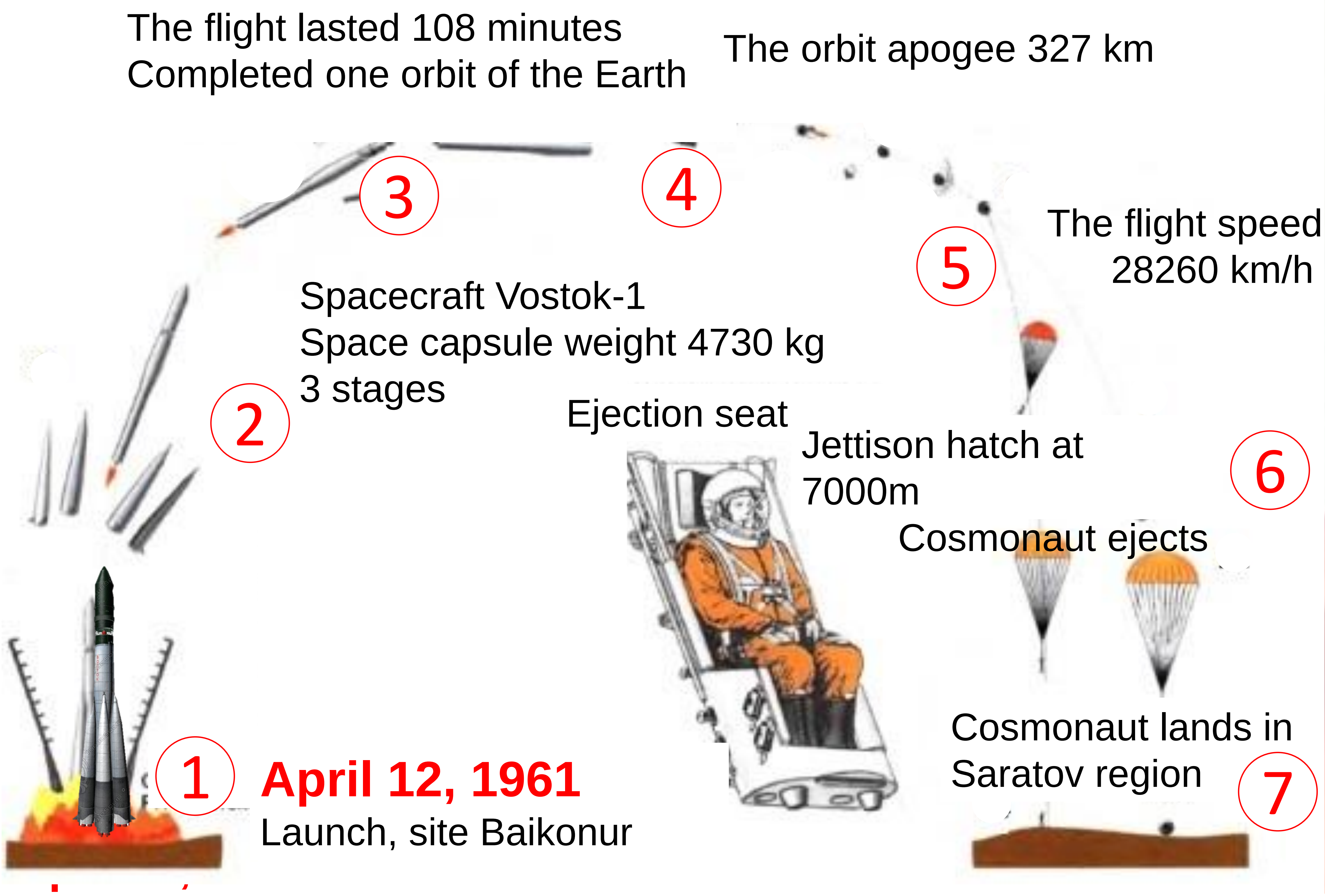
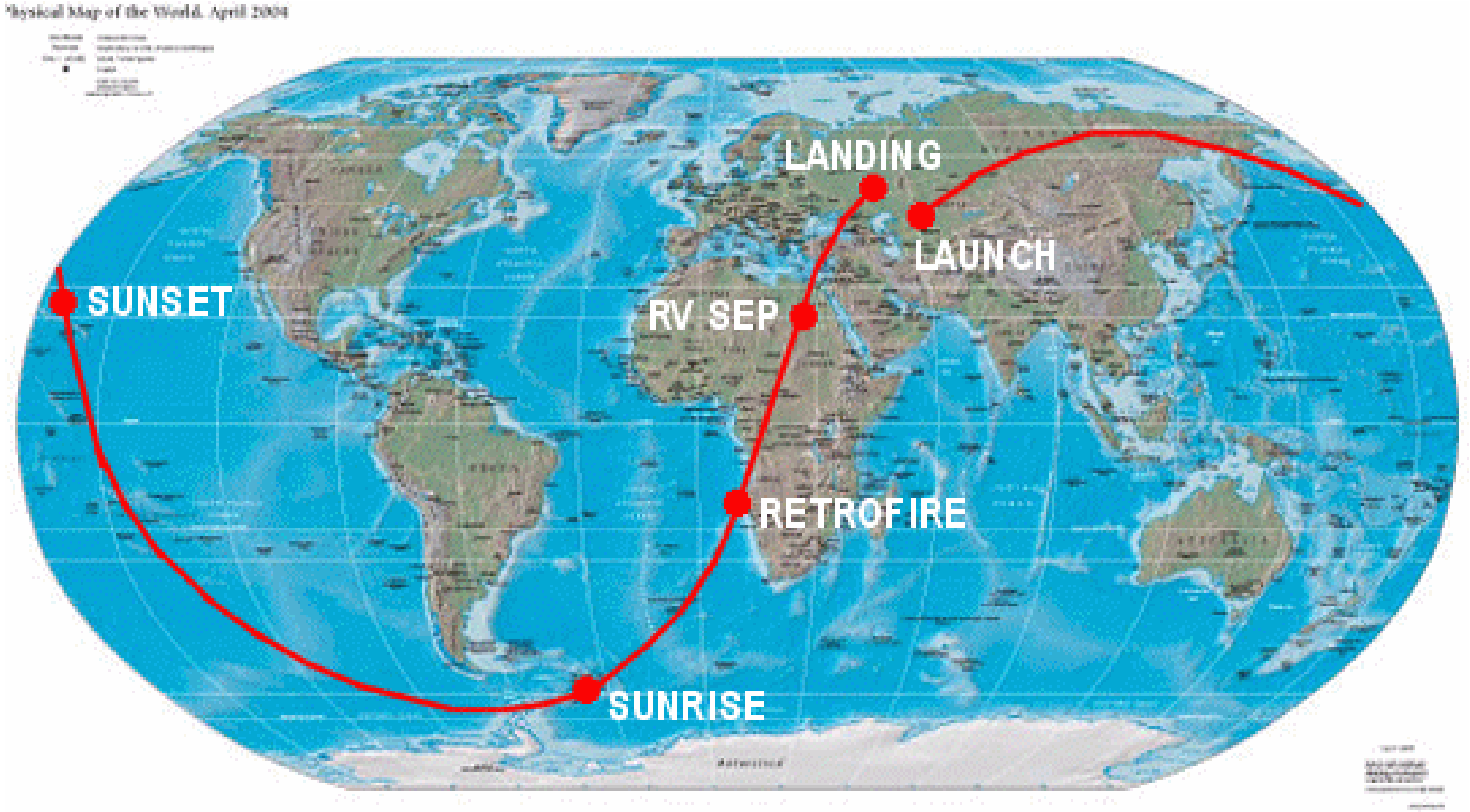
Space Mission Design and Operations

THE FIRST HUMAN SPACEFLIGHT IN HISTORY

Anniversary celebration Easter Sunday April 12, 2020
Cosmonaut Day and Yuri's Night



THE FIRST MANNED SPACEFLIGHT IN HISTORY



Achieving 2024 – A Parallel Path to Success

Artemis will see government and commercial systems moving in parallel to complete the architecture and deliver crew

NASA Programs SLS and Orion

Artemis 1

First flight test of SLS and Orion as an integrated system

Artemis 2

First flight of crew to the Moon aboard SLS and Orion

Artemis 3

First crew to the lunar surface;
Logistics delivered for 2024 surface mission

Between now and 2024, U.S. industry delivers the launches and human landing system necessary for a faster return to the Moon and sustainability through Gateway.

Commercially Provided Elements

Image © Spaceflightinsider.com 2019

PPE

Power Propulsion Element arrives at NRHO via commercial rocket

Crew Module

Small pressurized crew module launches to Gateway on a commercial rocket

Human Landing System

Descent Element

Descent Element launches to Gateway on commercial rocket

Transfer Element

Transfer Element launches to Gateway on commercial rocket

Ascent Element

Ascent Element launches to Gateway on commercial rocket

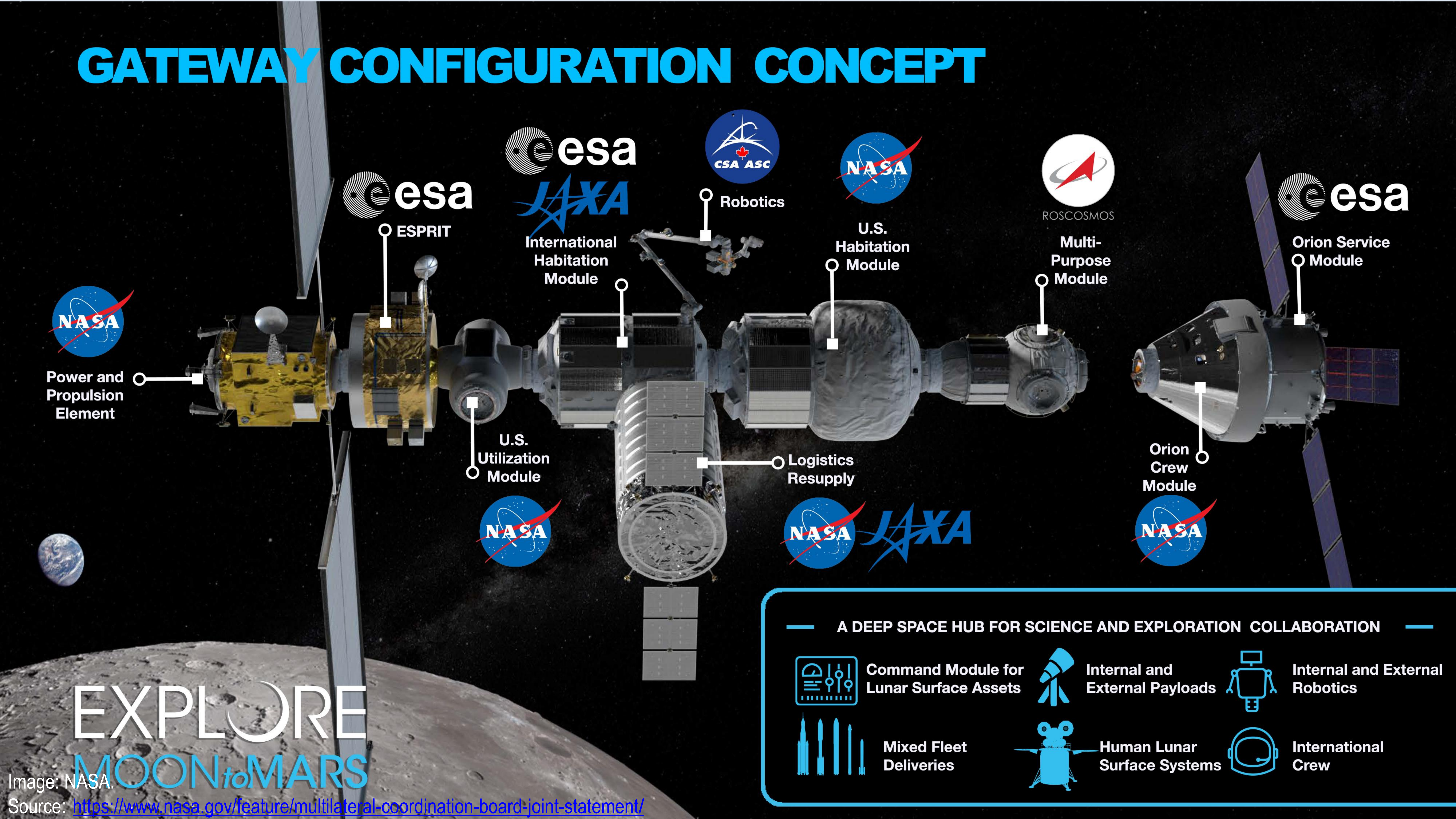


Image: NASA.
Source: https://www.nasa.gov/feature/multilateral-coordination-board-joint-statement/

Dragon XL will supply the Gateway - Gateway Logistics Services

- Gateway is a key part of NASA's Artemis exploration program.
- SpaceX will help to keep the Gateway supplied, delivering scientific experiments and a variety of other gear.
- SpaceX's Gateway missions provide: the huge Falcon Heavy rocket and a special capsule variant - Dragon XL.
- Dragon XL will carry more than 5 metric tons of cargo to the Gateway.
- Dragon XL will stay attached to the Gateway for 6-12 months.
- Gateway Logistic Service contract duration: 12 years with 15 year performance period.
- Contract price ~ $7 billion.

Artist's illustration of the SpaceX Dragon XL as it is deployed from the Falcon Heavy's second stage in high Earth orbit on its way to the Gateway in lunar orbit. (Image: © SpaceX)

Source: https://www.space.com/spacex-wins-cargo-contract-nasa-gateway-moon-station.html

Space Mission Design and Operations

Sections 4.2 to 4.5

Prof. Claude Nicollier

April 6 and 8, 2020

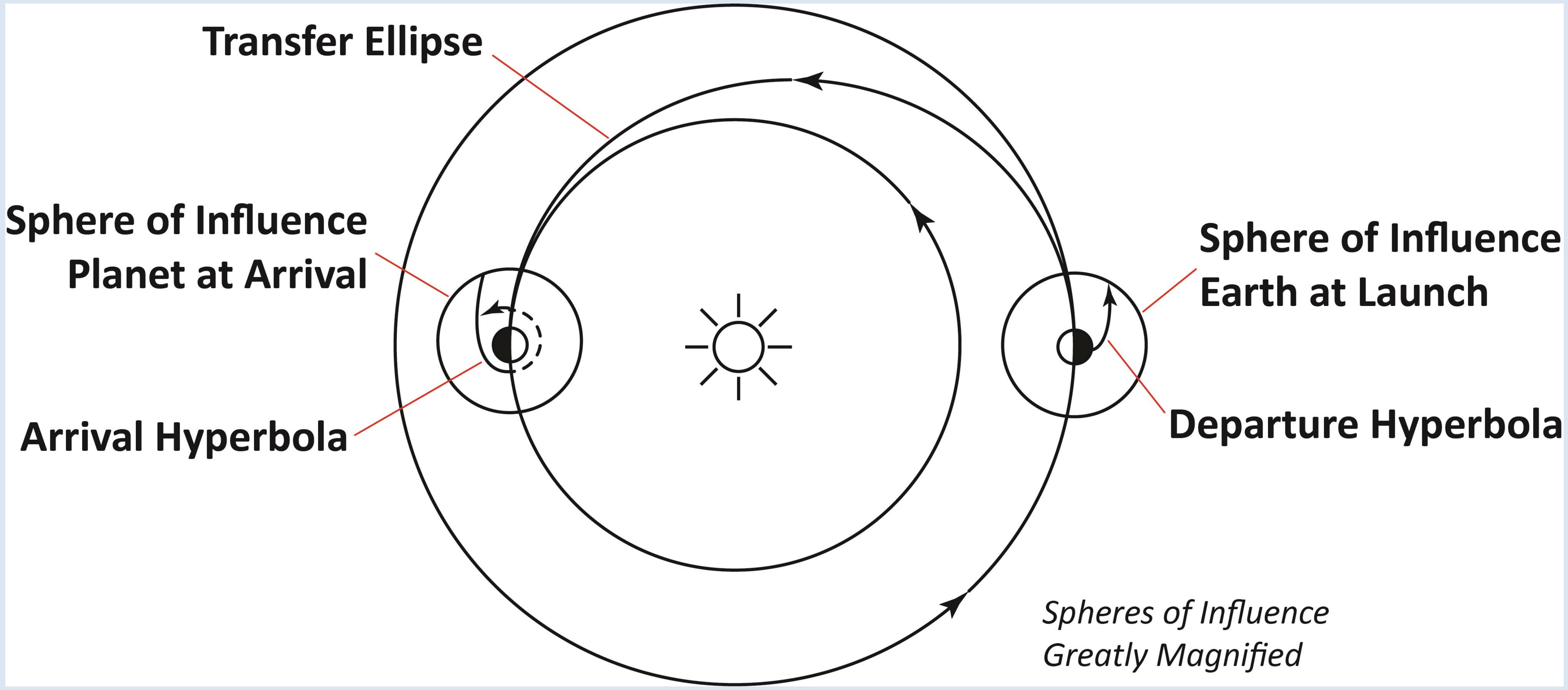
Outline

4.2 Interplanetary trajectories

4.3 Aerodynamic braking and slingshot maneuvers

4.4 Spacecraft propulsion

4.5 Ascent to space, and re-entry



4.2.1 Introduction to interplanetary trajectories

Space Mission Design and Operations

Prof. Claude Nicollier

Credits: Charles D. Brown, *Elements of Spacecraft Design*, AIAA

The Astronomical Unit (AU)

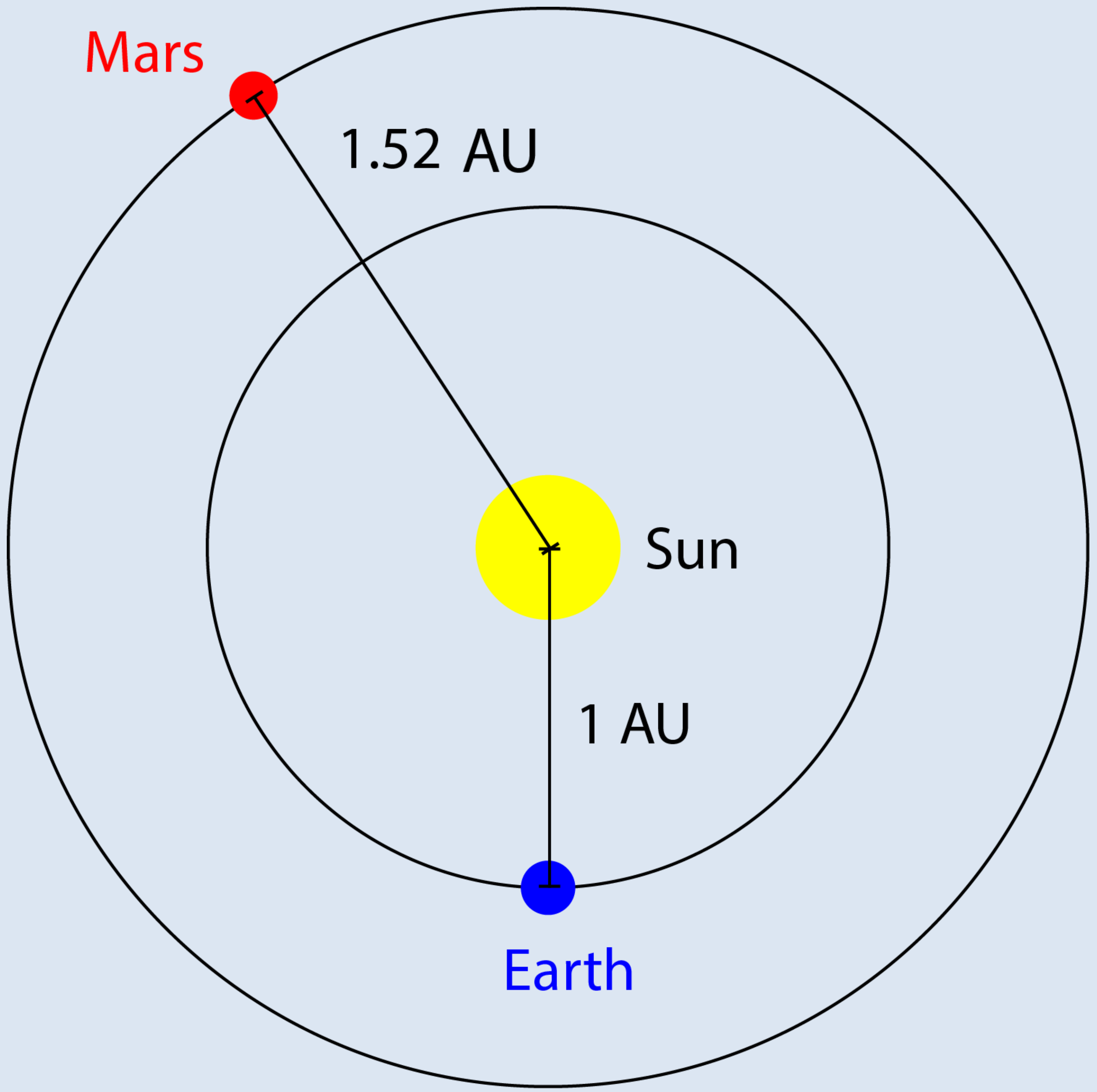
Astronomical unit = average distance Sun - Earth.

The orbit of the Earth around the Sun is slightly elliptical, eccentricity 0.017 at this time.

At perihelion on January 3rd, the Earth is about 147 million km to the Sun, at aphelion, on July 3rd, the distance is 152 million km to the Sun.

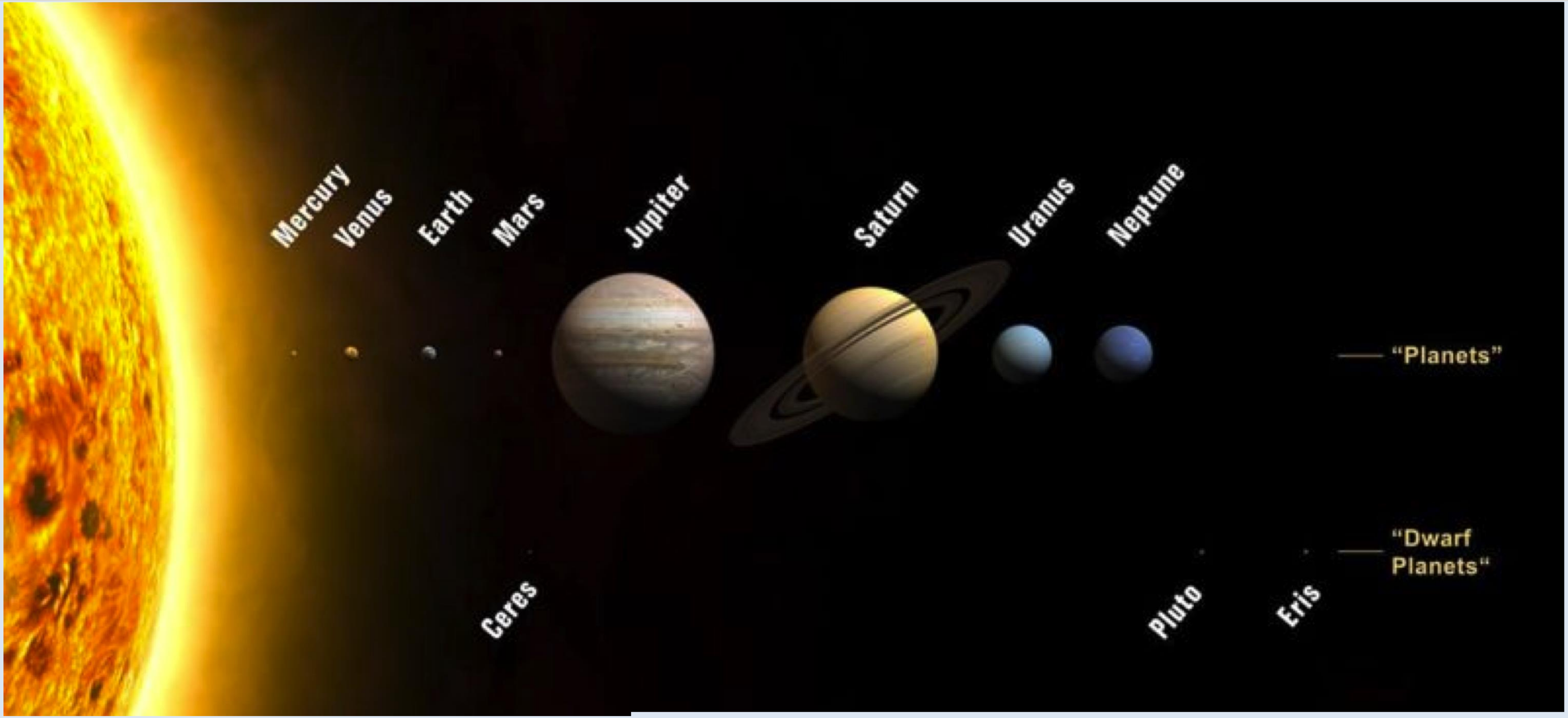
Mars has a 1.52 AU average distance to the Sun, which means that its distance to Earth varies from 0.52 AU to 2.52 AU on the average. Its orbit has a large eccentricity of 0.094 at this time.

