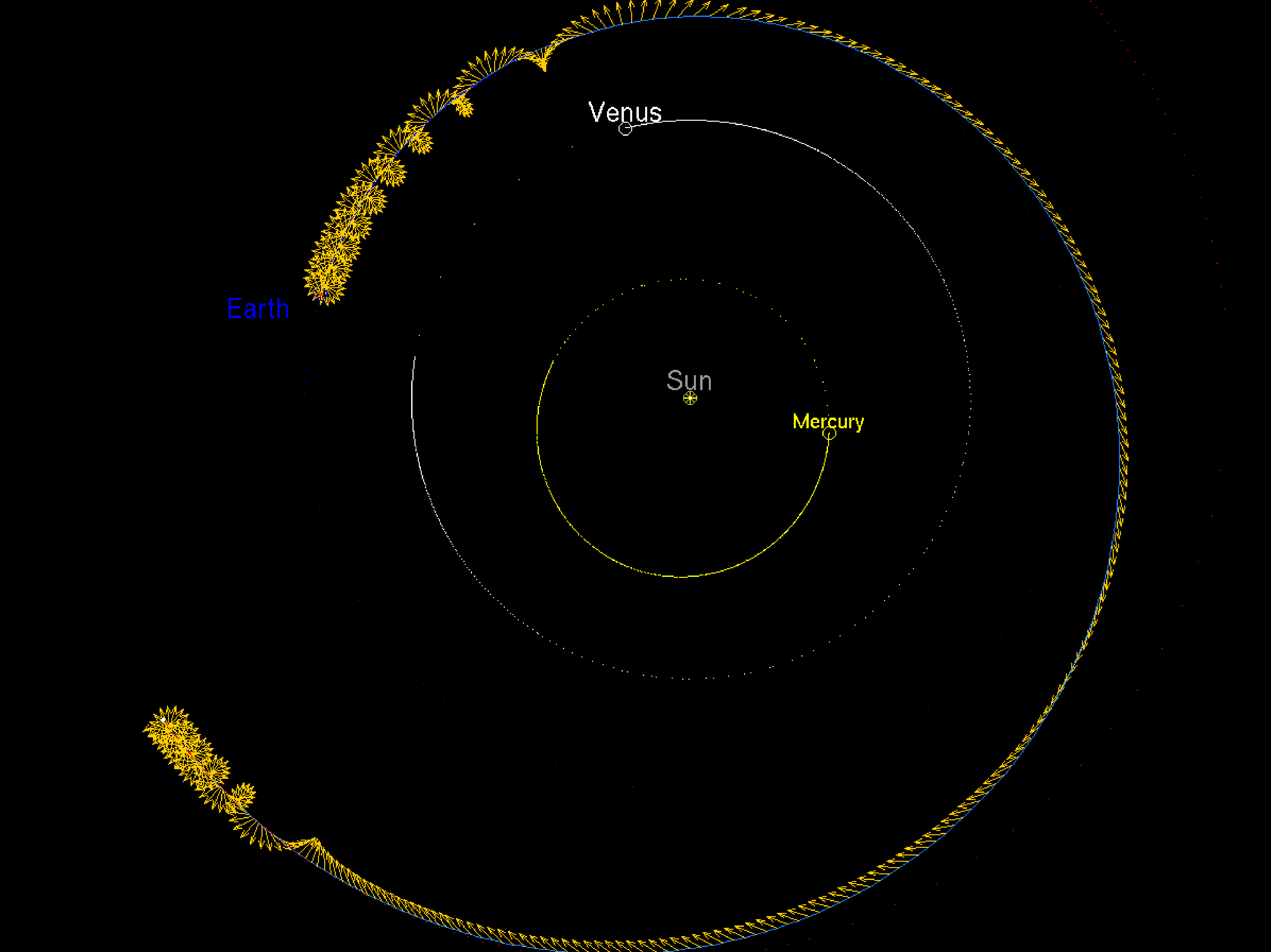

Ganymede

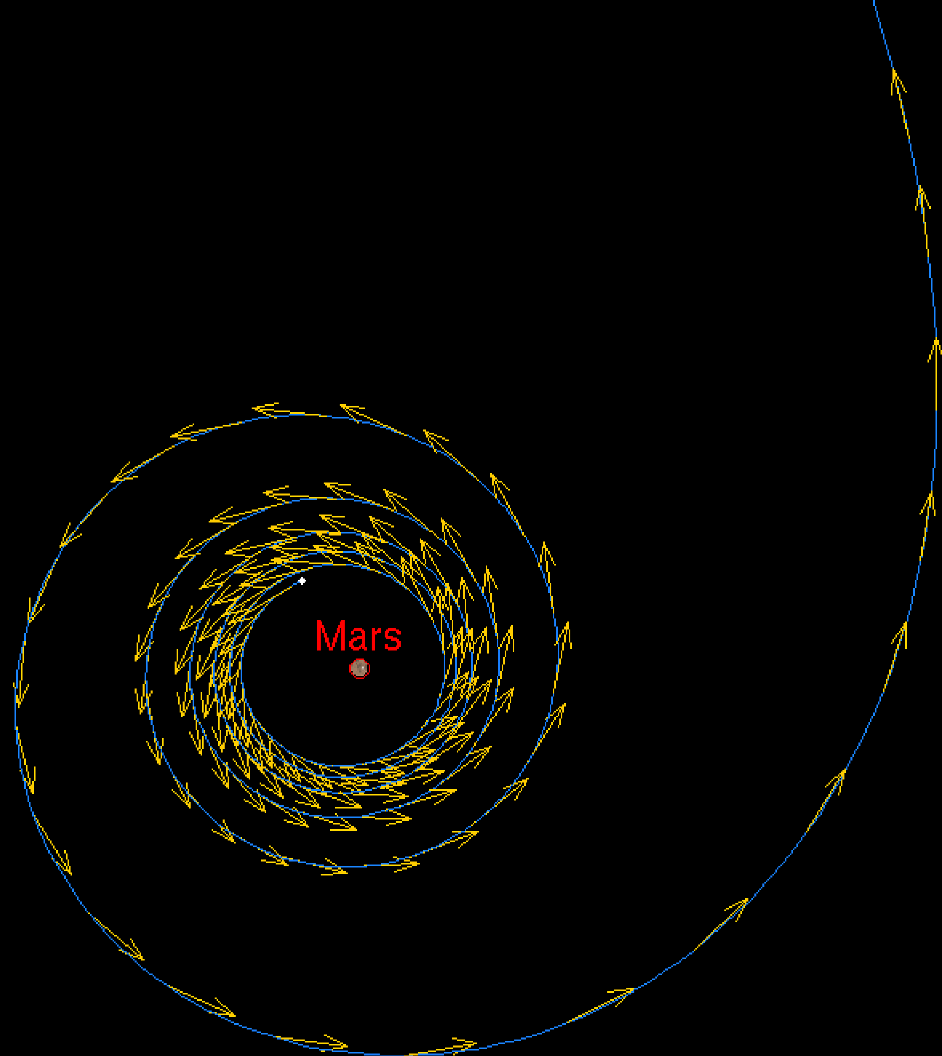


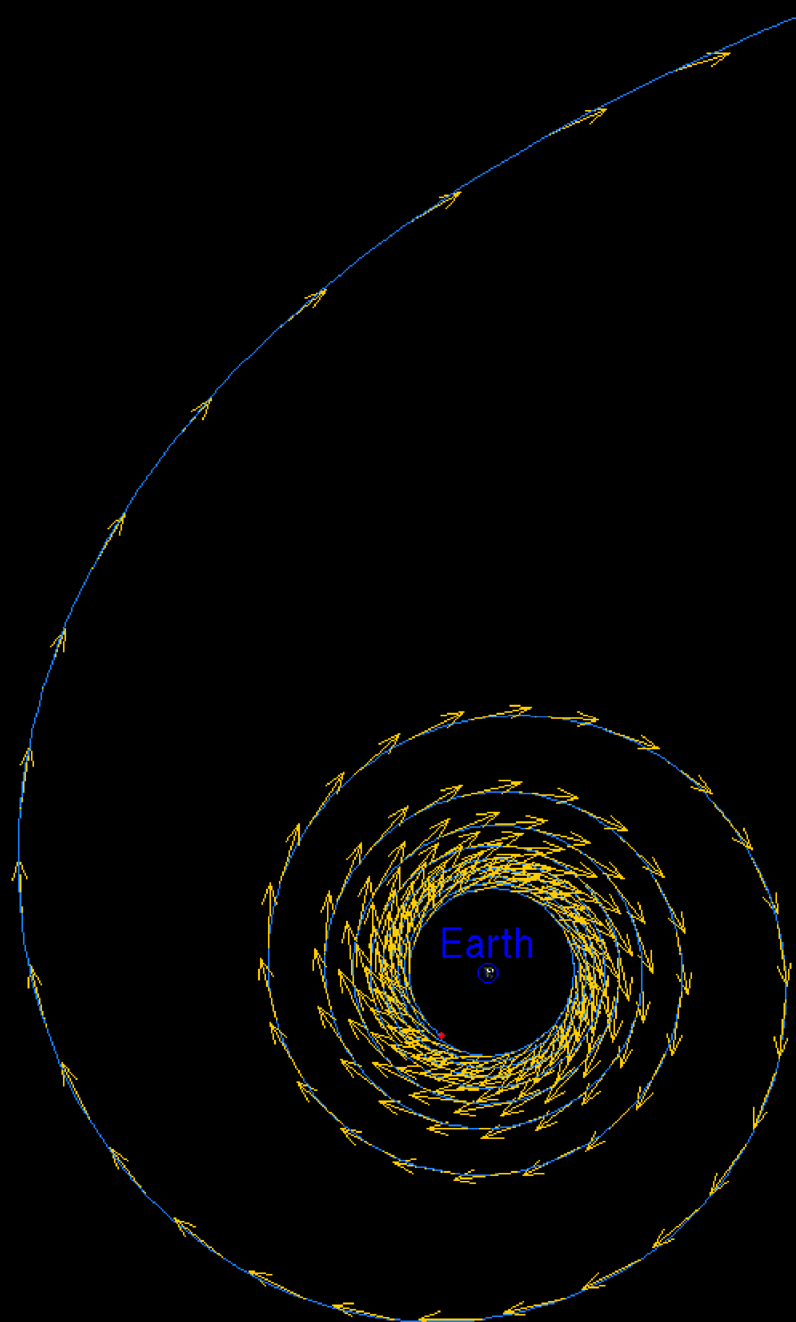