The boundary of the planetary component of the Solar System is about 30 AU from the Sun (orbit of Neptune)



1 AU = 149.5978707 x 10^6 km

The solar system



Credits: NASA, JPL

Orbital characteristics of planets

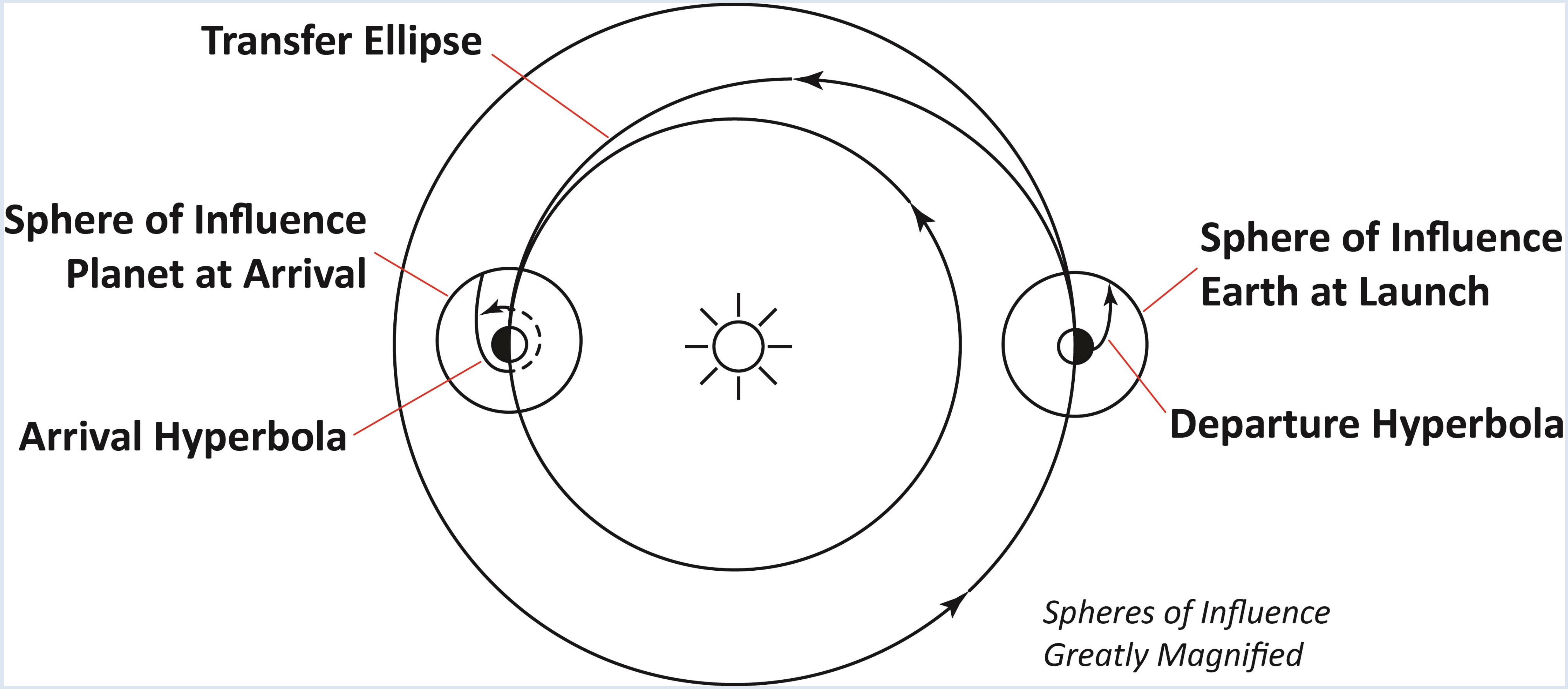
Planets	Semi-major axis a (AU)	Perihelion r_p (10^6 km)	Orbital eccentricity e	Orbital inclination i (deg)	Orbital velocity V $\left(\frac{km}{s}\right)$
Mercury	0.39	46.0	0.205	7.0	47.4
Venus	0.72	107.5	0.007	3.4	35.0
Earth	1.00	147.1	0.017	0.0	29.8
Mars	1.52	206.6	0.094	1.9	24.1
Jupiter	5.20	740.5	0.049	1.3	13.1
Saturn	9.65	1353.6	0.057	2.5	9.7
Uranus	19.20	2741.3	0.046	0.8	6.8
Neptune	30.04	4444.5	0.011	1.8	5.4

http://nssdc.gsfc.nasa.gov/planetary/factsheet/

Credits: NASA

Interplanetary trajectories – Patched conics approximation

In order to plan for and execute a mission to another planet, we consider the Sun, the planet of departure (the Earth), the planet of destination, and the spacecraft. It is a four-body problem that we divide into three segments, each of them a two-body problem.



- Departure phase (planetocentric 1)
- Cruise phase (heliocentric)
- Arrival phase (planetocentric 2)

Credits: Charles D. Brown, *Elements of Spacecraft Design*, AIAA

The concept of sphere of influence

- Sphere around each planet inside which the motion of a spacecraft is considered to be two-body Keplerian.
- The radius of the sphere of influence R_S has been determined by Laplace as:

$$\boxed{R_S = R\left(\frac{\mu_{\text{Planet}}}{\mu_{\text{Sun}}}\right)^{\frac{2}{5}}}$$

- R is the average distance between Sun and the planet.

As long as the spacecraft is within the sphere of influence of the Earth, its motion with respect to the Earth is a two-body problem with the Earth as a central body. We can ignore the Sun's gravitational influence.

When the spacecraft leaves the sphere of influence of the Earth, which happens to be about 1,000,000 km in radius, it comes on a heliocentric elliptical trajectory (Hohmann transfer) towards the destination planet, either larger than the Earth's orbit (outer planets) , or smaller (inner planets). The gravitational influence of both the departure and the destination planets is negligible on this heliocentric arc.

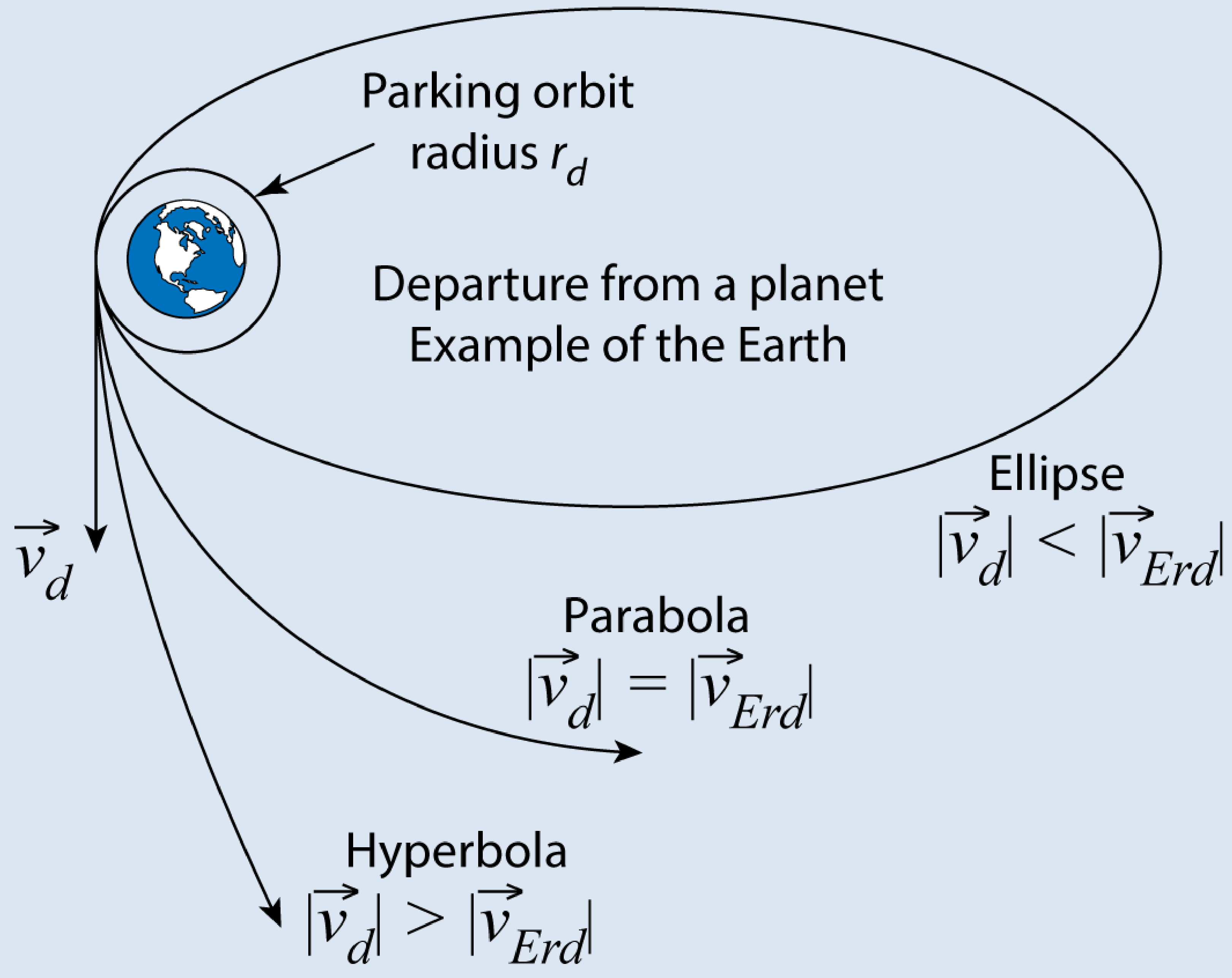
At the end its elliptic heliocentric arc, in the vicinity of the destination planet, the spacecraft enters the sphere of influence of the destination planet, then we can ignore the Sun and determine the spacecraft's trajectory as a two-body problem with the destination planet as the main and only attracting body.

Spheres of influence in the solar system

Planets	R_S (10^6 km)
Mercury	0.111
Venus	0.616
Earth	0.924
Mars	0.577
Jupiter	48.157
Saturn	54.796
Uranus	51.954
Neptune	80.196
Moon	0.0662

- The concept of the sphere of influence is usable for the motion of a spacecraft from the Earth to another planet
- For the Moon, the sphere of influence is calculated with the Laplace equation with R = distance Earth-Moon. It means is that within about 66'200 km from the Moon center, motion of a spacecraft is essentially dominated by the Moon's gravitational attraction.
- For all planets beyond Mercury, R_S is really large and will often be considered as being a location of zero potential energy with respect to the central body.

Credits: Charles D. Brown, *Elements of Spacecraft Design*, AIAA



4.2.2 Departure from a planet

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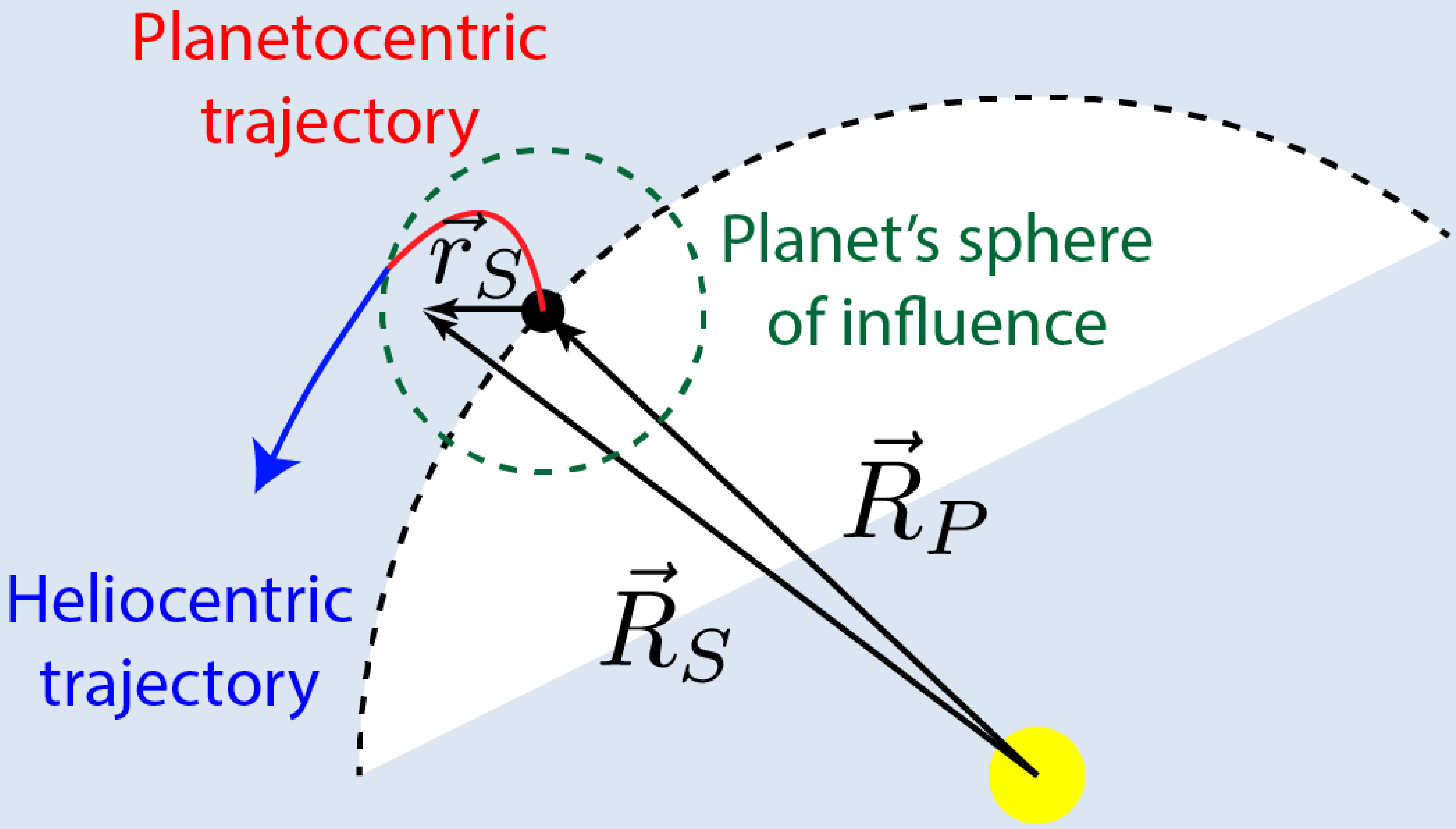
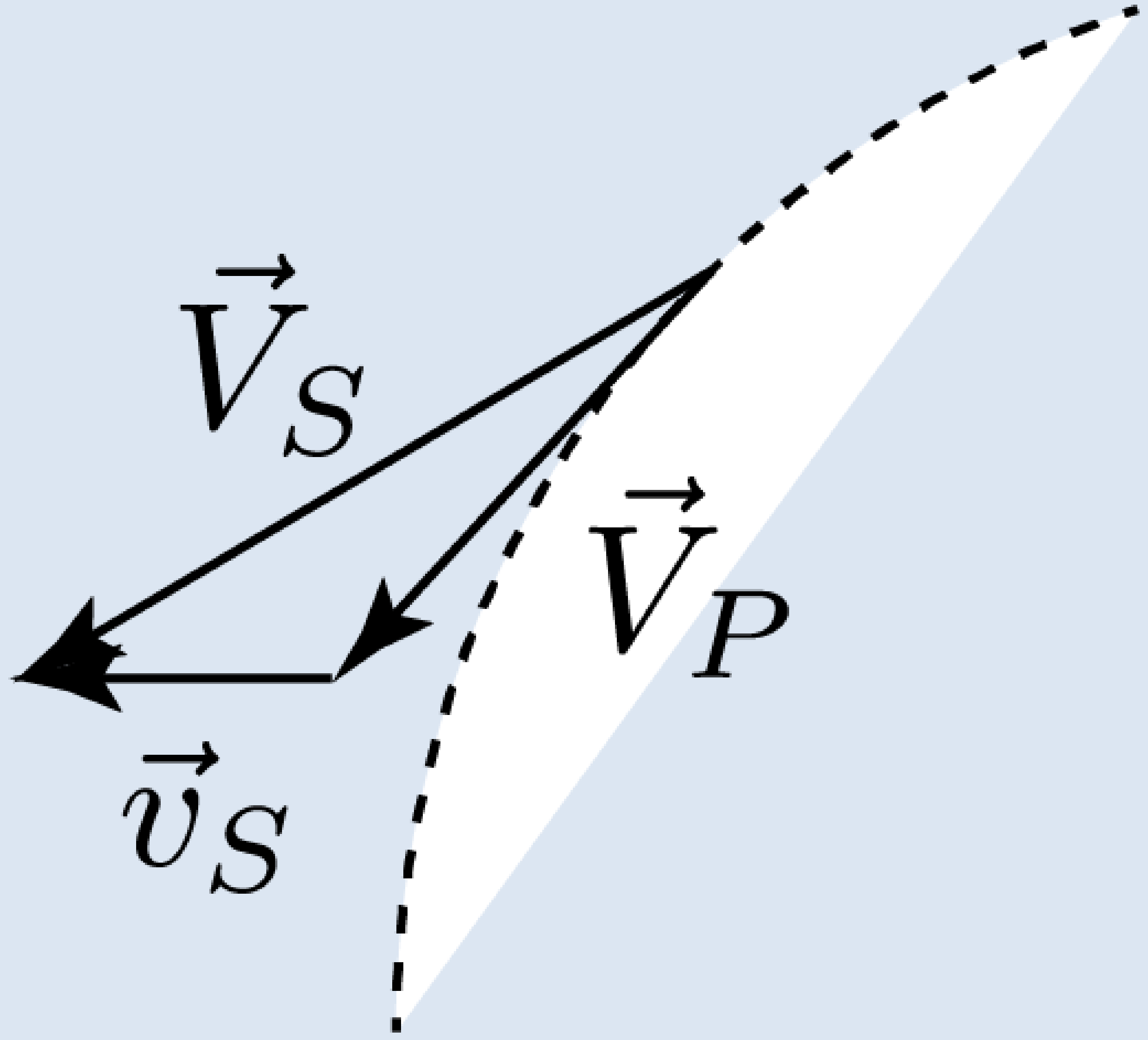
Symbol convention for position and velocity



P for Planet
S for Spacecraft



Symbol convention for position and velocity

$$\boxed{\begin{aligned} \vec{R}_S &= \vec{r}_S + \vec{R}_P \\ \vec{V}_S &= \vec{v}_S + \vec{V}_P \end{aligned}}$$

$$\boxed{\begin{aligned} \vec{r}_S &= \vec{R}_S - \vec{R}_P \\ \vec{v}_S &= \vec{V}_S - \vec{V}_P \end{aligned}}$$

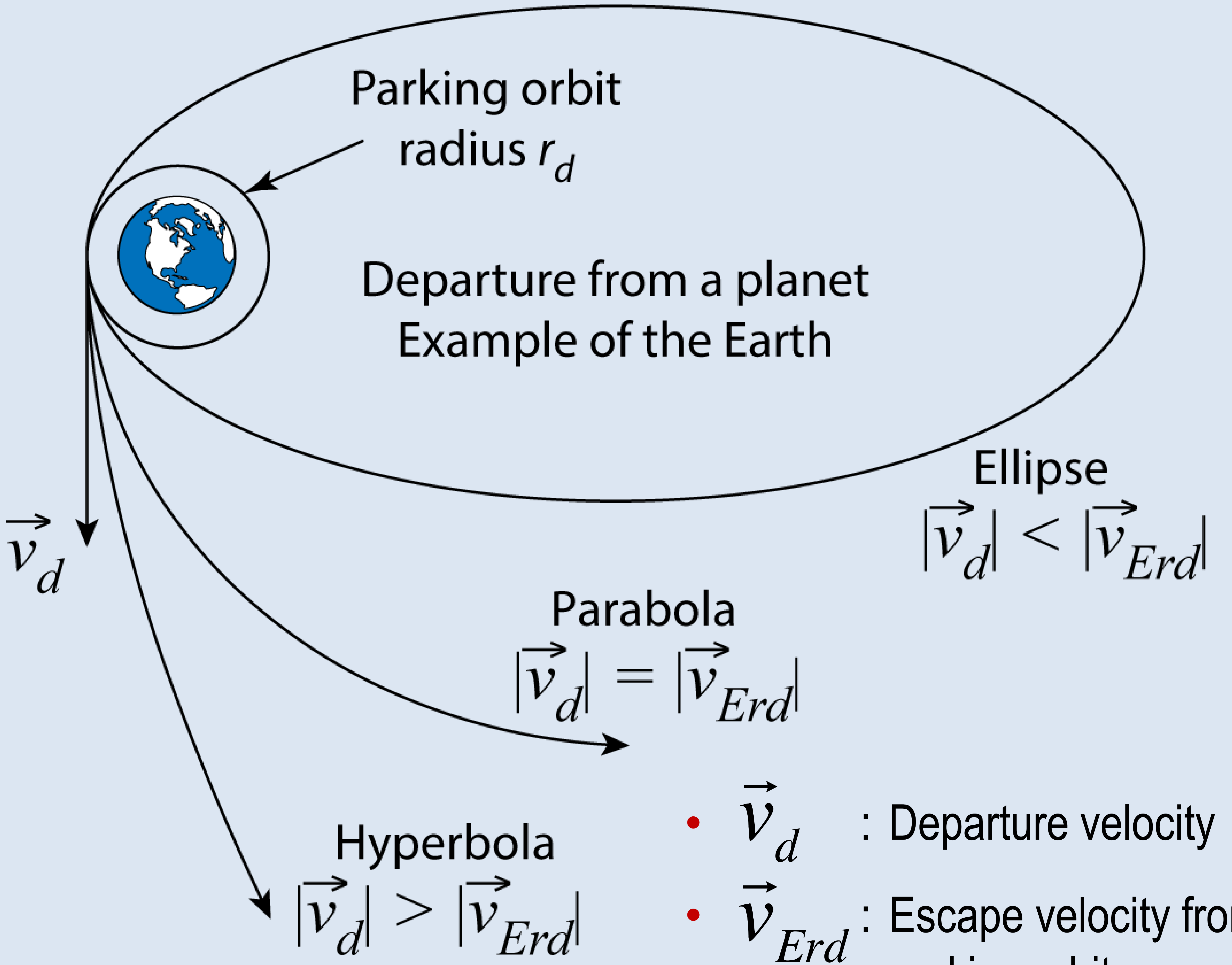
Heliocentric movement of the spacecraft	Planetocentric movement of the spacecraft	Heliocentric movement of the planet

Planetocentric movement of the planet	Difference between the heliocentric movement of the spacecraft and the planet

Departure from a planet

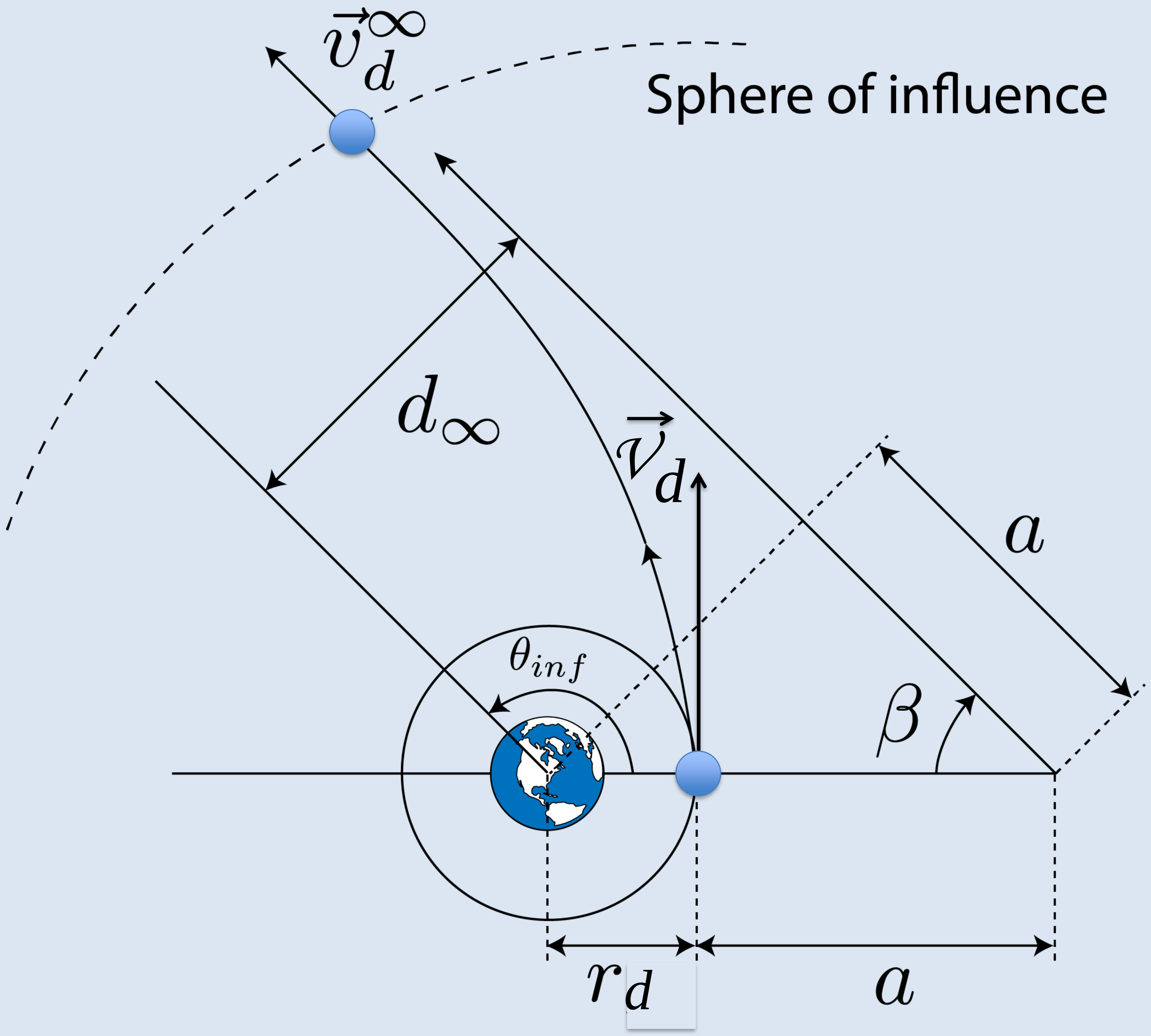
A journey to a destination in the solar system always starts with a parking orbit around Earth. Then the orbital velocity in increased to reach the **departure velocity** which is always larger than the escape velocity for the altitude of the parking orbit. The escape velocity is about 11.2 km/s at the surface, and decreases with the altitude.

On a planetocentric hyperbolic departure orbit, at a large distance from the Earth, the spacecraft comes to a certain constant velocity called the **hyperbolic excess velocity**.



- $\vec{v}_d$: Departure velocity
- $\vec{v}_{Erd}$: Escape velocity from the parking orbit

Reaching the sphere of influence



$$r_d = r_{\text{departure}} = r_{\text{perigee}}$$

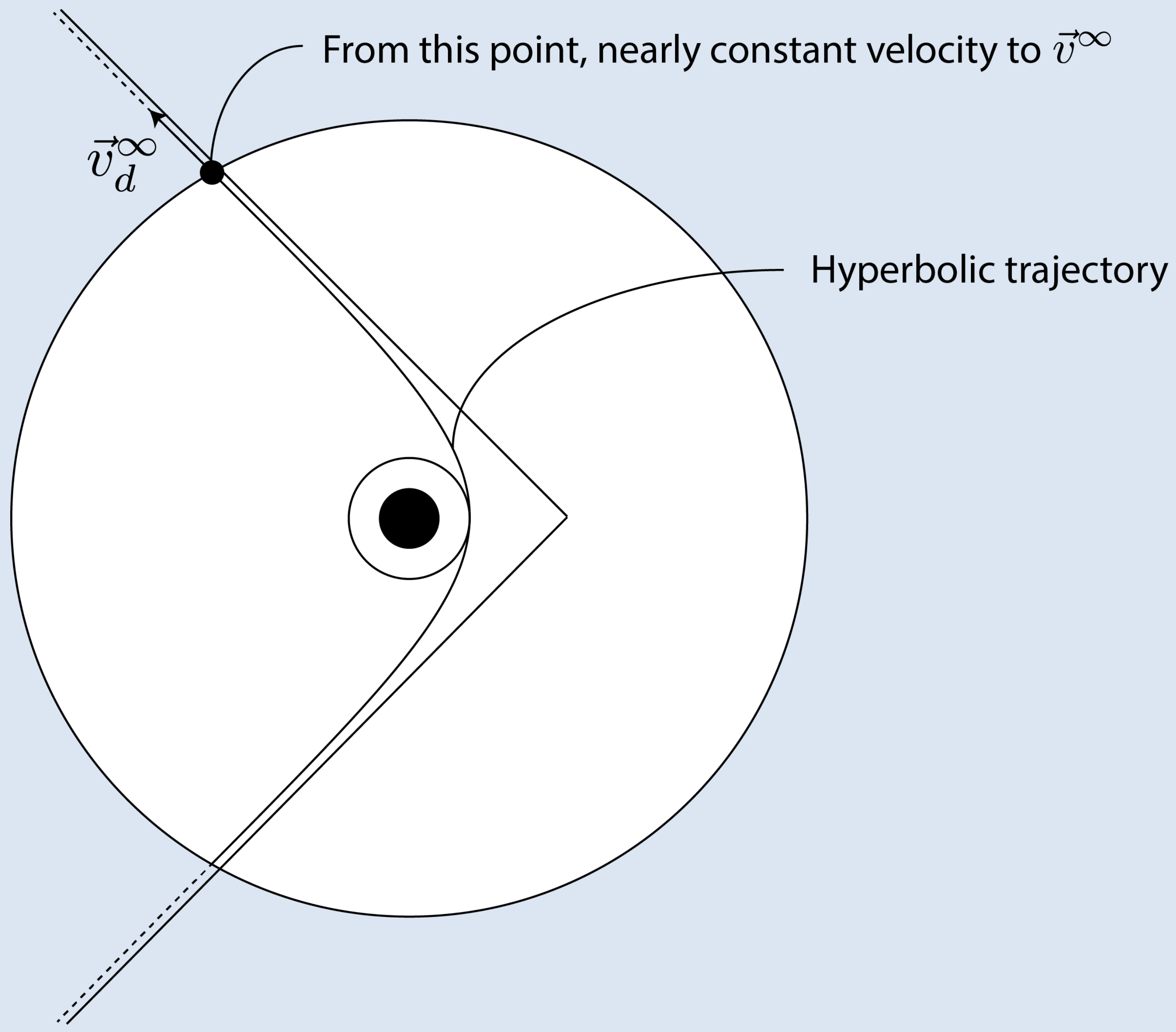
Conservation of total mechanical energy $\rightarrow$

$$\frac{(v_d^\infty)^2}{2} - \left(\frac{\mu}{r}\right)_\infty = \frac{v_d^2}{2} - \frac{\mu}{r_d}$$

$$\frac{(v_d^\infty)^2}{2} \cong \frac{v_d^2}{2} - \frac{v_{Erd}^2}{2}$$

$$\boxed{v_d^2 = (v_d^\infty)^2 + v_{Erd}^2}$$

- Reminder
  - $v_{E\ \text{Surface}}$ is 11.2 km/s for Earth
  - v_{Erd} is escape velocity at distance r_d



4.2.3 More on departure from a planet

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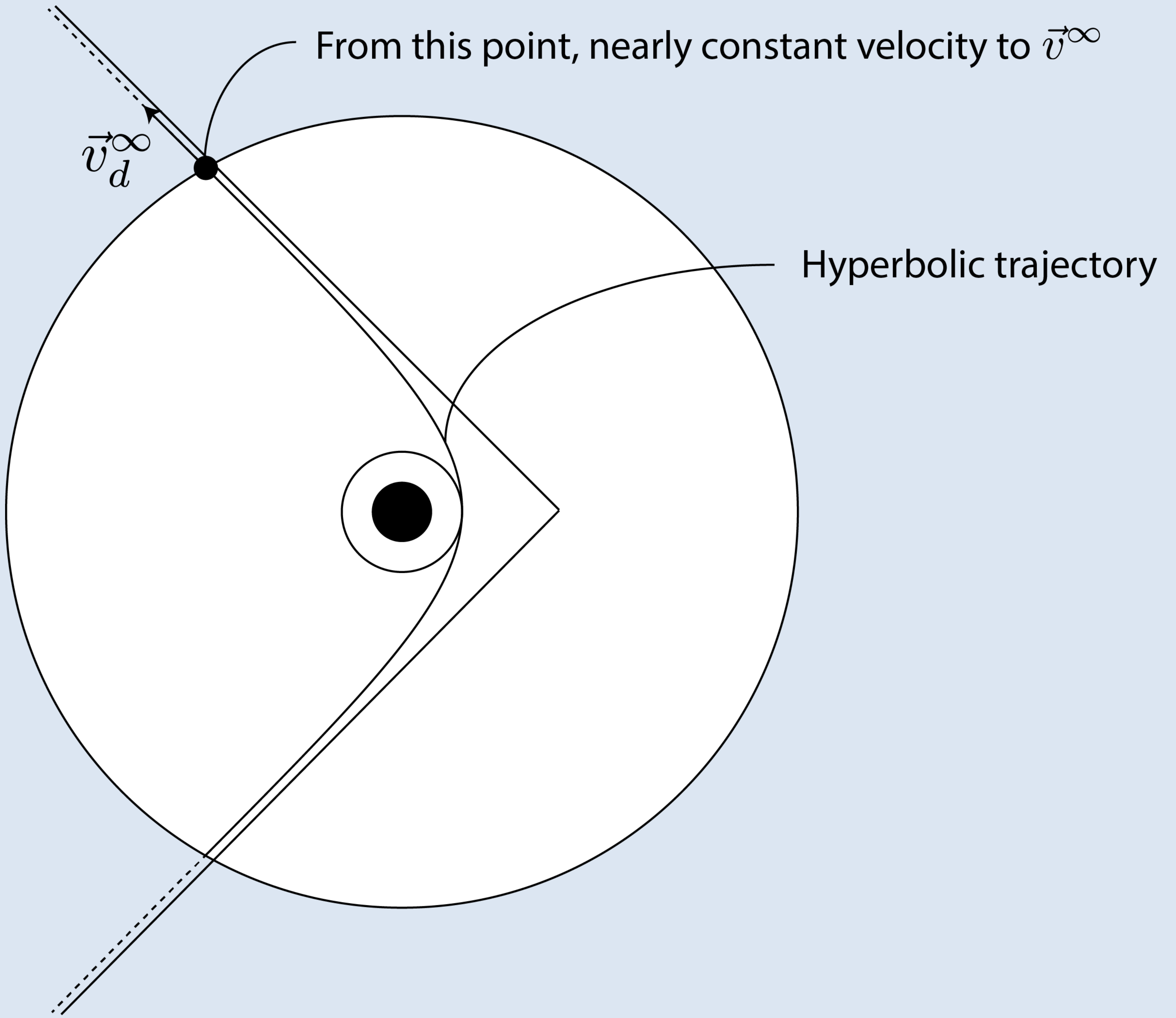
Departure from a planet



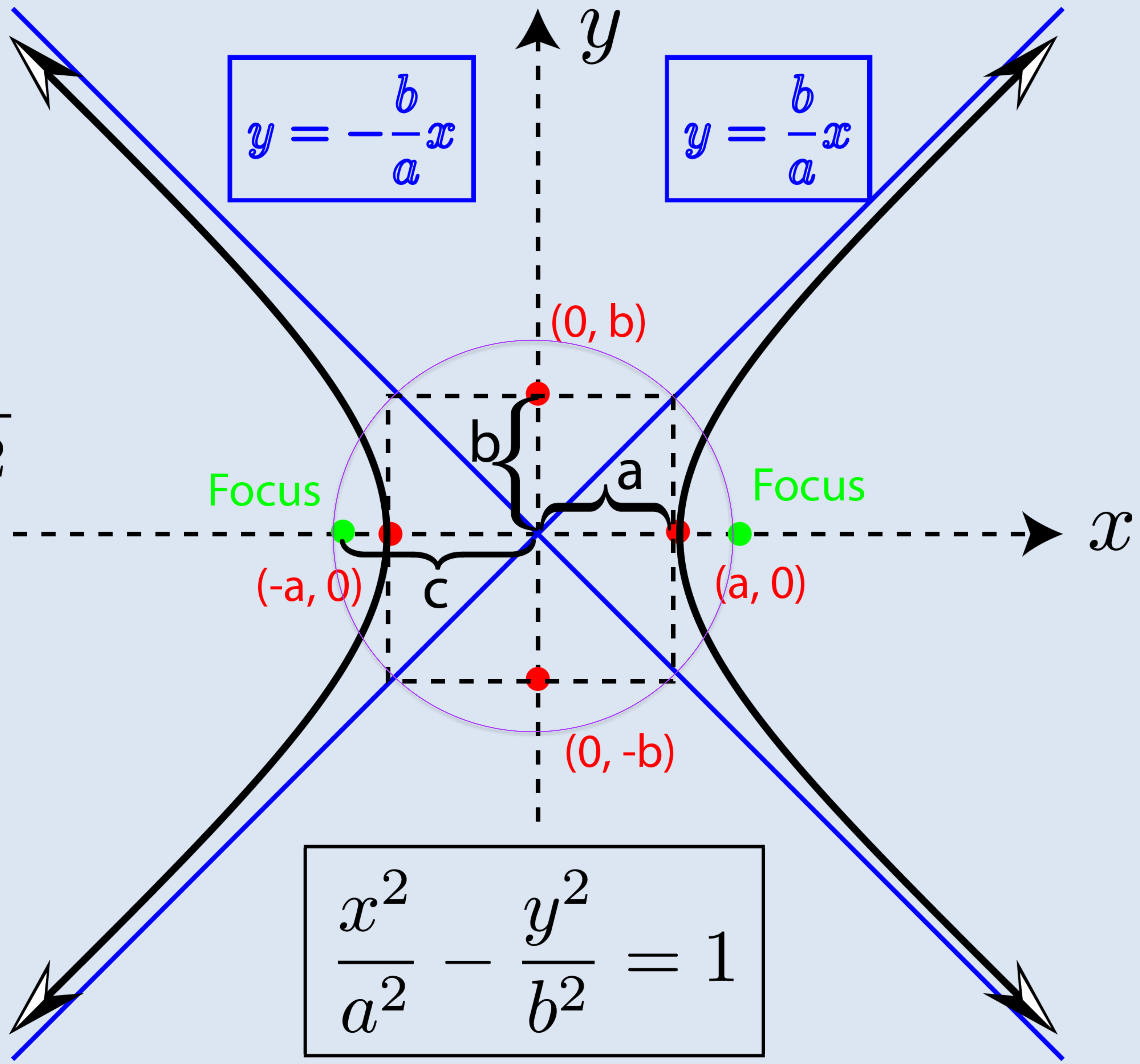
→ hyperbolic excess velocity

Hyperbola basics

EPFL

$$c = \sqrt{a^2 + b^2}$$

$$\boxed{c = ae}$$



$$\boxed{\frac{x^2}{a^2} - \frac{y^2}{b^2} = 1}$$

Vis-Viva equation – Velocities on elliptical and hyperbolic orbits

Ellipse:

$$\boxed{V = \sqrt{\frac{2\mu}{r} - \frac{\mu}{a}}}$$

$$\boxed{\epsilon = -\frac{\mu}{2a}}$$

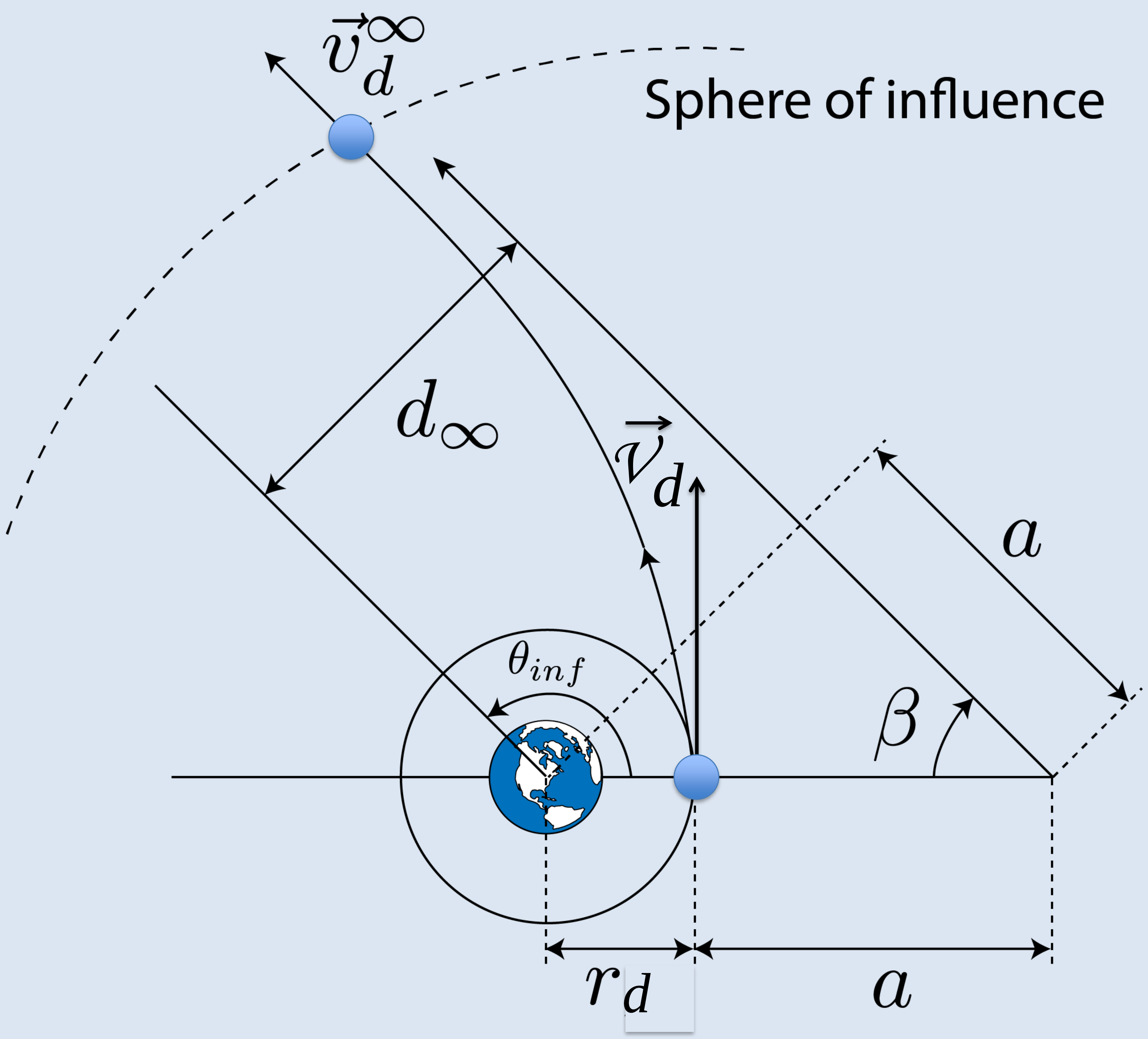
Hyperbola:

$$\boxed{V = \sqrt{\frac{2\mu}{r} + \frac{\mu}{a}}}$$

$$\boxed{\epsilon = +\frac{\mu}{2a}}$$

The energy per unit mass on an elliptical or hyperbolic trajectory is only dependent on the mass of the central object μ, and on the value of the semi-major axis a, and ***not*** on the eccentricity.

Departure from a planet



$$r_d = r_{\text{departure}} = r_{\text{perigee}}$$

$$\boxed{v = \sqrt{\frac{2\mu}{r} + \frac{\mu}{a}}}$$

$$a \cong \frac{\mu}{\left(v_d^{\infty}\right)^2}$$

$$e = \frac{a + r_d}{a} = \frac{c}{a} > 1$$

$$\theta_{\text{inf}} = \arccos\left(-\frac{1}{e}\right)$$

EPFL

4.2.4 Arrival to a planet

Space Mission Design and Operations

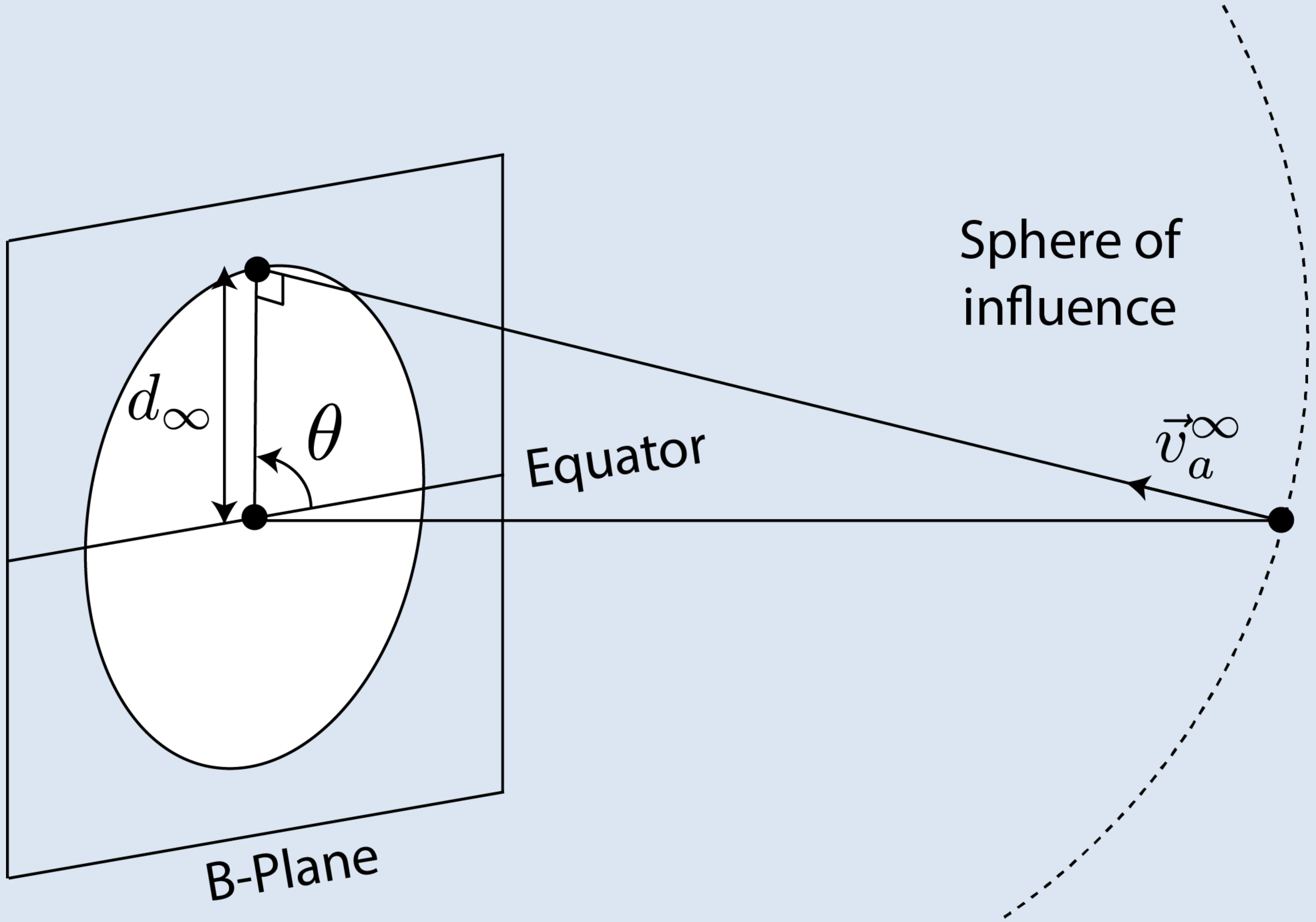
Prof. Claude Nicollier

Definitions

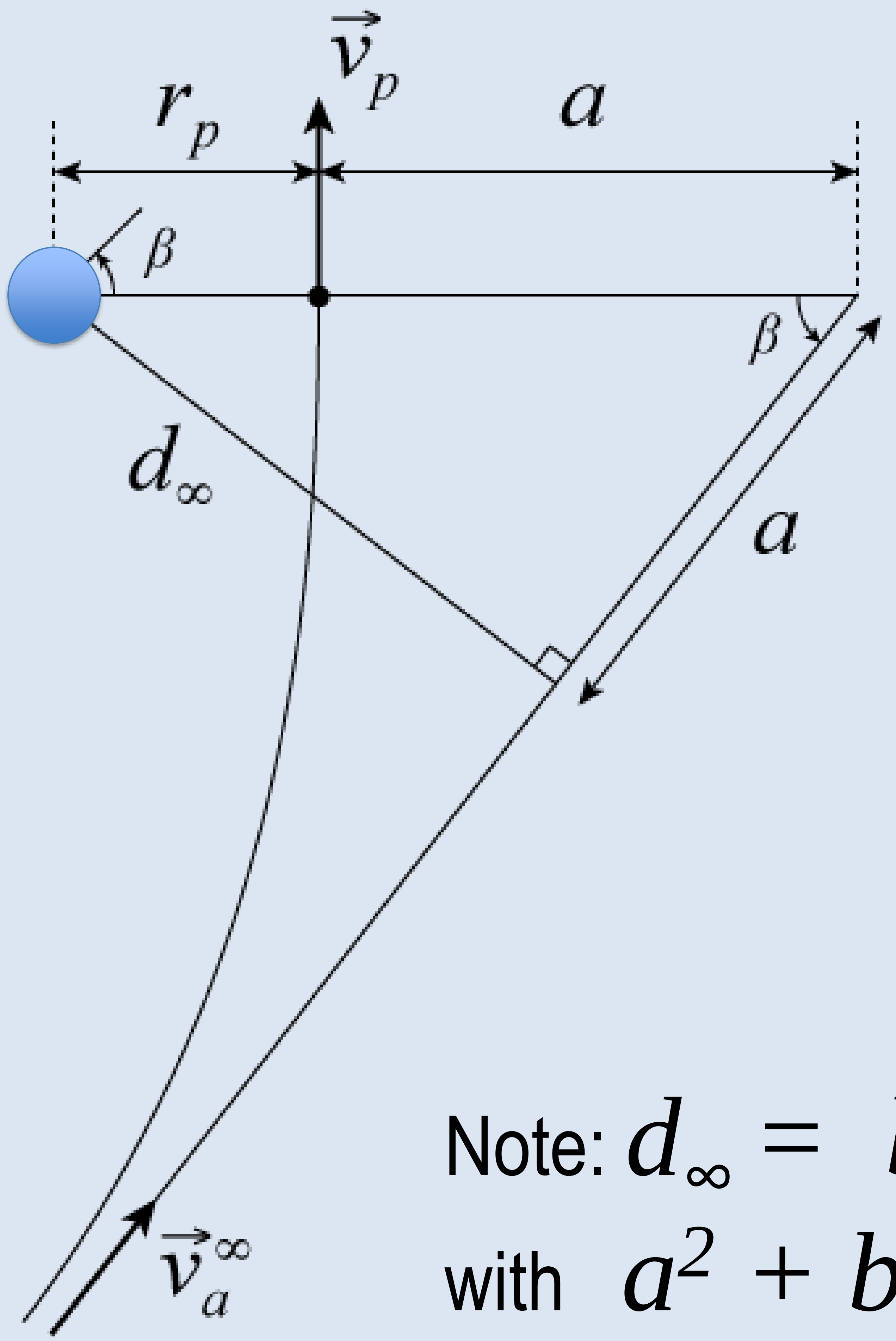
On its journey to the planet of destination, the spacecraft is on an elliptical heliocentric trajectory, until it gets close to that planet. Then only the motion of the spacecraft with respect to the destination planet is considered as hyperbolic trajectory inside the sphere of influence of this planet.

Flight controllers steer the motion of the spacecraft and select the value of d_∞, the impact parameter, and θ in order to accomplish the mission objective, either a flyby (and possible orbit insertion) or a direct landing on the surface of the planet.

Arrival at velocity $\vec{v}_a^{\infty} = \vec{V}_S - \vec{V}_P$ on the sphere of influence of the destination planet.