Ganymede

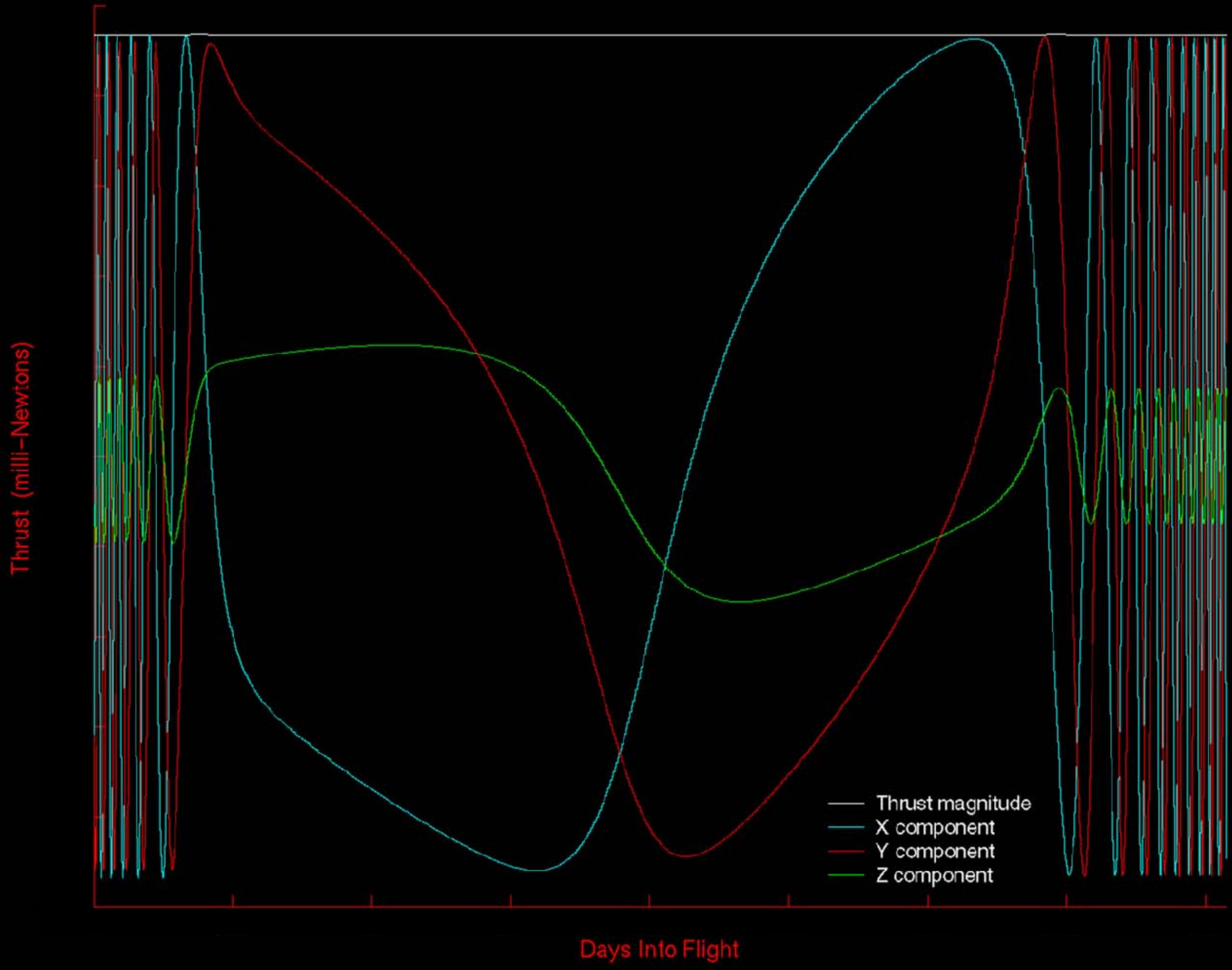

4:3 Ganymede
3:4 Europa
Ganymede
Europa
Jupiter
Io
Ganymede
Europa
Transfer


Europa


Venus
Earth
Sun
Mercury


Mars


Earth



Thrust (milli-Newtons)
Thrust magnitude
X component
Y component
Z component
Days Into Flight