Determination of important parameters



Conservation of total energy $\rightarrow$ $\boxed{v_p^2 = (v_a^\infty)^2 + v_{Erp}^2}$

also $a \cong \dfrac{\mu}{(v_a^\infty)^2}$

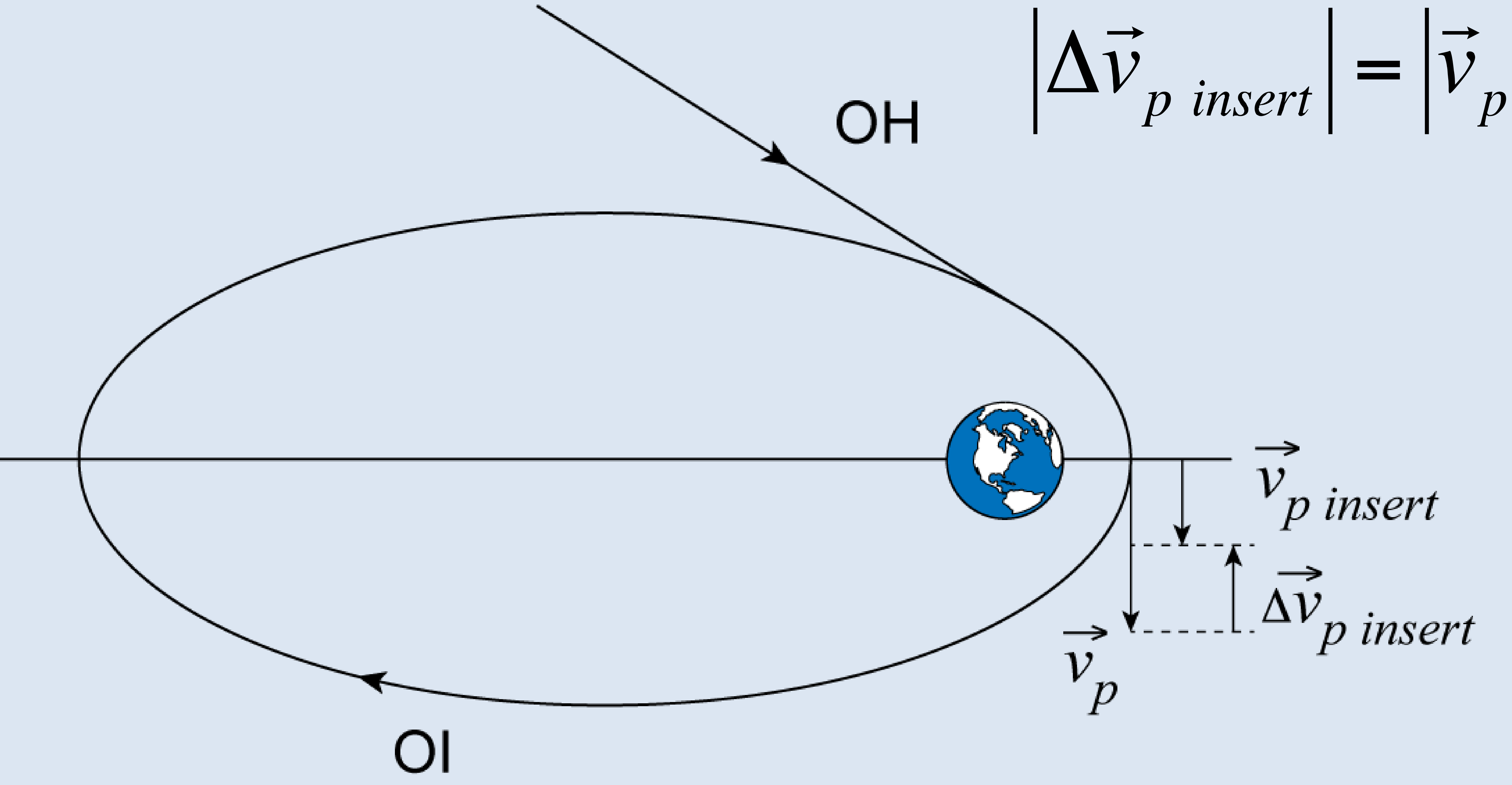
$$r_p = -\frac{\mu}{(v_a^\infty)^2} + \sqrt{\frac{\mu^2}{(v_a^\infty)^4} + d_\infty^2}$$

$$\cos\beta = \frac{a}{a + r_p} = \frac{a}{c}$$

Note: $d_\infty = b$

with $a^2 + b^2 = c^2 = (a + r_p)^2$

Orbit insertion around the destination planet

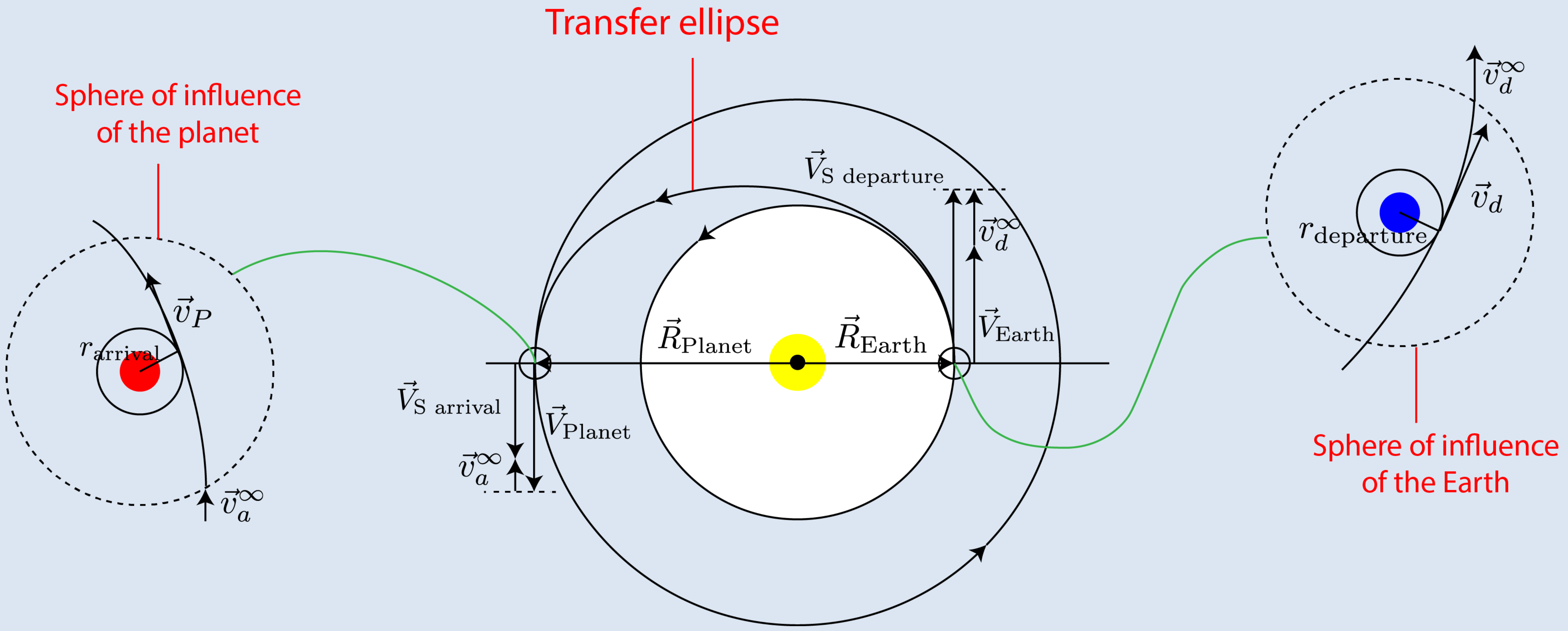


$$\left|\Delta\vec{v}_{p\ insert}\right| = \left|\vec{v}_p\right| - \left|\vec{v}_{p\ insert}\right| = \sqrt{\frac{2\mu}{r_p} + \left(v_a^\infty\right)^2} - \sqrt{\frac{2\mu}{r_p} - \frac{\mu}{a_i}}$$

p for "periapsis"

a_i is the semi major axis of the elliptical orbit at insertion

$\Delta v_{p\ insert}$ is how much the spacecraft needs to brake in km/sec in order to be inserted in an elliptical orbit around the destination planet.



4.2.5 Summary

Space Mission Design and Operations

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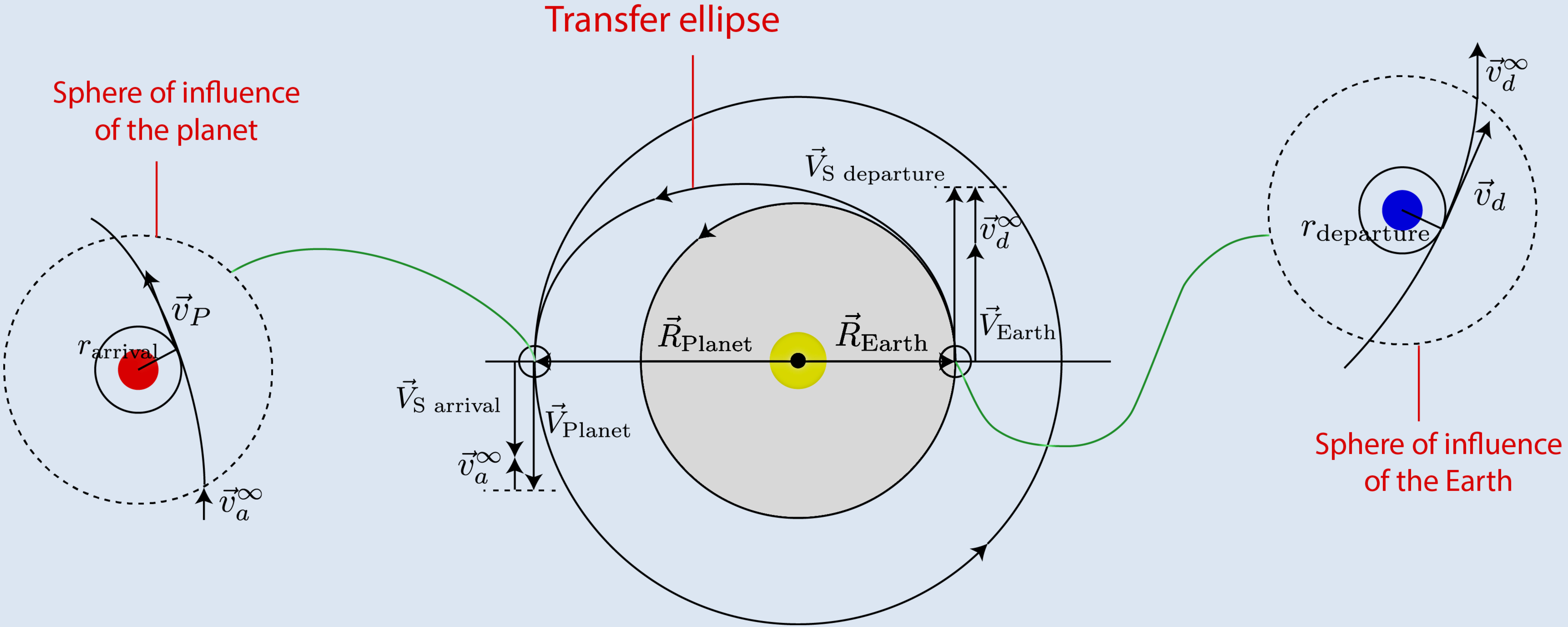
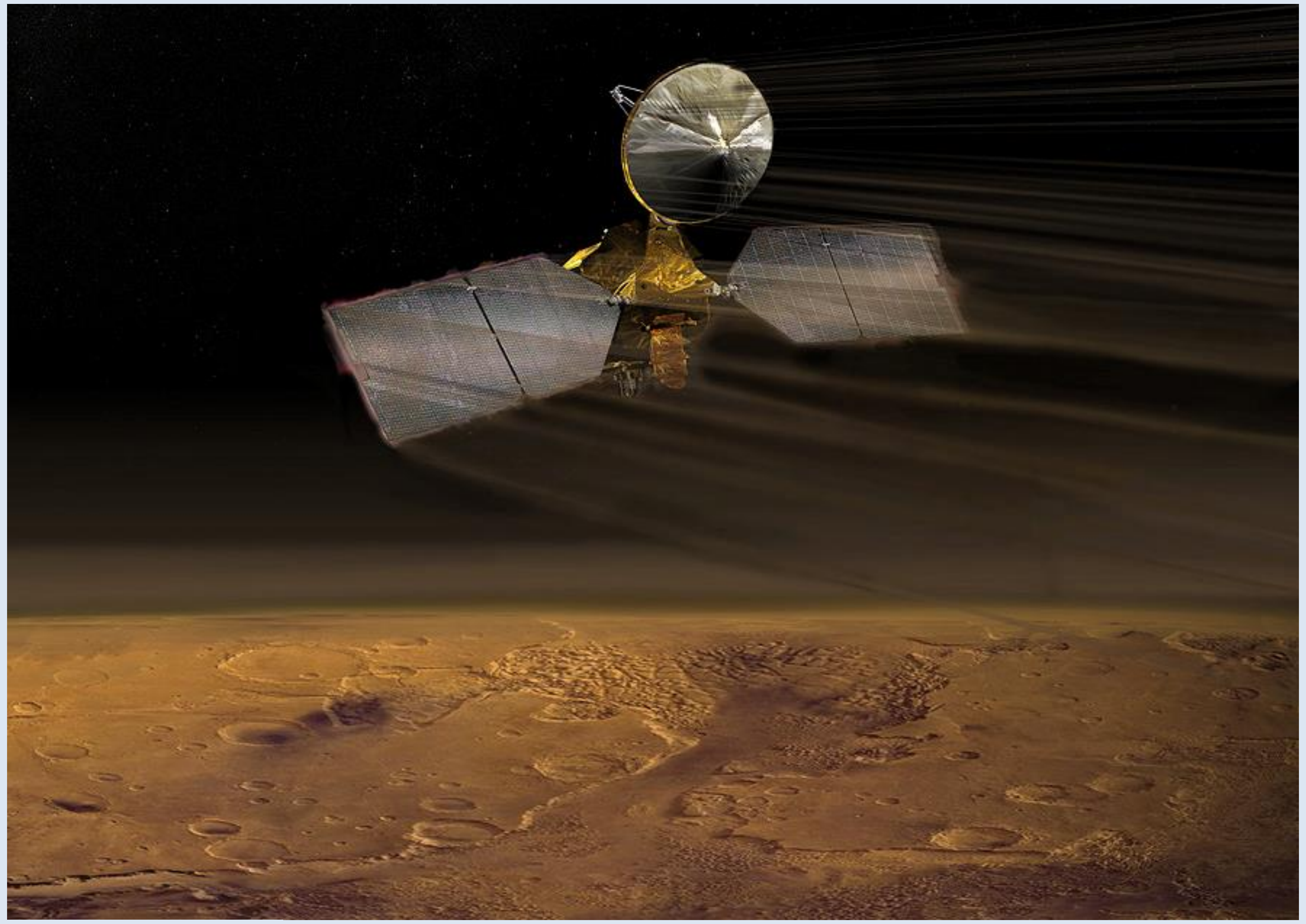
Strategy for interplanetary transfer



Some data for interplanetary trajectories

	Venus	Mars	Jupiter	Saturn
Semi-major axis of the planet (AU)	0.723	1.523	5.204	9.554
Sidereal period of the planet (years)	6.615	1.880	11.865	29.531
Periodicity of the Hohmann transfer (years)	1.596	2.137	1.092	1.035
Transfer orbit periapsis radius (AU)	0.723	1	1	1
Transfer orbit apoapsis radius (AU)	1	1.523	5.202	9.554
Duration of the Hohmann transfer (years)	0.399	0.708	2.730	6.061
Heliocentric Earth departure velocity (km/s)	27.29	32.73	38.58	40.08
Earth departure excess velocity (km/s)	−2.50	2.94	8.79	10.29
Arrival heliocentric velocity (km/s)	37.74	21.49	7.42	4.19
Target planet heliocentric velocity (km/s)	35.03	24.13	13.06	9.64
Arrival excess velocity (km/s)	2.71	−2.65	−5.64	−5.45



4.3.1 Aerodynamic braking maneuvers

Space Mission Design and Operations

Prof. Claude Nicollier

Credits: NASA, JPL

Aerodynamic braking maneuvers

The three techniques use braking of a spacecraft through the atmosphere of a planet (Earth, Mars, Venus, or other) for capture by the planet, or to cause a change of trajectory, or else a full entry in the atmosphere of the planet.

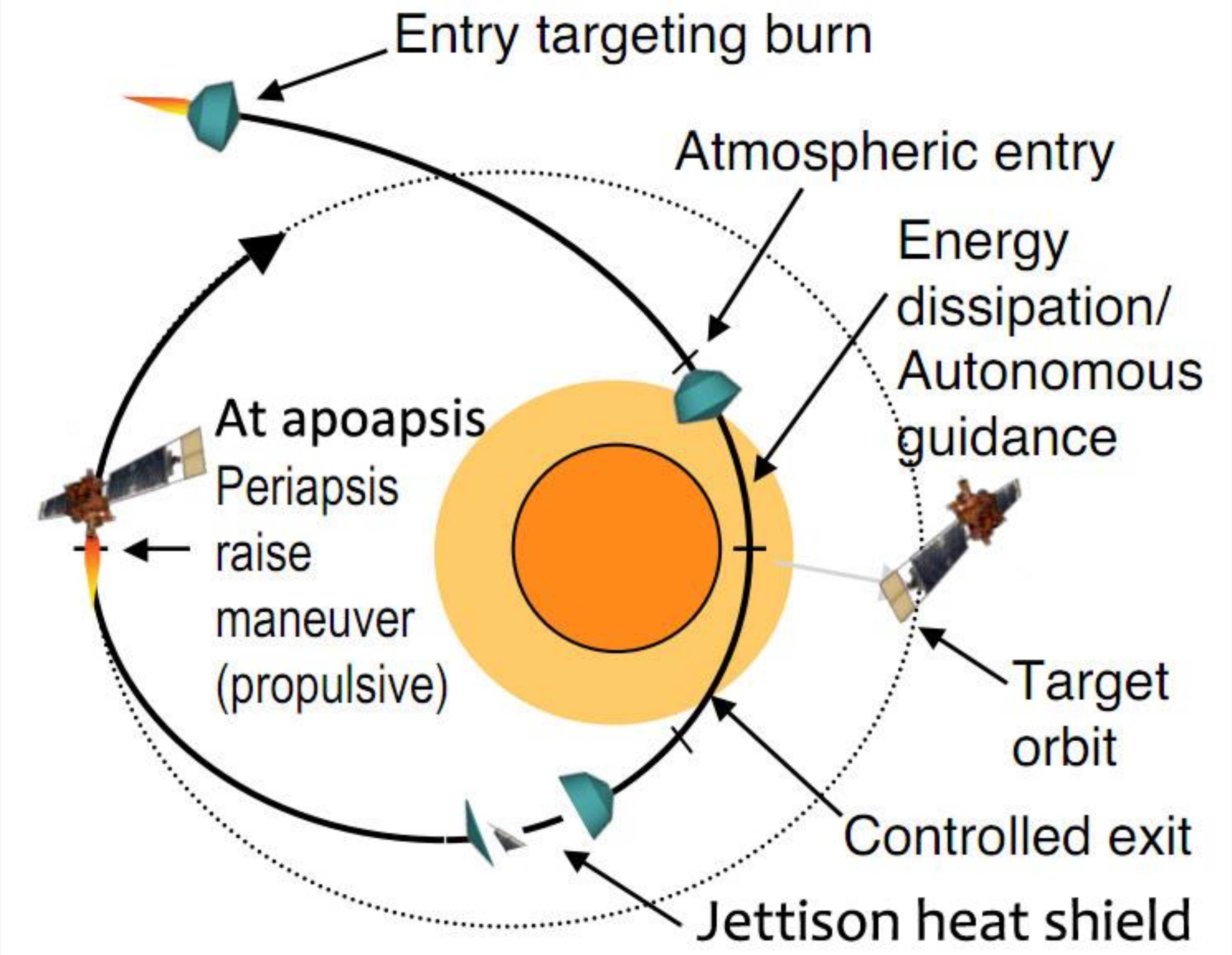
In this last case, the entry is followed by the deployment of a parachute for a capsule (Apollo CM, Soyuz, Orion), or transition to atmospheric flight for a winged spacecraft. (Shuttle).

A thermal shield on the spacecraft is needed to avoid overheating during the braking maneuver.

- **Aerocapture**
  - Transfers the spacecraft from a hyperbolic approach trajectory to an elliptical orbit around the target planet.
  - Further loss of energy will occur at every subsequent crossing of the periapsis (through aerobraking).
- **Aerobraking**
  - Transfers the spacecraft from an initial elliptical orbit to a less energetic (i.e. lower apoapsis) elliptical orbit.
  - Involves relatively small ΔV.
- **Aeroentry**
  - Transfers the spacecraft from either a hyperbolic, parabolic or elliptical approach orbit to the planet surface.

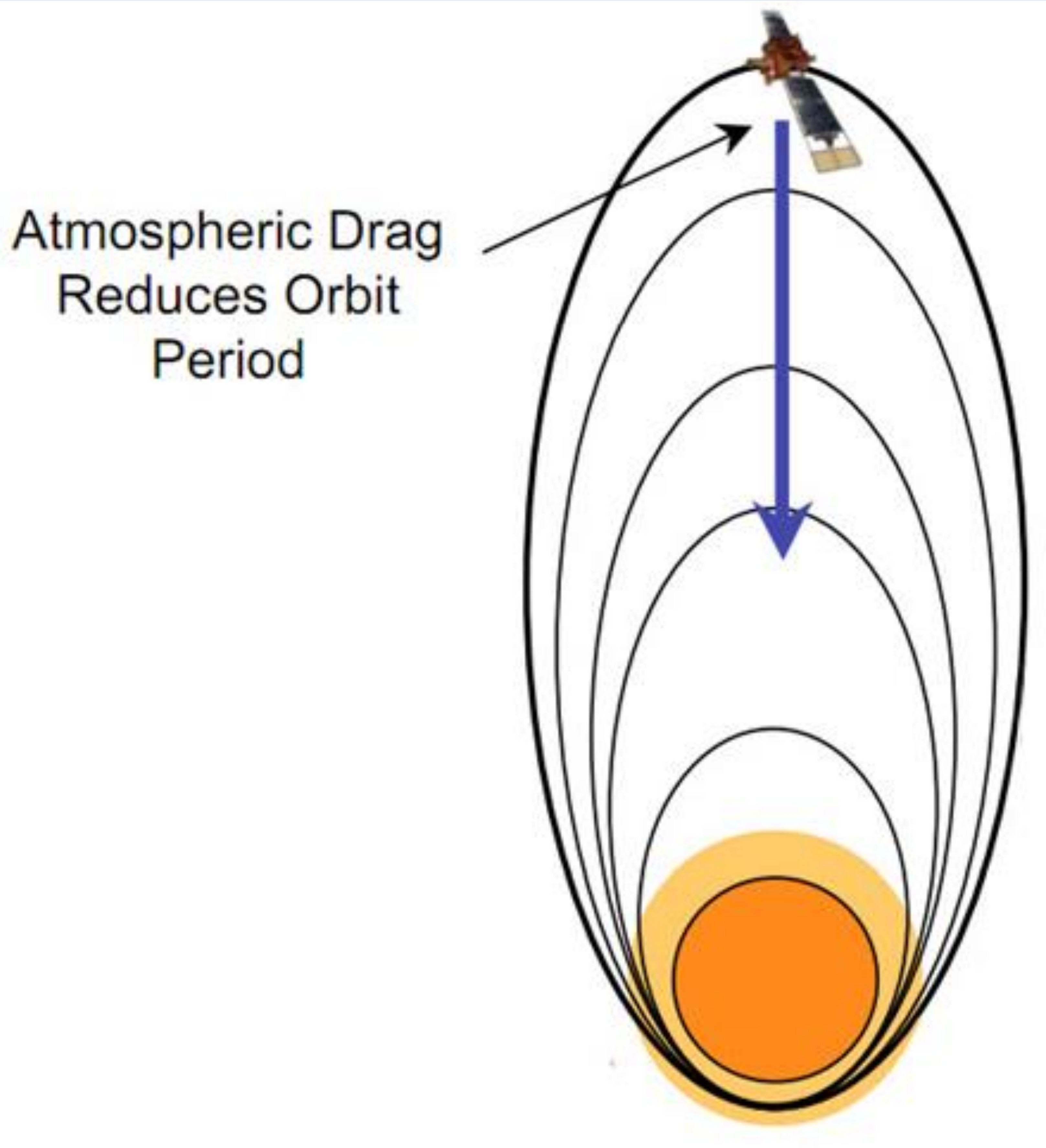
Aerocapture

Aerocapture has never been used in practice: if the spacecraft is too close to the planet and the high-density layers of the atmosphere, with a higher *ΔV* than expected, the entry into the planet may be uncontrolled. If the spacecraft is too high, it may have a too small *ΔV*, and not be able to reach a velocity under the escape velocity. It may still escape the planet.



Credits: NASA, In-Space Propulsion Technology (ISPT), Michelle M. Munk & Tibor Kremic, March 24, 2008

Aerobraking example



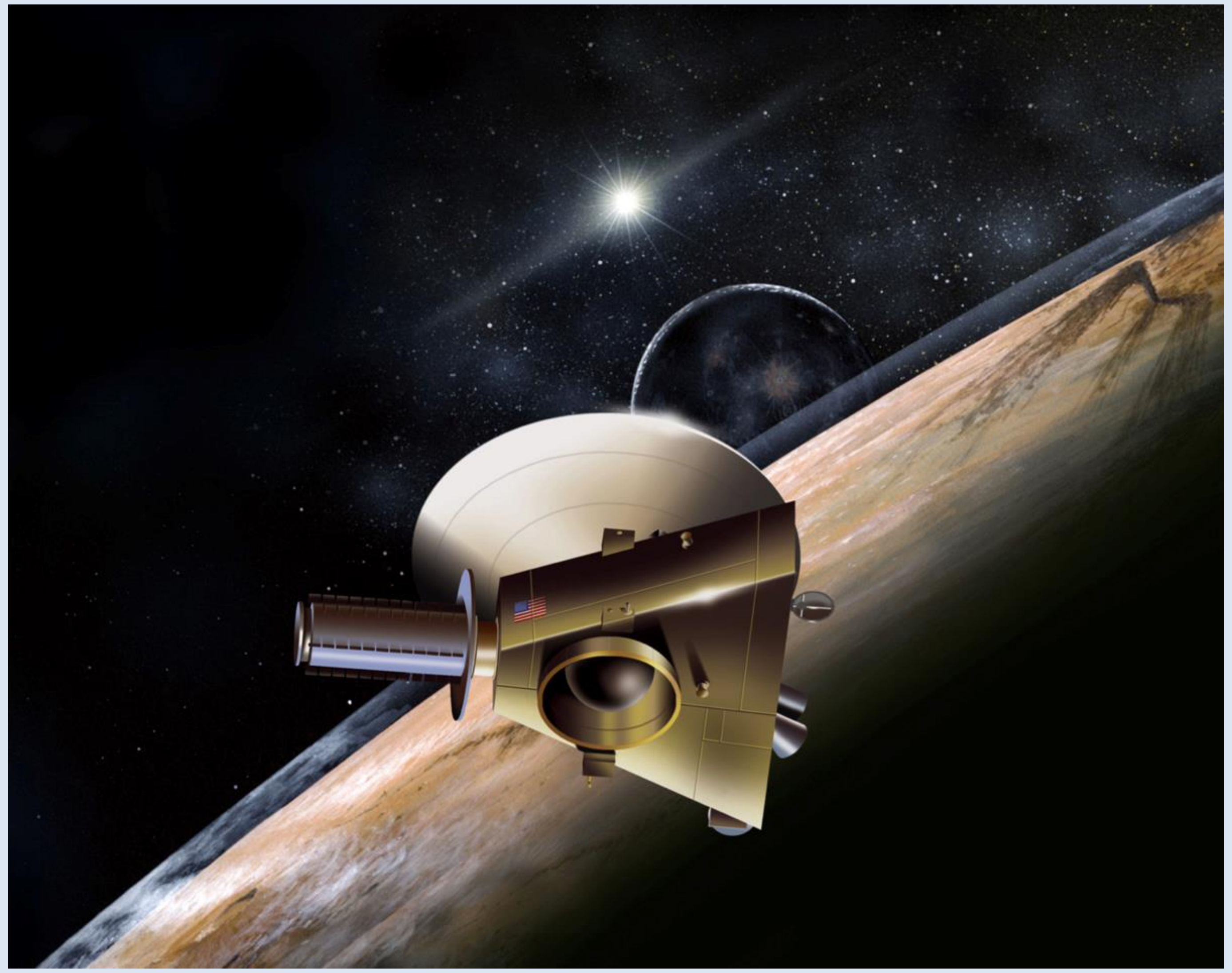
Credits: NASA, In-Space Propulsion Technology (ISPT), Michelle M. Munk & Tibor Kremic, March 24, 2008

Aeroentry

Example of the Apollo Command Module coming back from a 3 days journey from the Moon.

The velocity of entry of the Command Module with the three crew members on board in the high atmosphere was very high, about 11 km/sec.

Credits: NASA



4.3.2 Slingshot maneuver (gravity assist)

Space Mission Design and Operations

Prof. Claude Nicollier

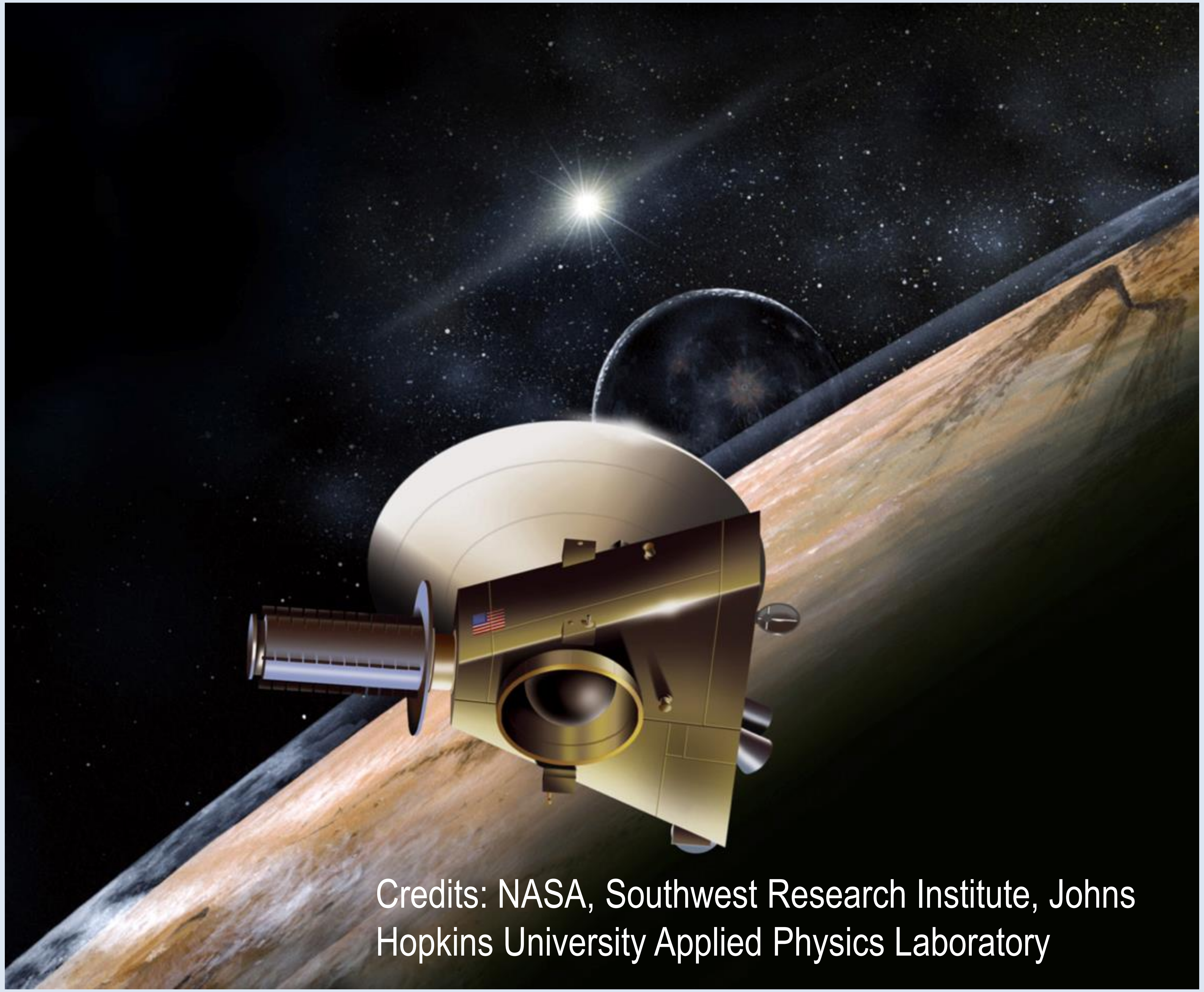
Credits: NASA, Southwest Research Institute, Johns Hopkins University Applied Physics Laboratory

New Horizons example

- Launched on January 19, 2006.
- Gravity assist by Jupiter on February 28, 2007.
- Pluto flyby on July 14, 2015.

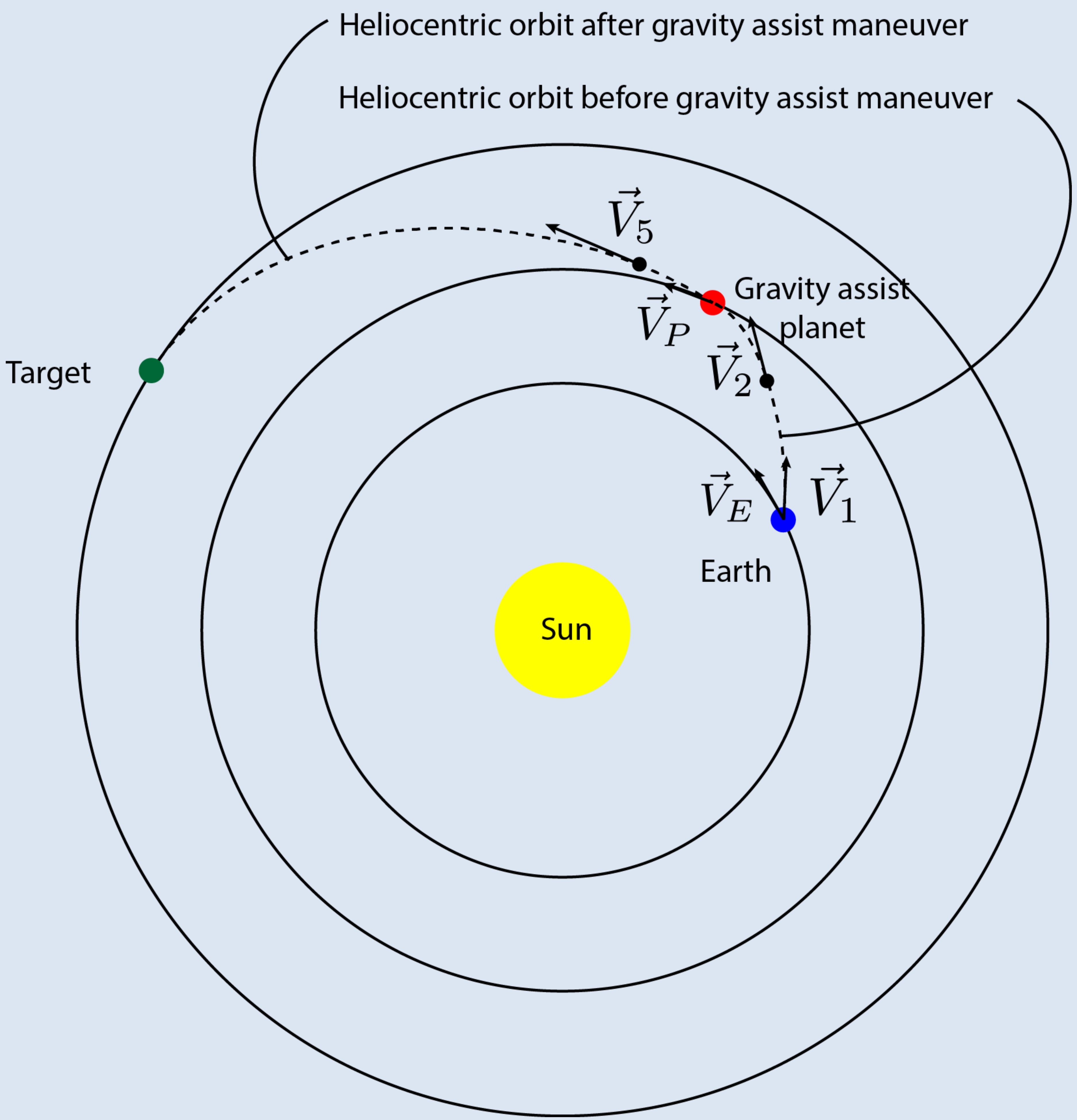
When a spacecraft on an heliocentric trajectory approaches another planet and comes in close proximity to it, the gravity of that planet takes over, pulling the spacecraft and altering its heliocentric velocity in amplitude and direction.

The amount by which the spacecraft speeds up or slows down is determined by the geometry of the approach, passing behind or in front of the planet.



Credits: NASA, Southwest Research Institute, Johns Hopkins University Applied Physics Laboratory

Definitions



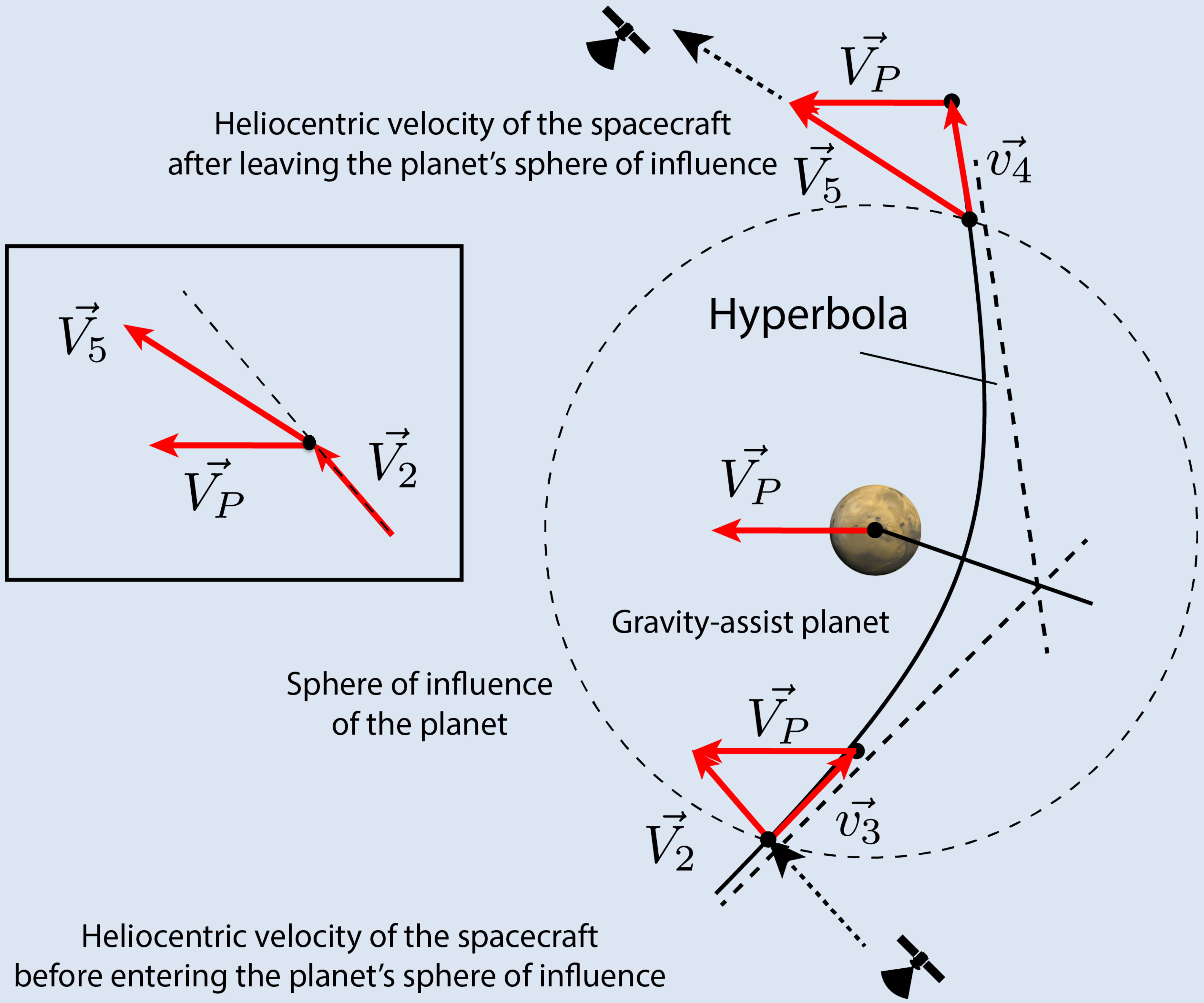
- $\vec{V}_P$: Heliocentric velocity of the gravity-assist-planet.
- $\vec{V}_E$: Heliocentric velocity of the Earth.
- $\vec{V}_1$: Heliocentric velocity of the spacecraft after leaving Earth.
- $\vec{V}_2$: Heliocentric velocity of the spacecraft entering the gravity-assist planet's sphere of influence.
- $\vec{V}_5$: Heliocentric velocity of the spacecraft leaving the gravity-assist planet's sphere of influence.

Slingshot maneuver profile

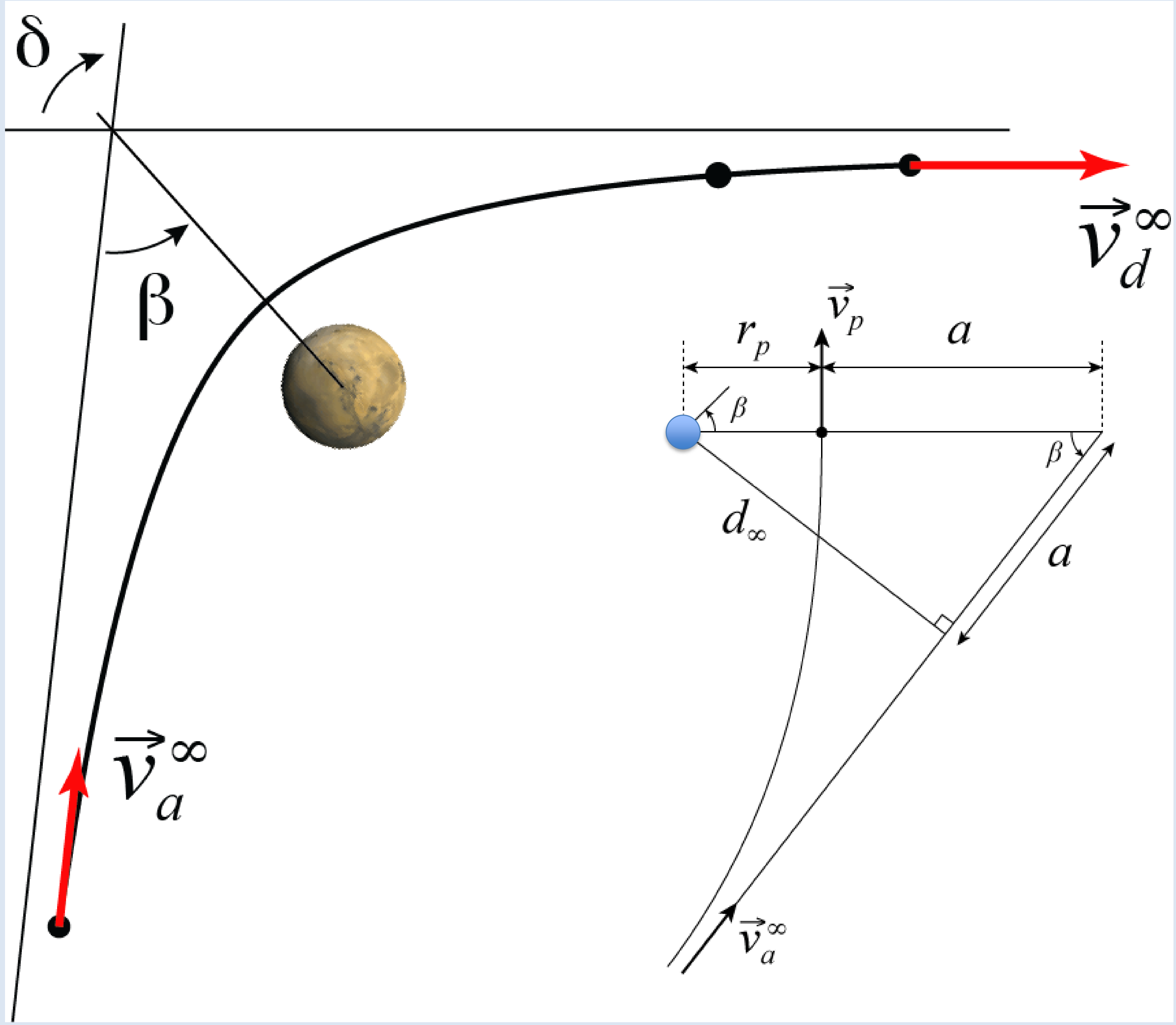
- $\vec{V}_2$: Heliocentric velocity of the spacecraft entering the planet's sphere of influence.
- $\vec{V}_5$: Heliocentric velocity of the spacecraft leaving the planet's sphere of influence.
- $\vec{V}_P$: Heliocentric velocity of the planet.
- $\vec{v}_3$: Planetocentric velocity of the spacecraft entering the planet's sphere of influence.
- $\vec{v}_4$: Planetocentric velocity of the spacecraft leaving the planet's sphere of influence.

$$|\vec{v}_3| = |\vec{v}_4|$$

$$|\vec{V}_P + \vec{v}_4| > |\vec{V}_P + \vec{v}_3|$$



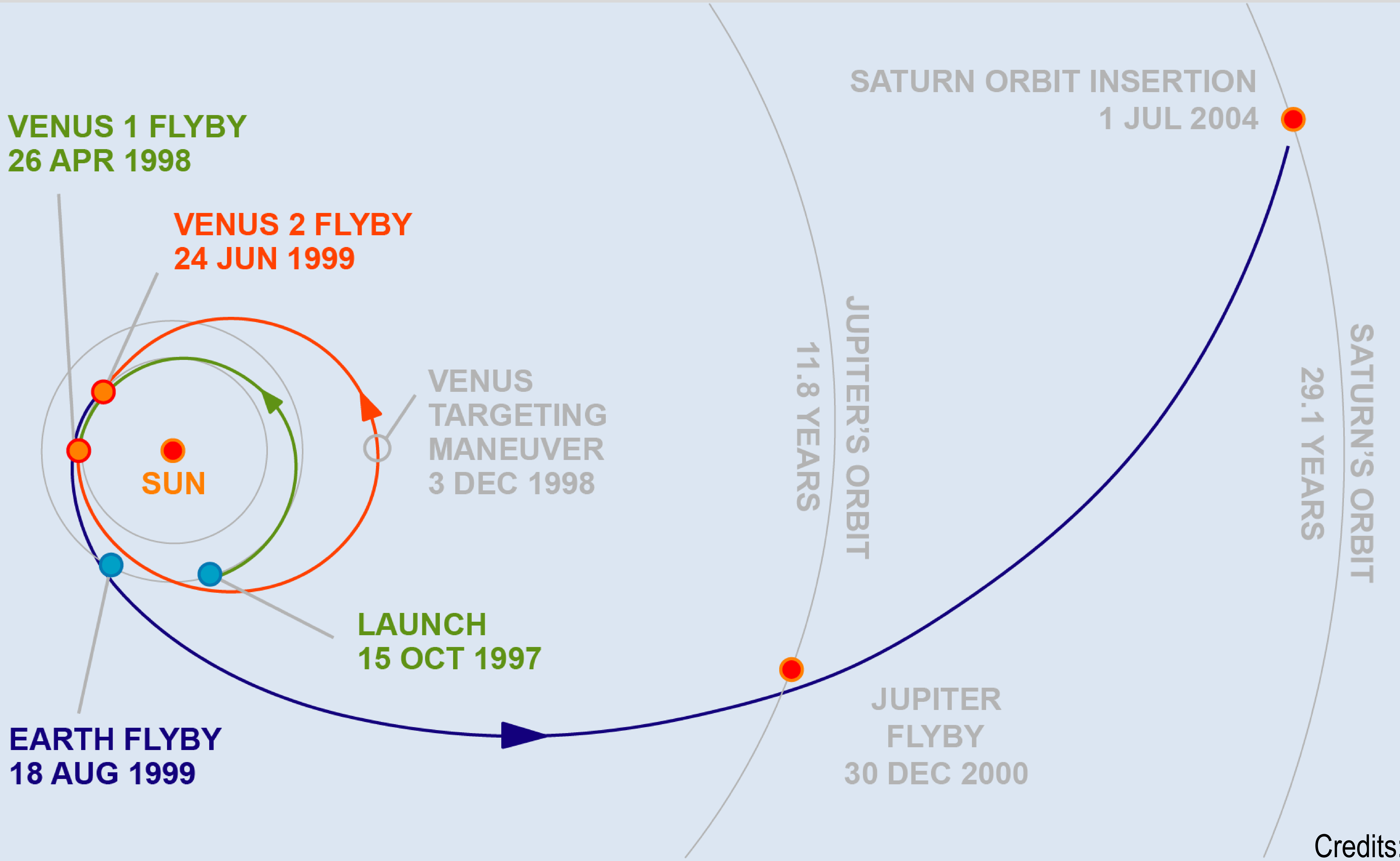
Slingshot maneuver parameters



- In the vicinity of a planet, the trajectory of a spacecraft is hyperbolic with a periapsis at distance r_p to the center of the planet.
- δ is the angle between the directions of $\vec{v}_a^\infty$ and $\vec{v}_d^\infty$.

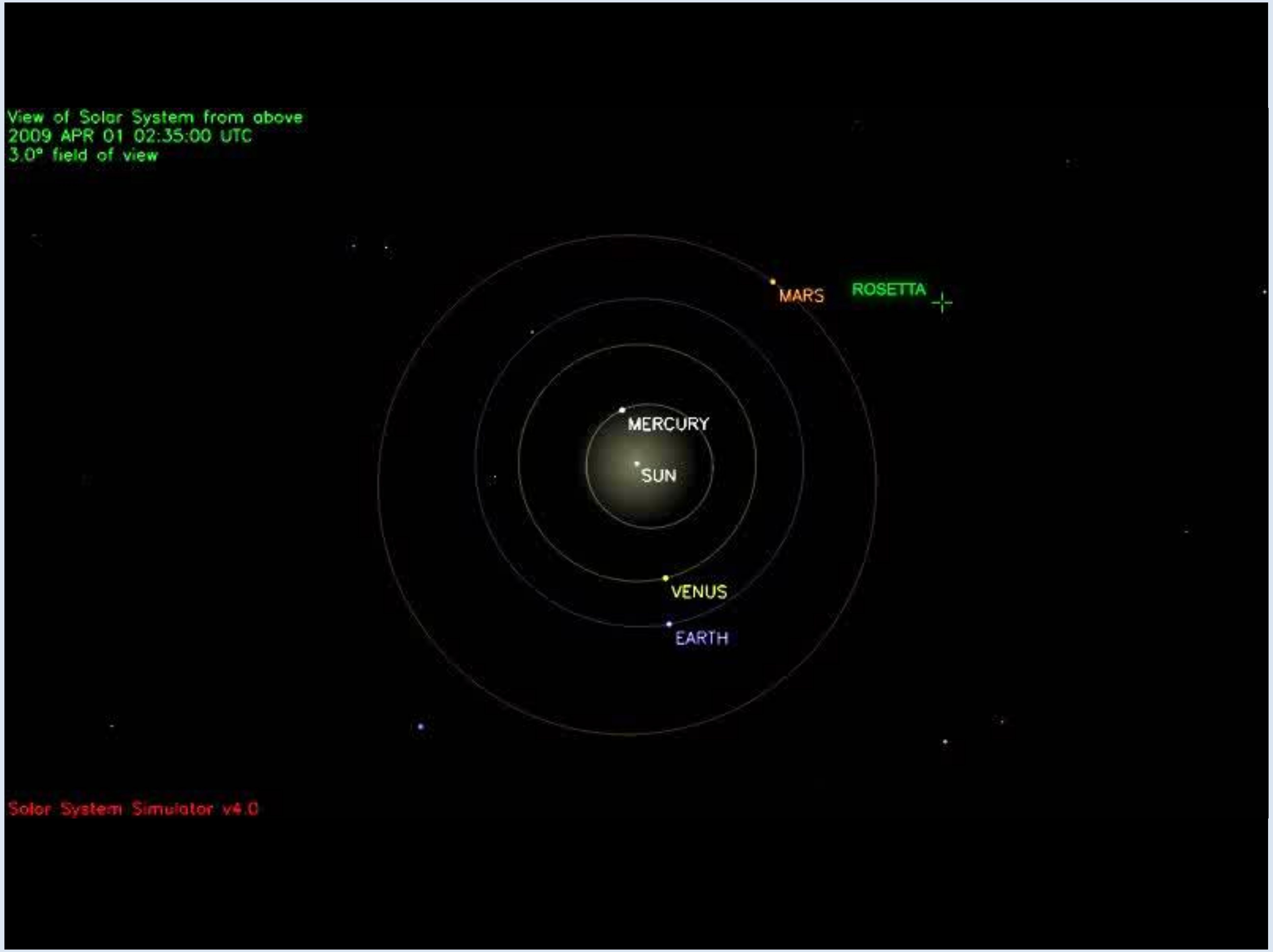
$$\cos\left(\frac{\delta}{2}\right) = \cos\beta = \frac{a}{a + r_p}$$

Cassini-Huygens example (4 flybys)



Credits: PD, USGOV

Rosetta's Earth flyby



4.4.1 Spacecraft propulsion

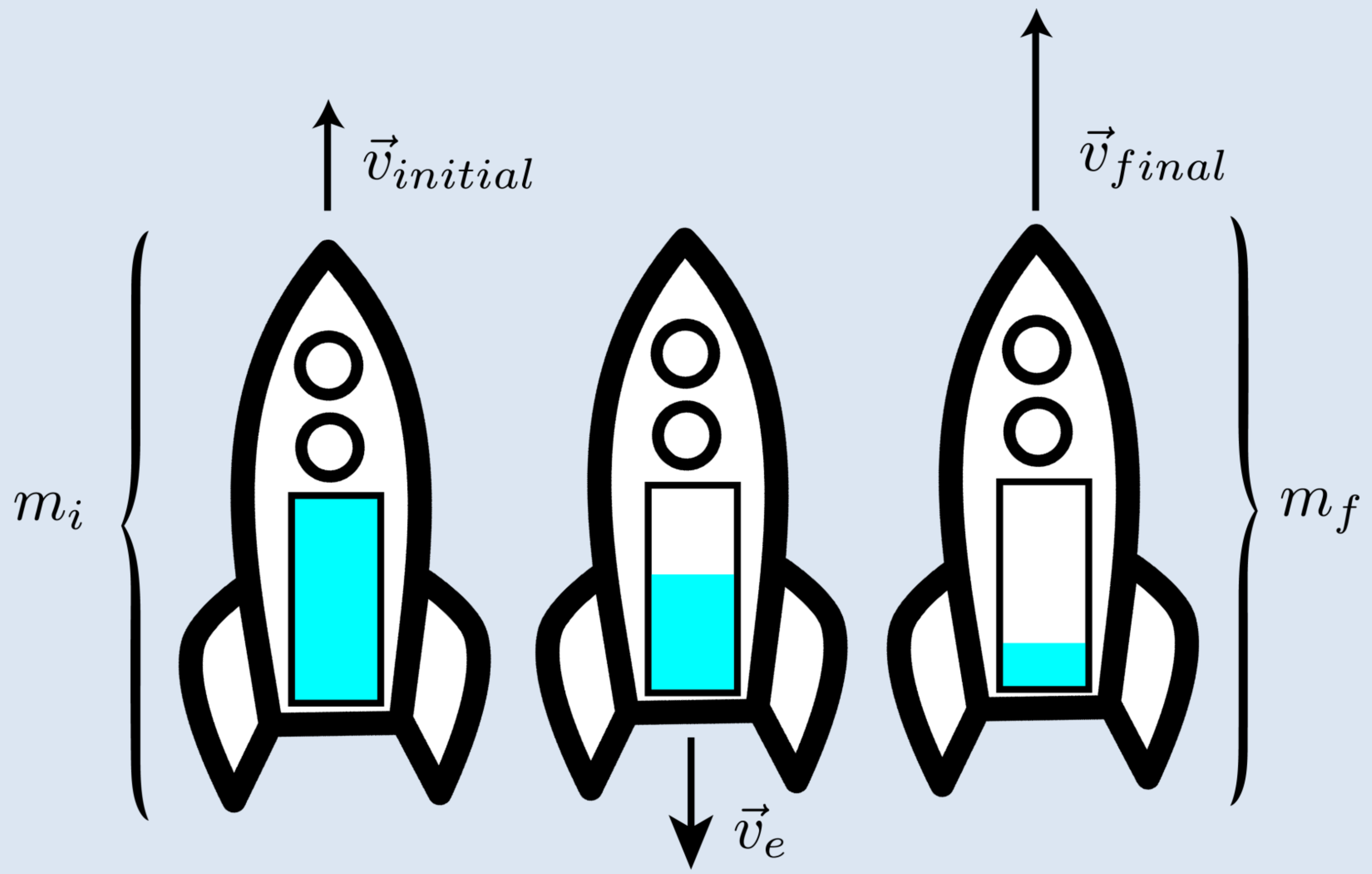
Space Mission Design and Operations

Prof. Claude Nicollier

Ariane 5, Kourou

Credits: ESA, CNES,
Arianespace-Service Optique CSG

Tsiolkovsky equation or rocket equation



$$\boxed{\Delta V = v_e \log_e \left(\frac{m_i}{m_f} \right)}$$

- ΔV = change of velocity induced by the propulsion system
- v_e = exhaust velocity of the gas in the propulsion system
- m_i, m_f = initial and final mass
- Valid in free space – gravitational field-induced and drag-induced ΔVs will be added to the propulsion-induced ΔVs.

Thrust and acceleration

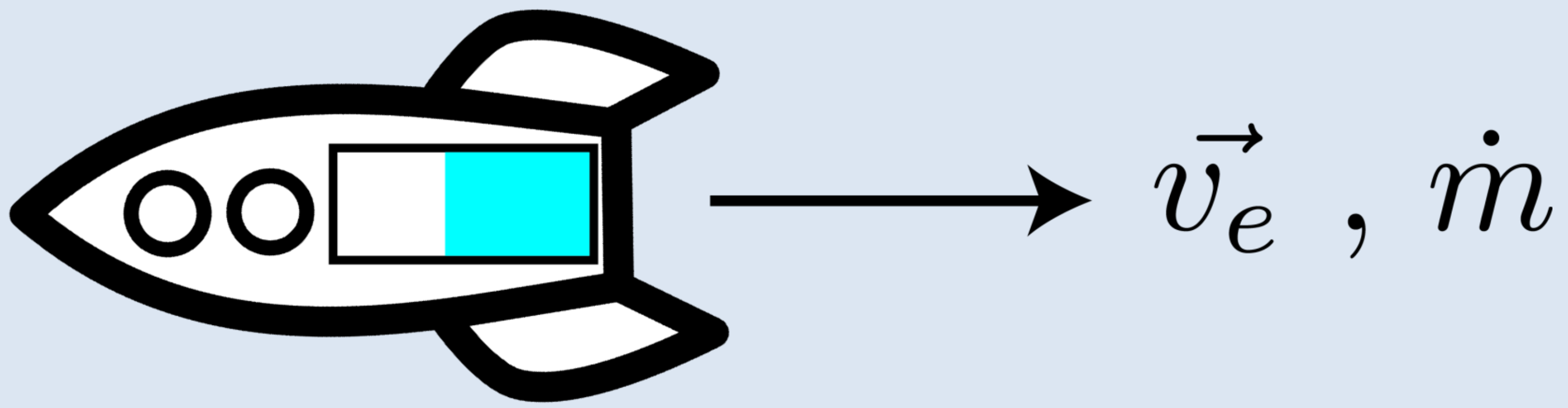
- Thrust of the propulsion system (static case):

$$F = v_e \dot{m} \text{ with } \dot{m} = \text{mass flow and } v_e = \text{ejection velocity}$$

- Resulting acceleration of the spacecraft:

$$\frac{dV}{dt} = \frac{F}{m} \rightarrow \frac{dV}{dt} = v_e \frac{\dot{m}}{m}$$

- Integrating between the initial and final conditions:



$$\boxed{\Delta V = v_e \log_e \left(\frac{m_i}{m_f} \right)}$$

Propulsion system efficiency

- Other form of the Tsiolkovsky equation:

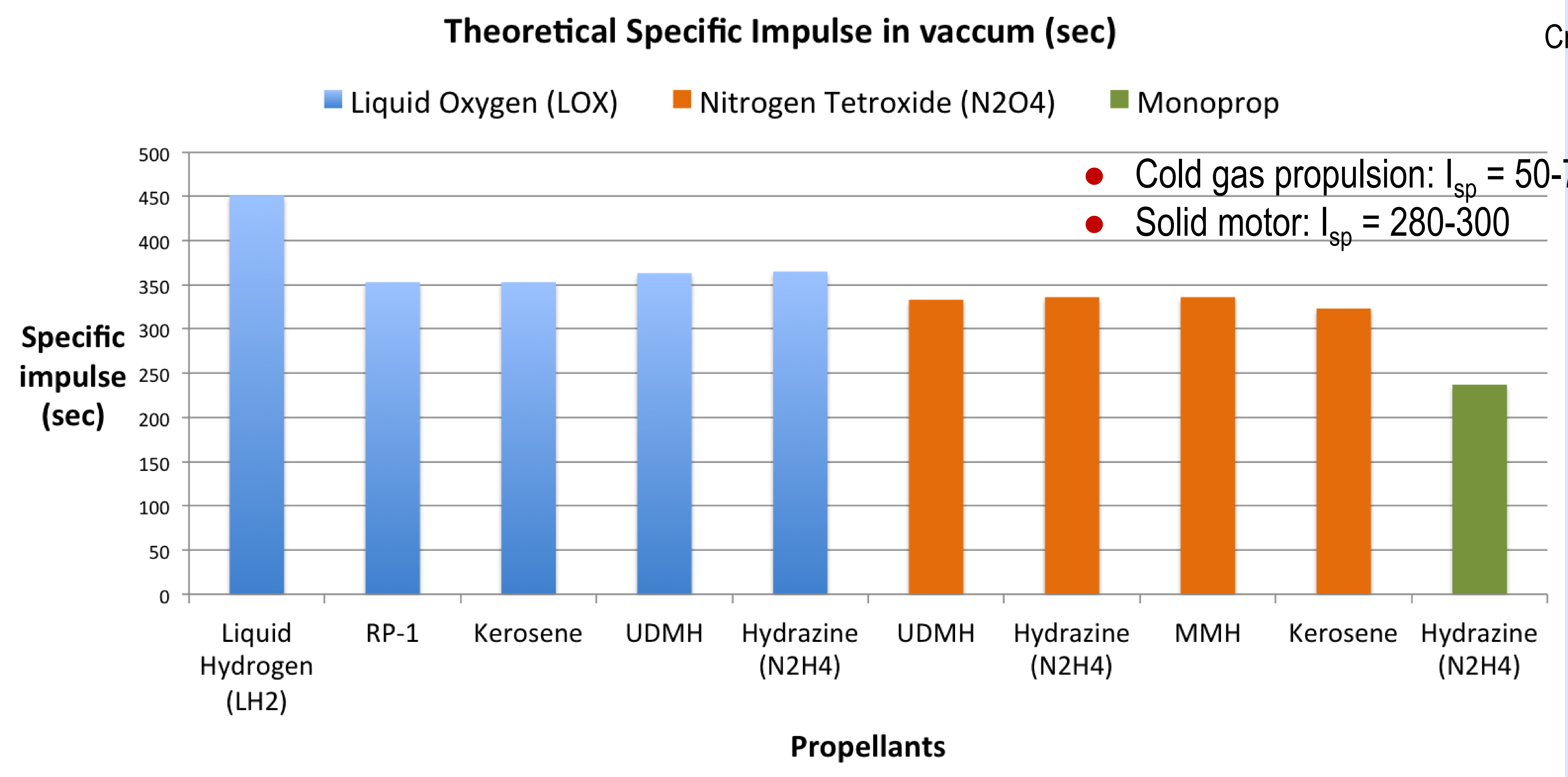
$$\boxed{\Delta V = g \cdot I_{sp} \log_e \left(\frac{m_i}{m_f} \right)}$$

- I_{sp} = specific impulse (s).
- g = Earth's gravity acceleration, 9.81 (m/s^2).
- I_{sp} is a measure of the propulsion system efficiency, it is its thrust (kg-force) divided by the mass flow of propellants (fuel and oxidizer, kg/s), in seconds $I_{sp} = F/\dot{m}g$

$$F = v_e \dot{m} \text{ with } \dot{m} = \text{mass flow and } v_e = \text{ejection velocity}$$

There is a factor of about 10 between the value of I_{sp} in seconds and the exhaust velocity in m/s.

Kind of chemical propulsion



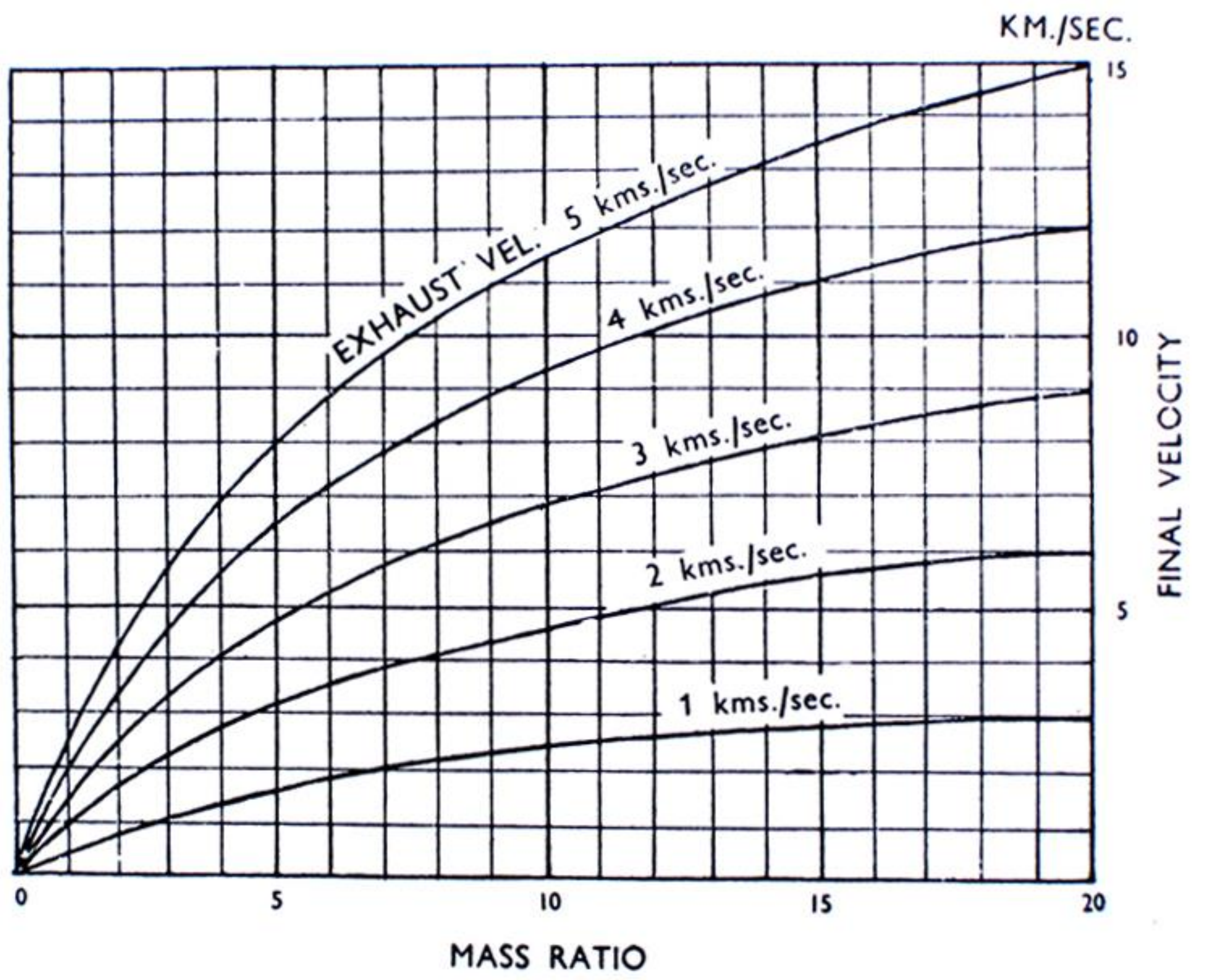
Credits: NASA

- Cold gas propulsion: I_{sp} = 50-75
- Solid motor: I_{sp} = 280-300

Chemical propulsion is normally used for all of the stages until orbit insertion. The highest value Isp is for liquid hydrogen as propellant and liquid oxygen as oxidizer. It will give 450s for the specific impulse, which means an exhaust velocity of 4.5 km/s.

Final velocity VS Mass ratio

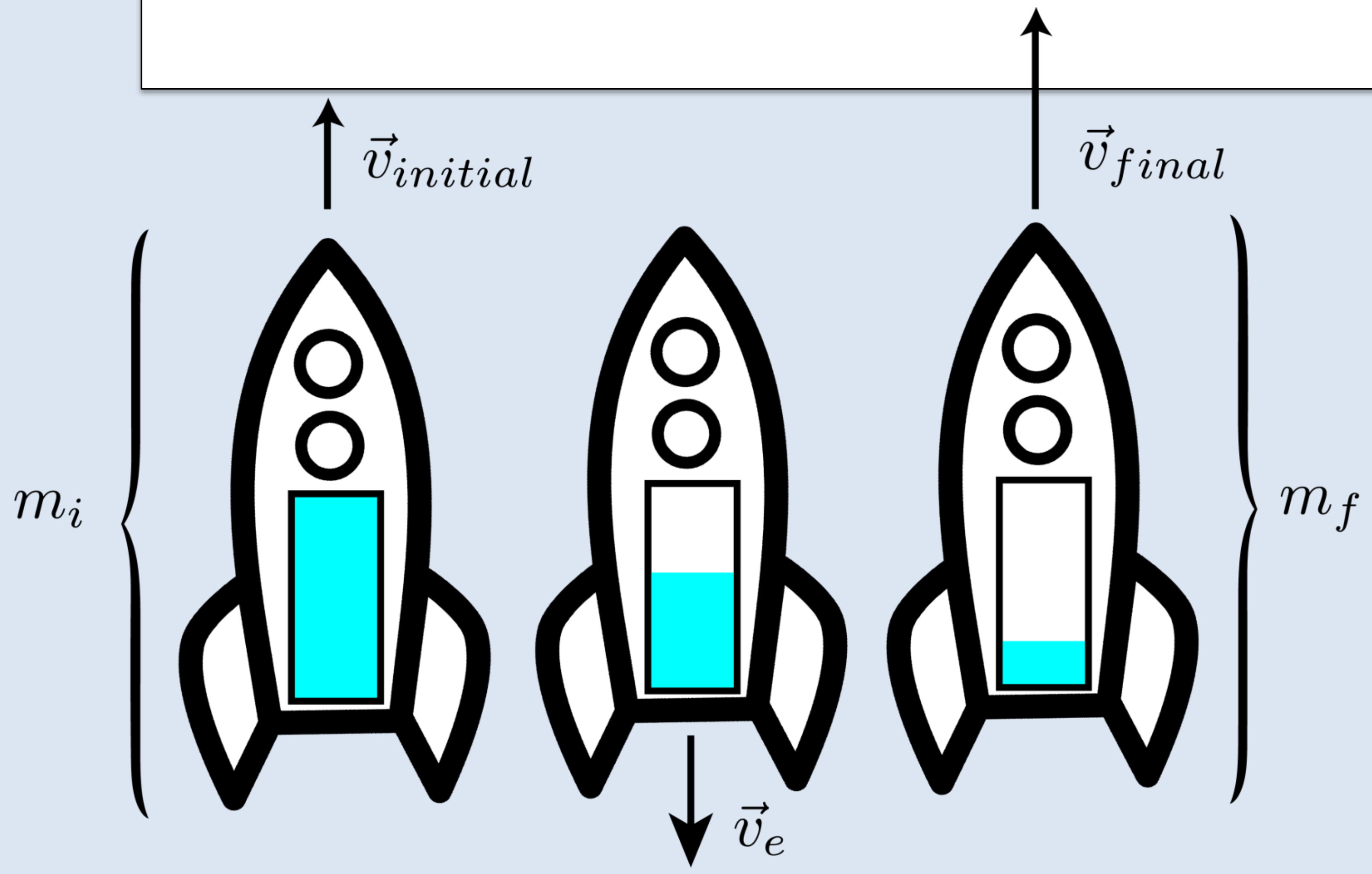
For a capability of about 10 km/s, which is what a rocket needs to reach LEO and make up for gravitational losses and drag losses, and for a LH2/LOX mix (4.5 km/s exhaust velocity), the mass ratio is about 10, about 90% of the total mass at lift off is propellant mass.



Credits: « Ascent to Orbit », Arthur C. Clarke

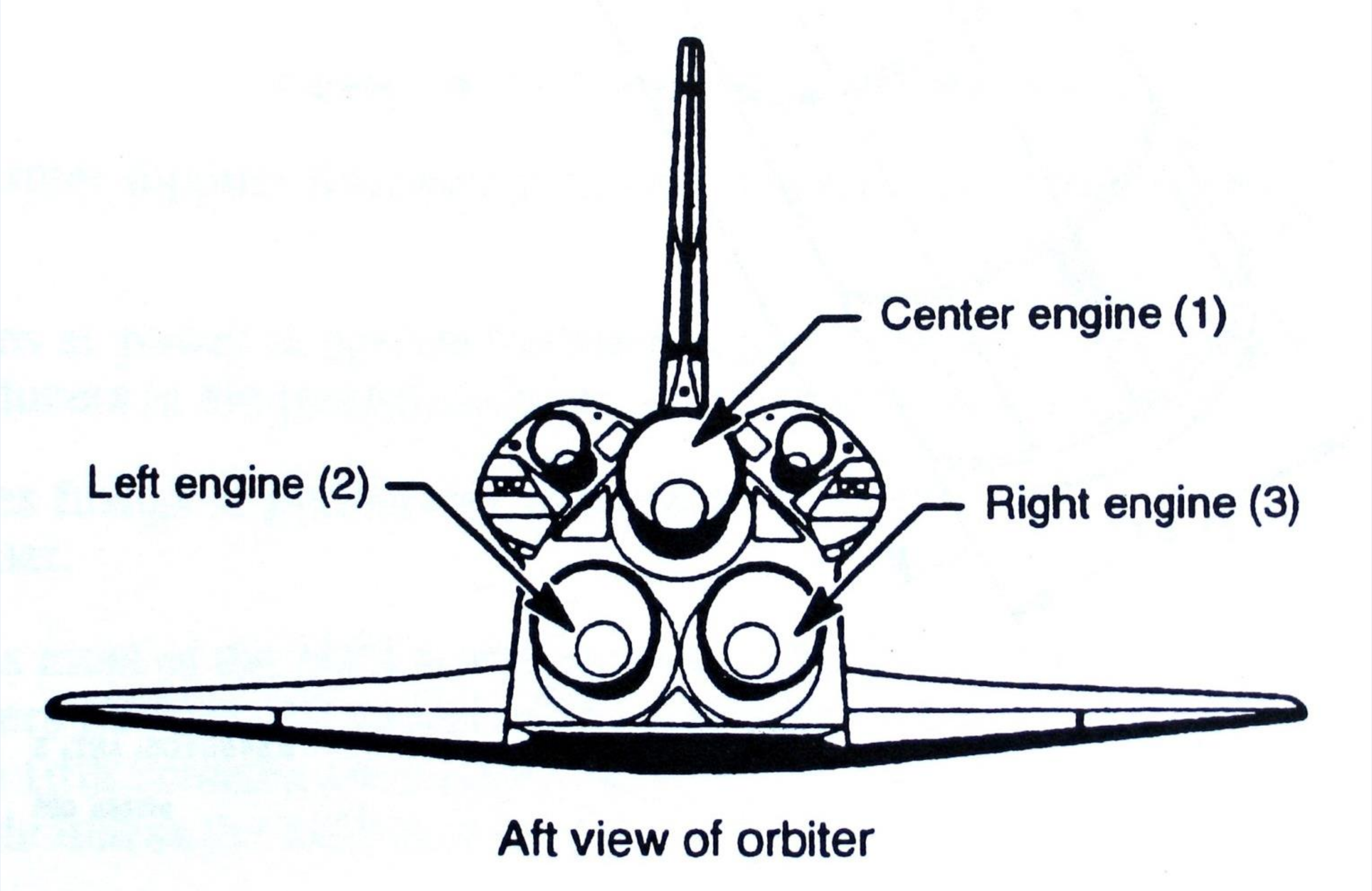
Mass of propellant needed

$$\Delta V = g I_{sp} \log_e \left(\frac{m_i}{m_f} \right) \Rightarrow \begin{cases} m_p = m_i \left[1 - \exp\left(- \frac{\Delta V}{g_0 I_{sp}} \right) \right] \\ m_p = m_f \left[\exp\left(\frac{\Delta V}{g_0 I_{sp}} \right) - 1 \right] \end{cases}$$



$$m_p = m_i - m_f$$

- m_i : initial vehicle mass (kg).
- m_f : final vehicle mass (kg).
- m_p : propellant mass consumed to produce the given ΔV (kg).
- ΔV : velocity increase of the vehicle (m/s).
- g : gravitational acceleration 9.81 (m/s^2).
- I_{sp} : specific impulse.



4.4.2 More on spacecraft propulsion

Space Mission Design and Operations

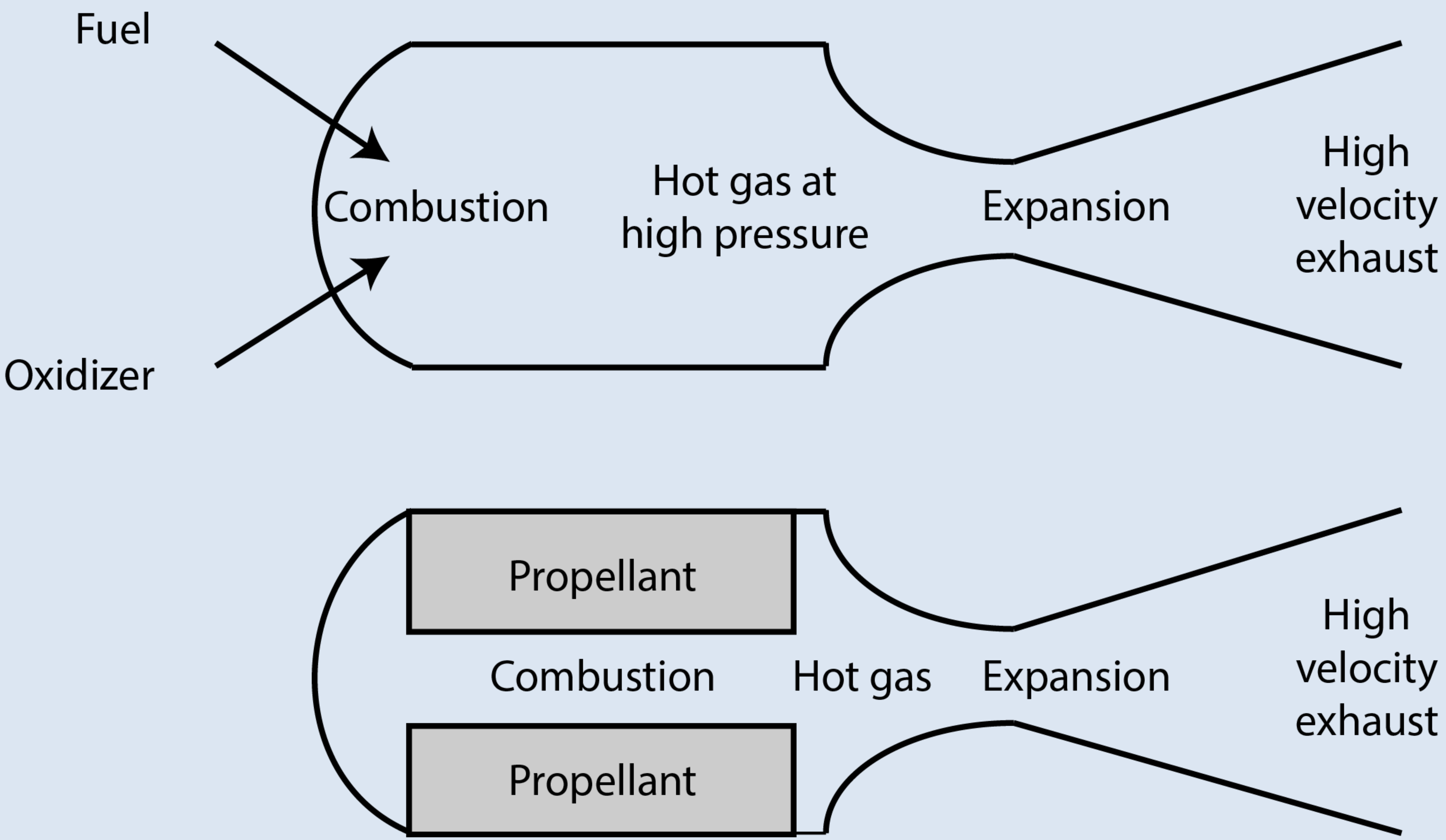
Prof. Claude Nicollier

Credits: Documention of the training division for NASA astronauts in the 90's.

Liquid and solid propellant rocket motors

In addition to the 3 orbiter-attached liquid propellant main engines (SSMEs with LH2 and LOX), the Space Shuttle had 2 solid rocket boosters (SRBs) with a central cavity in order to increase the surface of the burning propellant generating thrust.

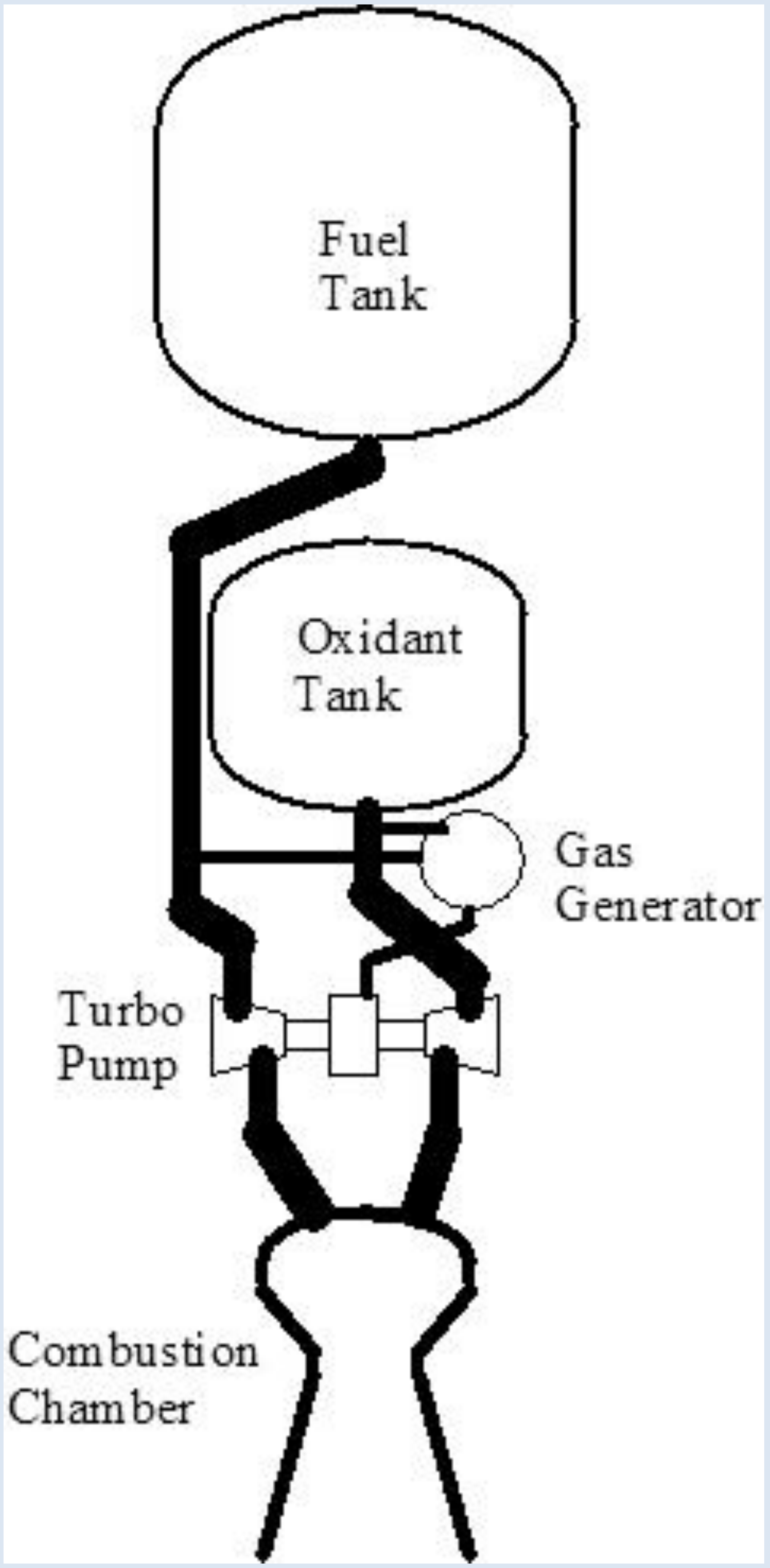
The same for Ariane 5 solid propellant boosters.



Liquid propellant rocket engine – Viking 5C example

This motor was used on the Ariane 4 launcher with hypergolic propellant, UH25 which was unsymmetrical monomethylhydrazine as fuel, and nitrogen tetroxide as oxidizer.

The specific impulse of this system was about 350-380 sec, less than a cryogenic motor using LH2 and LOX.

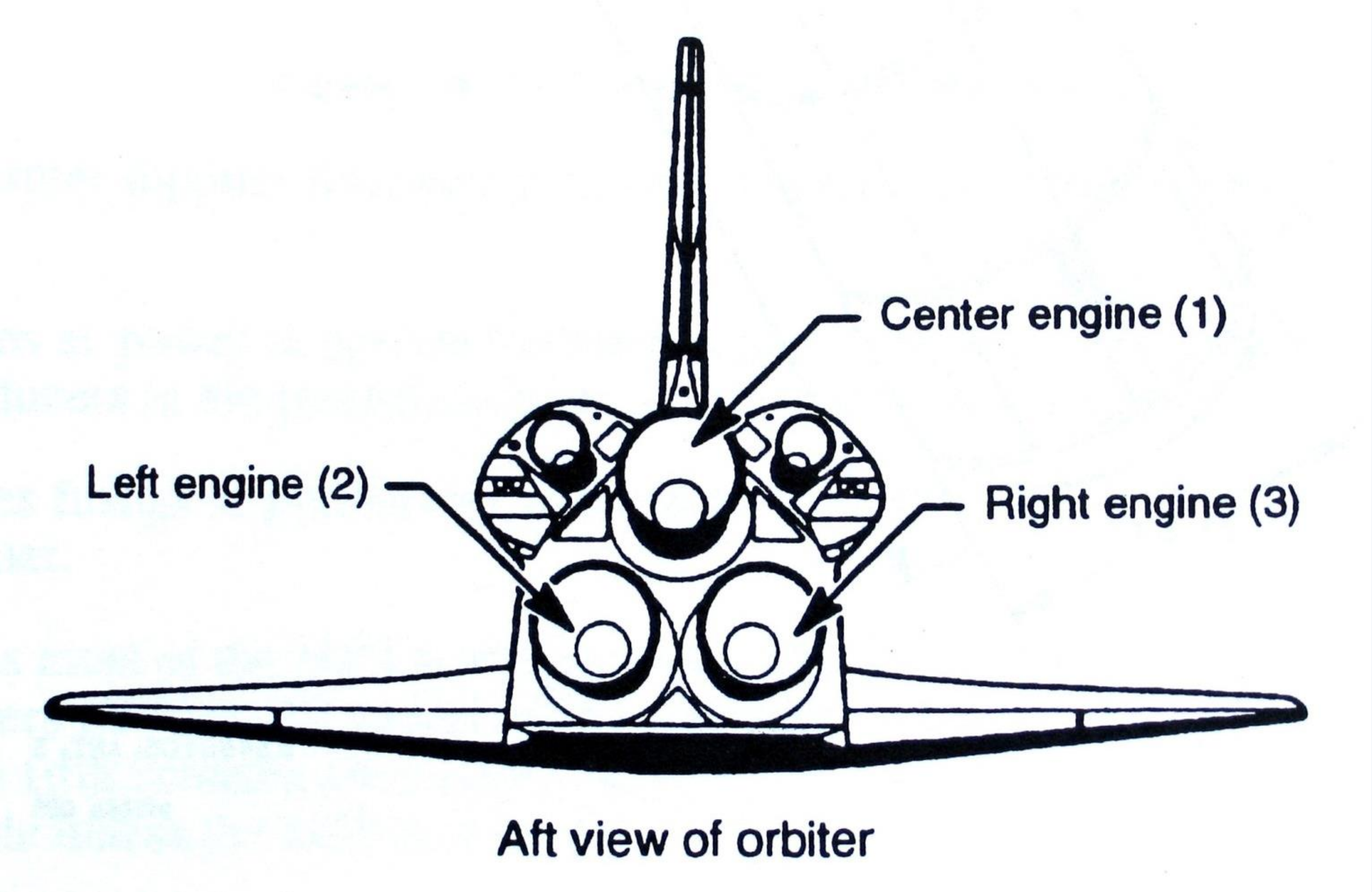


Credits: Wikipedia, Sanjay Acharya

Space Shuttle Main Engines – SSME

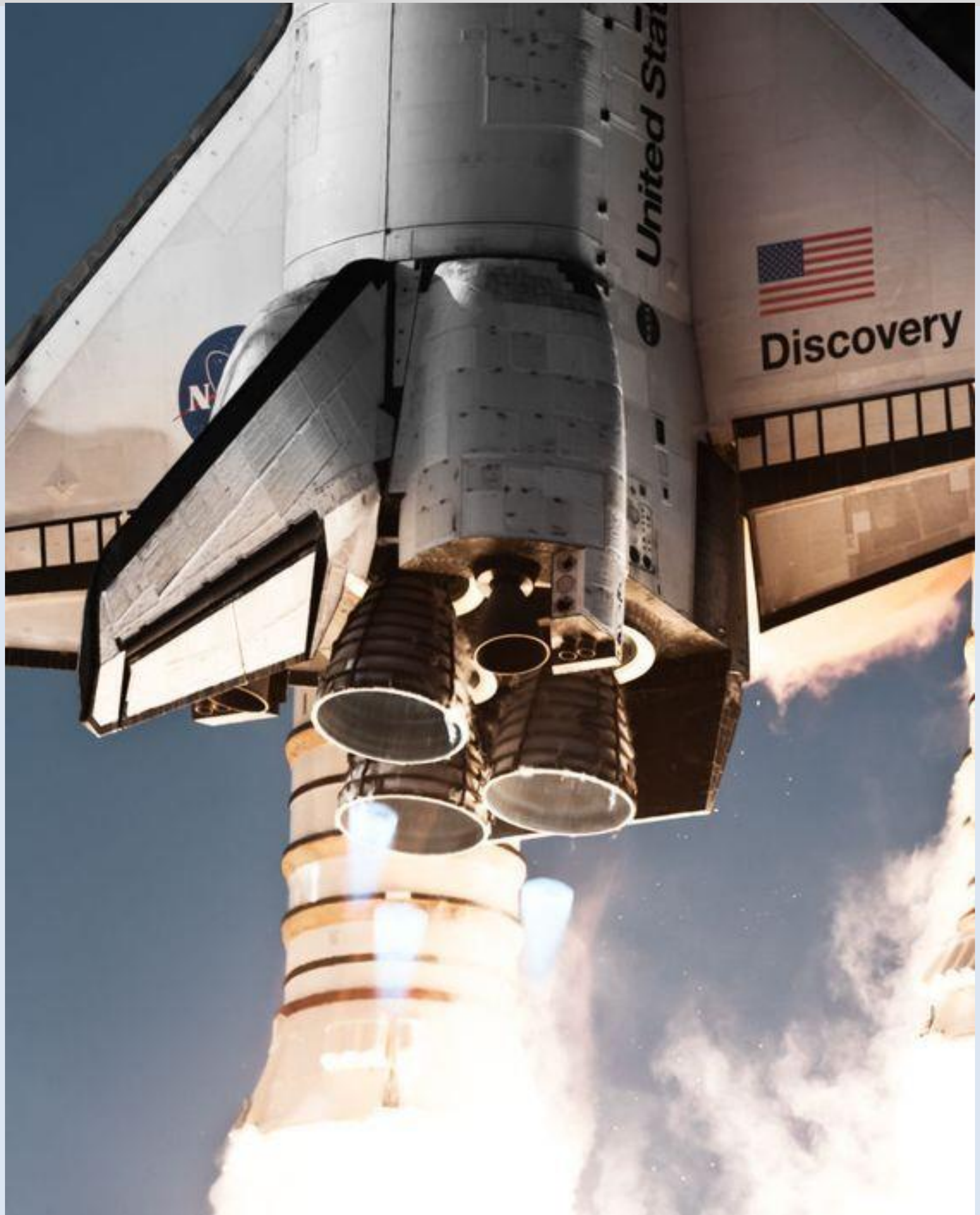
Three cryogenic engines with LH2 as fuel and LOX as oxidizer.

Oxidizer and fuel were contained in an external tank to which the Orbiter was attached.

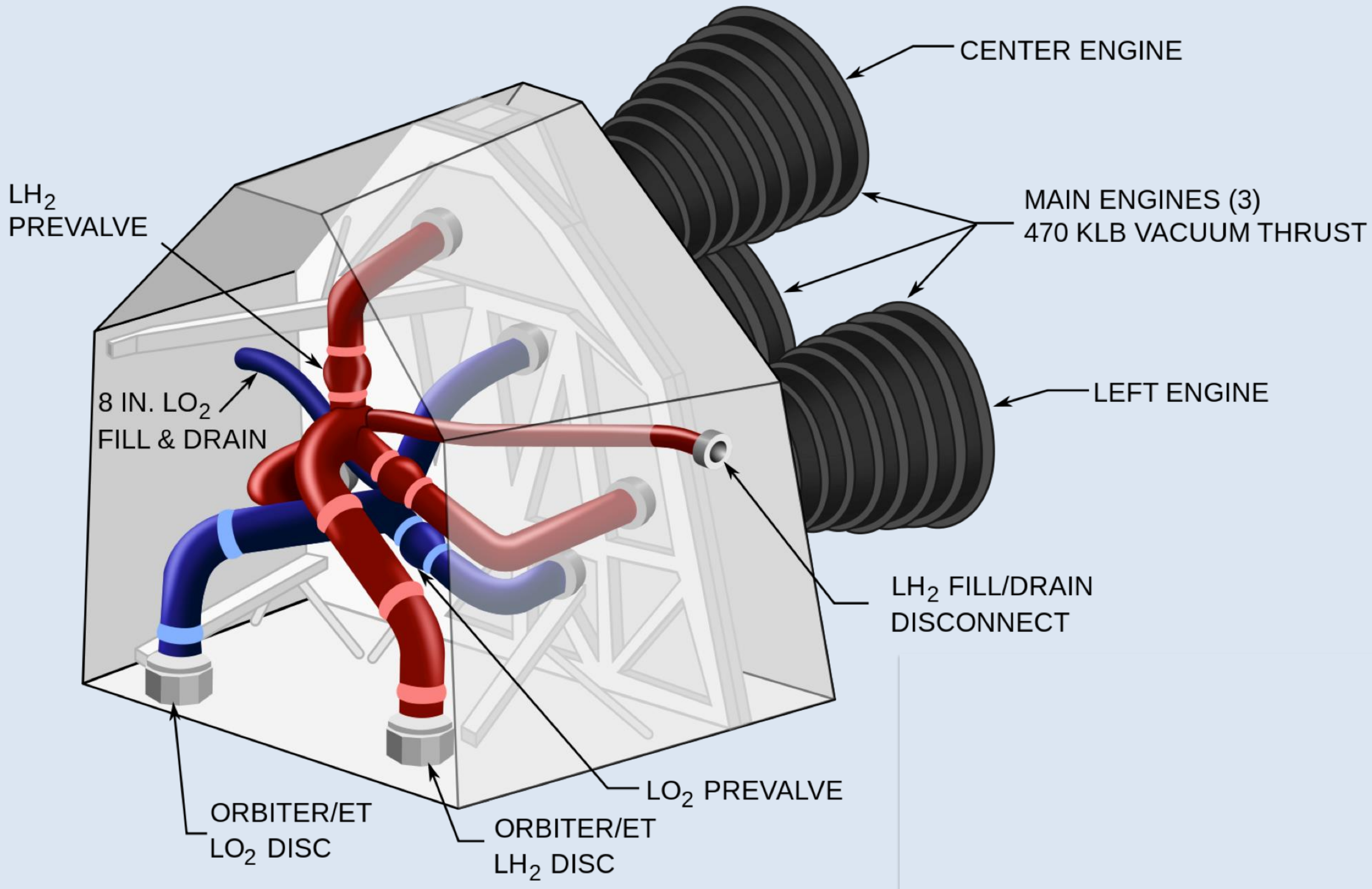


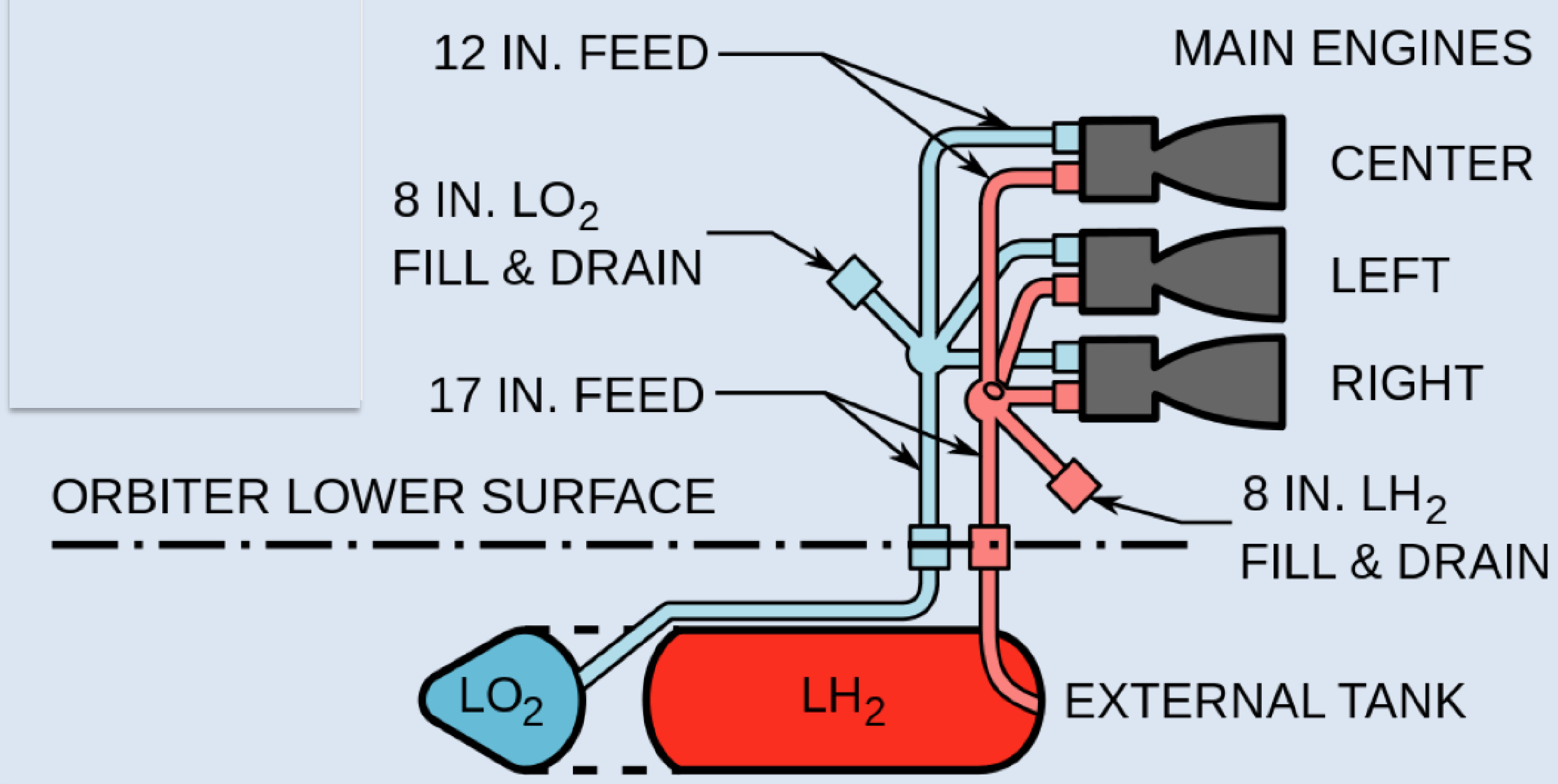
Credits: Documention of the training division for NASA astronauts in the 90's.

Space Shuttle Main Engines – SSME



SSME – Fuel and oxidizer supply

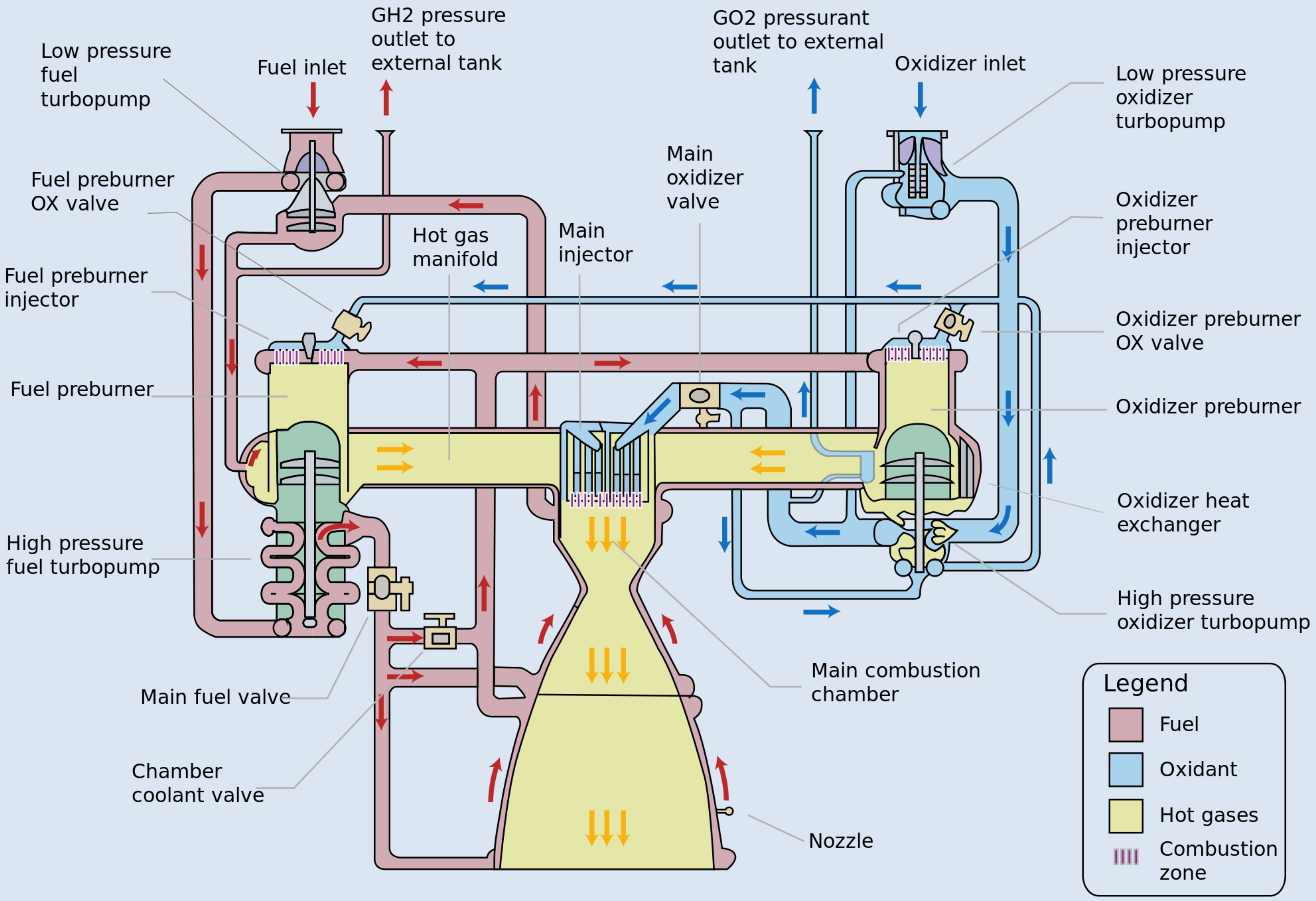




The liquid-oxygen tank contained about 1,400,000 lbs. of LOX. The liquid-hydrogen tank contained about 230,000 lbs. of LH2.

Credits: NASA, JSC, Booster Systems Briefs, October 1, 1984

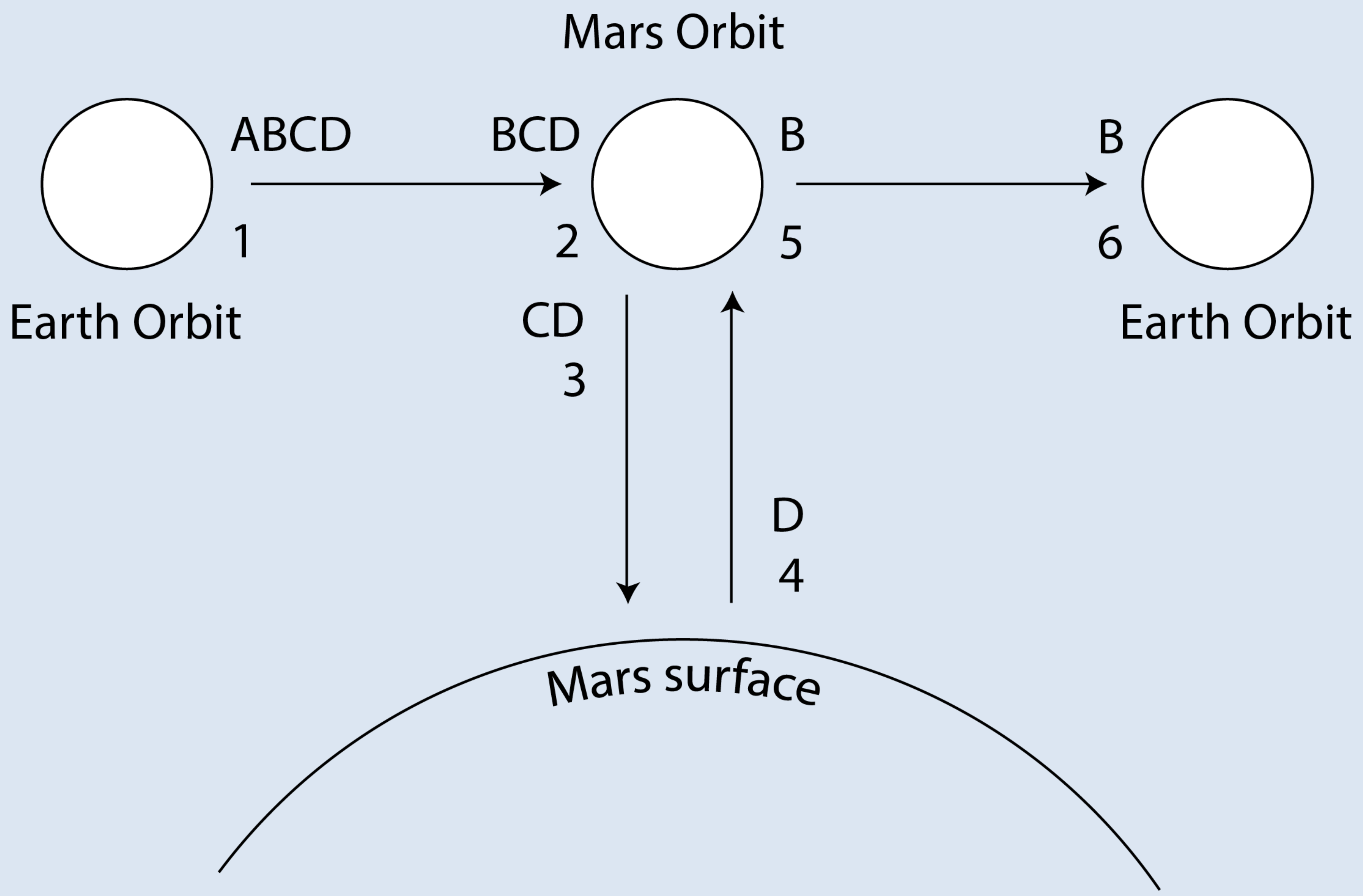
SSME – Schematic



Two-stage combustion in the SSME:
the first stage was in the high-pressure turbopumps for fuel and oxidizer. The second stage is was in the main combustion chamber to produce thrust.

Credits: « Shuttle Press Kit.com », Boeing, NASA and United Space Alliance, October 6, 1998

Propellant needed for a mission to Mars



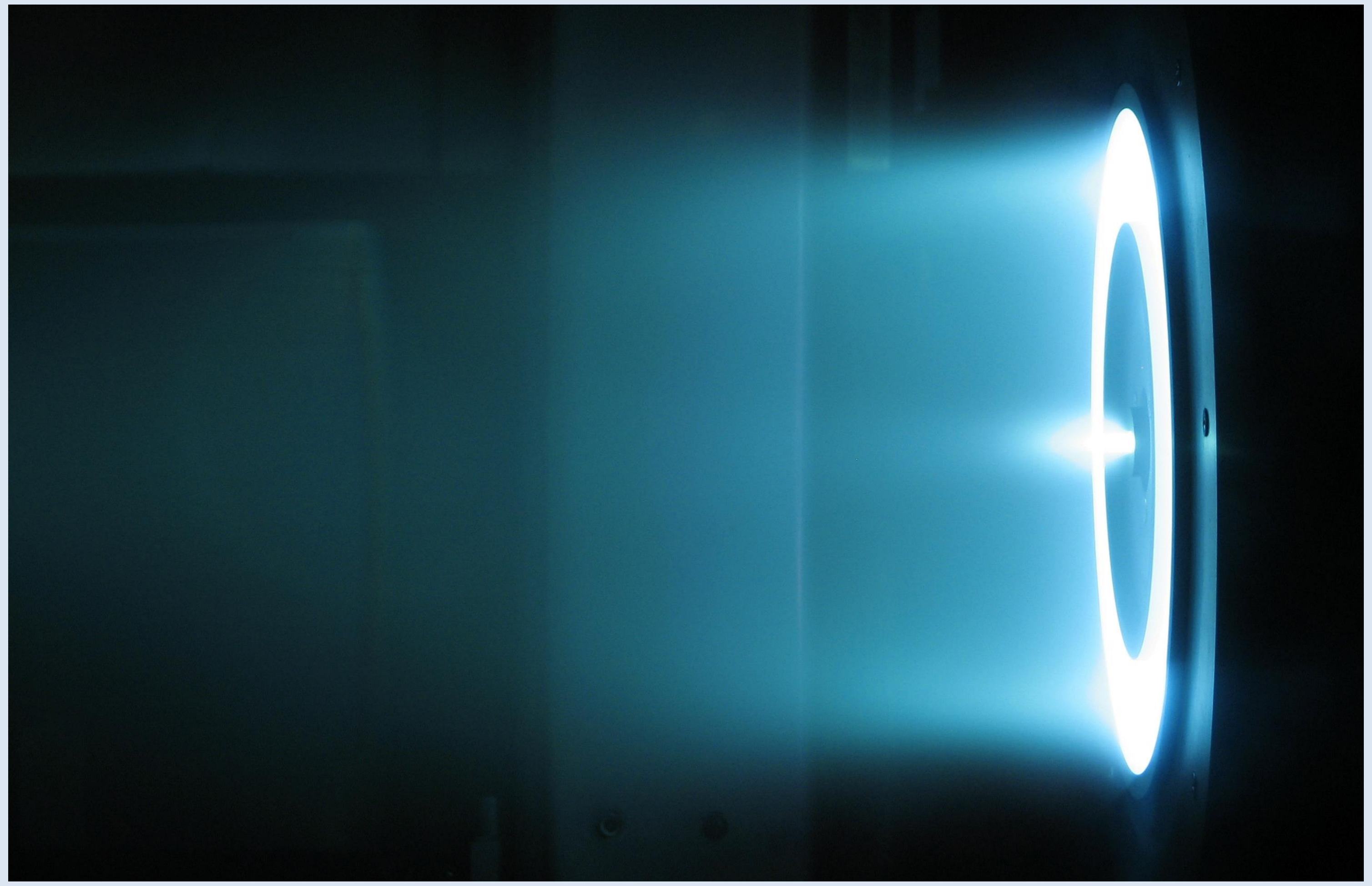
Vehicles

A Discarded Booster
B Earth-Mars Transfer Vehicle
C Mars Lander
D Ascent Vehicle

Maneuvers

1 LEO to MTO
2 MTO to LMO
3 LMO to Surface
4 Surface to LMO
5 LMO to ETO
6 ETO to LEO

Based on Tsiolkovsky equation, about 90% of the mass of the spacecraft leaving the Earths' surface is propellant, if the mass of the payload is increased, the propellants needed to leave the gravity well of the Earth have to be considerably increased. The need for High Isp, or altermative propulsion system for Mars exploration is clear!



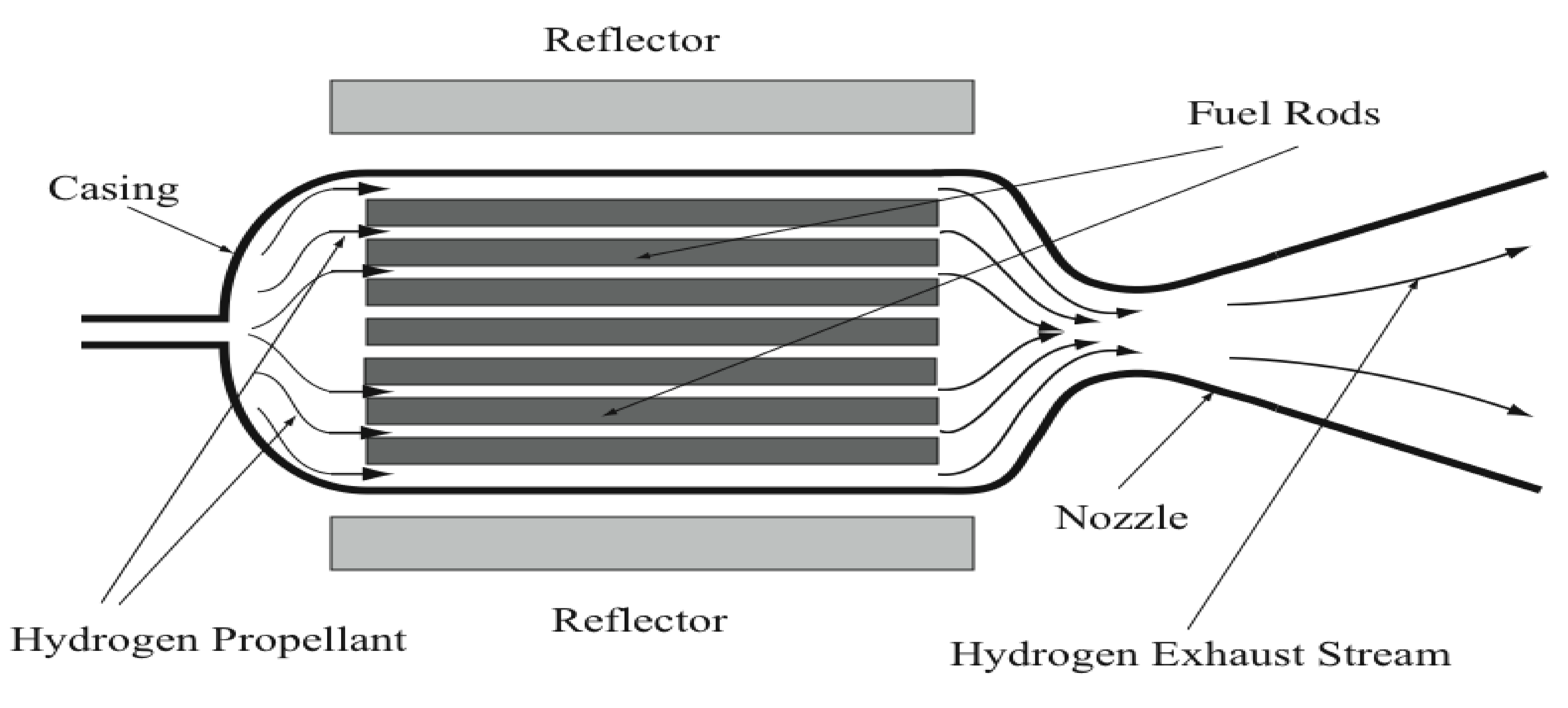
Credits: Nasa, JPL-Caltech

4.4.3 Nuclear and electric propulsion

Space Mission Design and Operations

Prof. Claude Nicollier

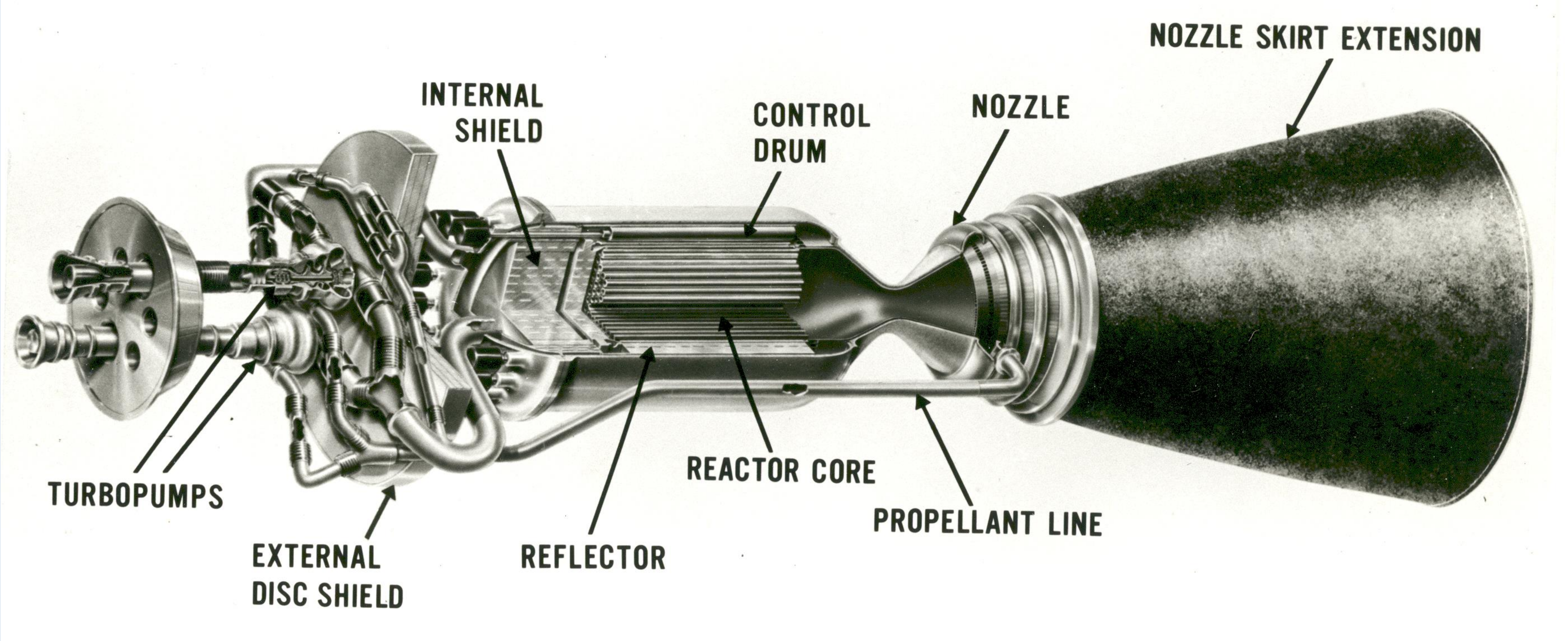
Nuclear rocket principle



- Hot fuel rods heat hydrogen propellant.
- The hot hydrogen expands in the nozzle as in a chemical rocket motor.

NERVA – From 1968 to 1972

NERVA = Nuclear Engine for Rocket Vehicle Application

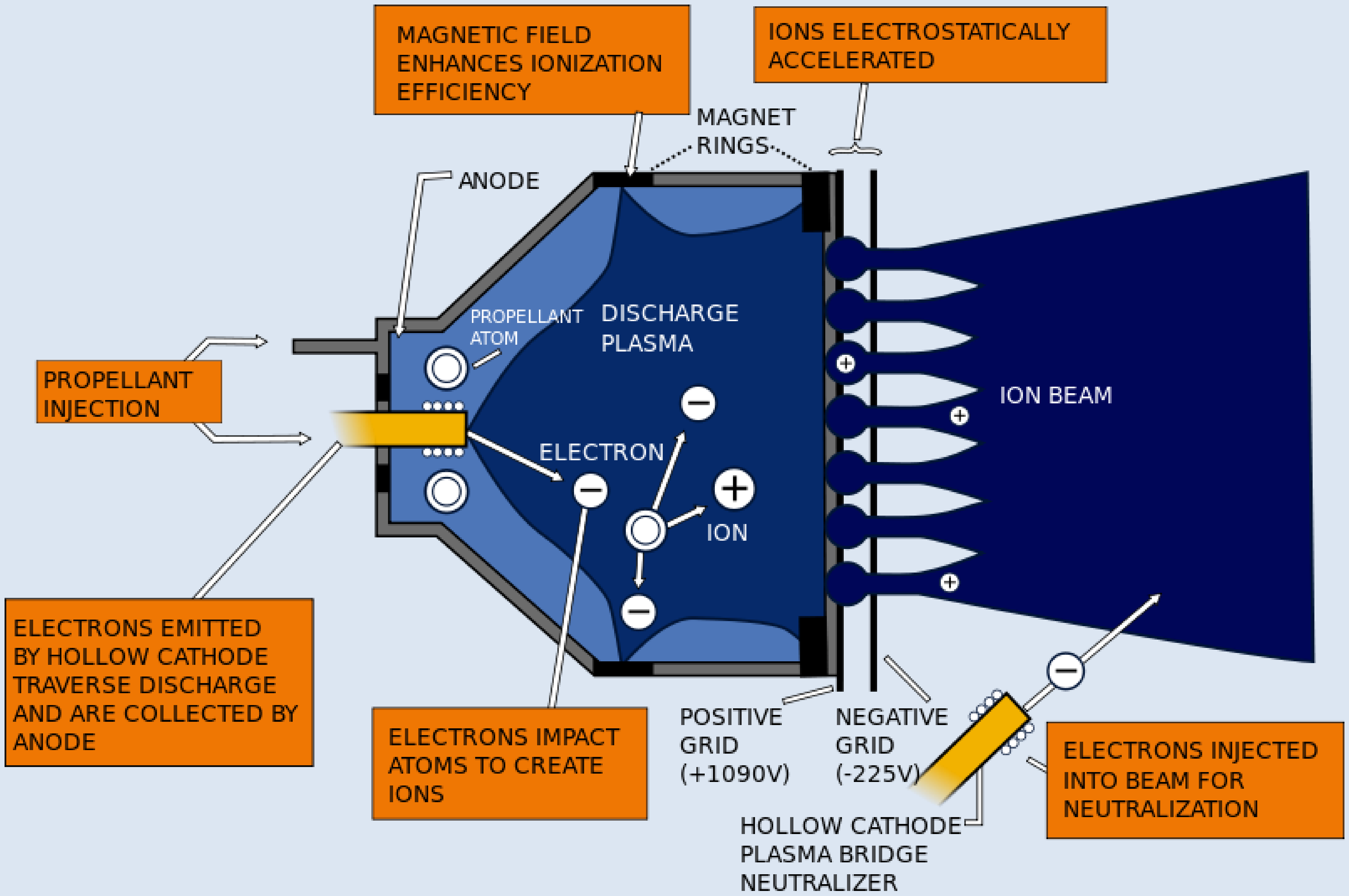


Credits: PD, USGOV, NASA

Electric or ion propulsion

Ionization of the propellant material, and acceleration with an electric field. Higher exhaust velocities than with a liquid-fueled or solid propellant rocket engine.

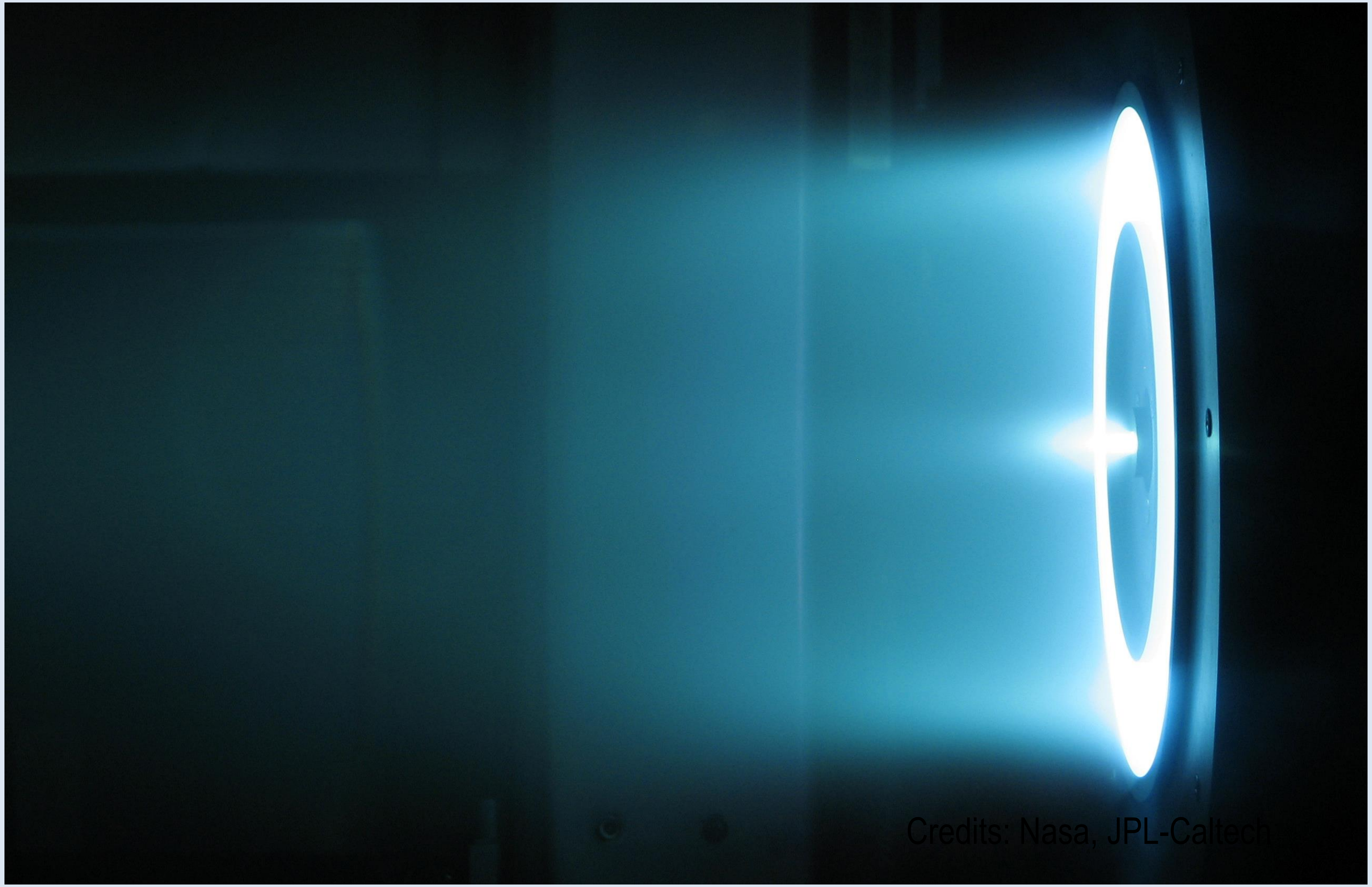
There is very high ejection velocity and high efficiency, but relatively low thrust, of the order of a fraction of a Newton.
Such a system can be used for propulsion in space, but not for leaving the Earth's surface and bring a spacecraft to orbit.



Credits: Wikipedia, Chabacano, retrieved from NASA

An example of electric propulsion

- Exhaust velocities of typically 10 to 100 km/s.
- Thrust values of typically 10^{-2} N.
- Here a 6 kW Hall thruster in operation at the NASA Jet Propulsion Laboratory (JPL) →



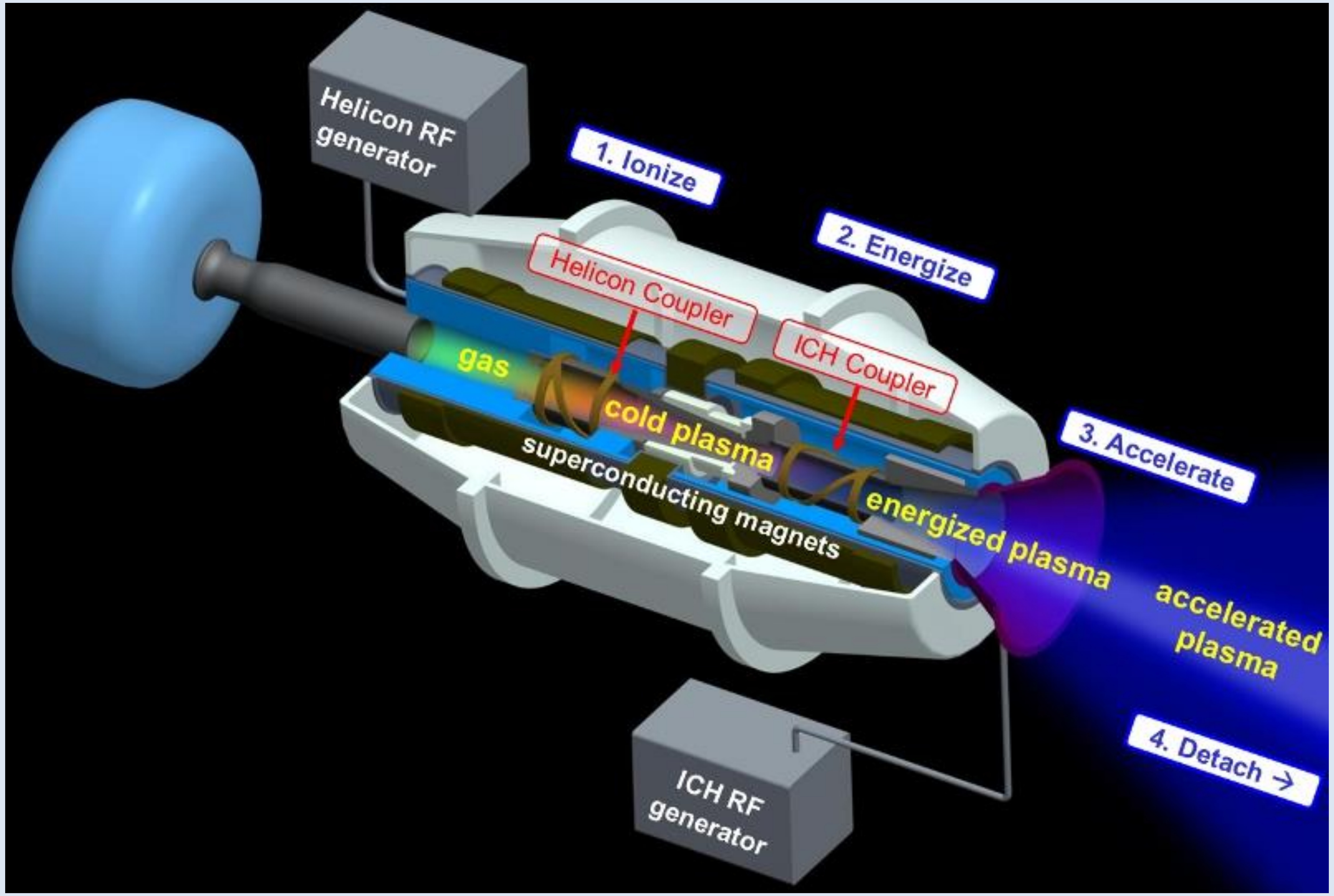
VASIMR project – Ad Astra Rocket Company

- Variable specific impulse magnetoplasma rocket.
- Development since 1977 under the lead of Franklin Chang Diaz.

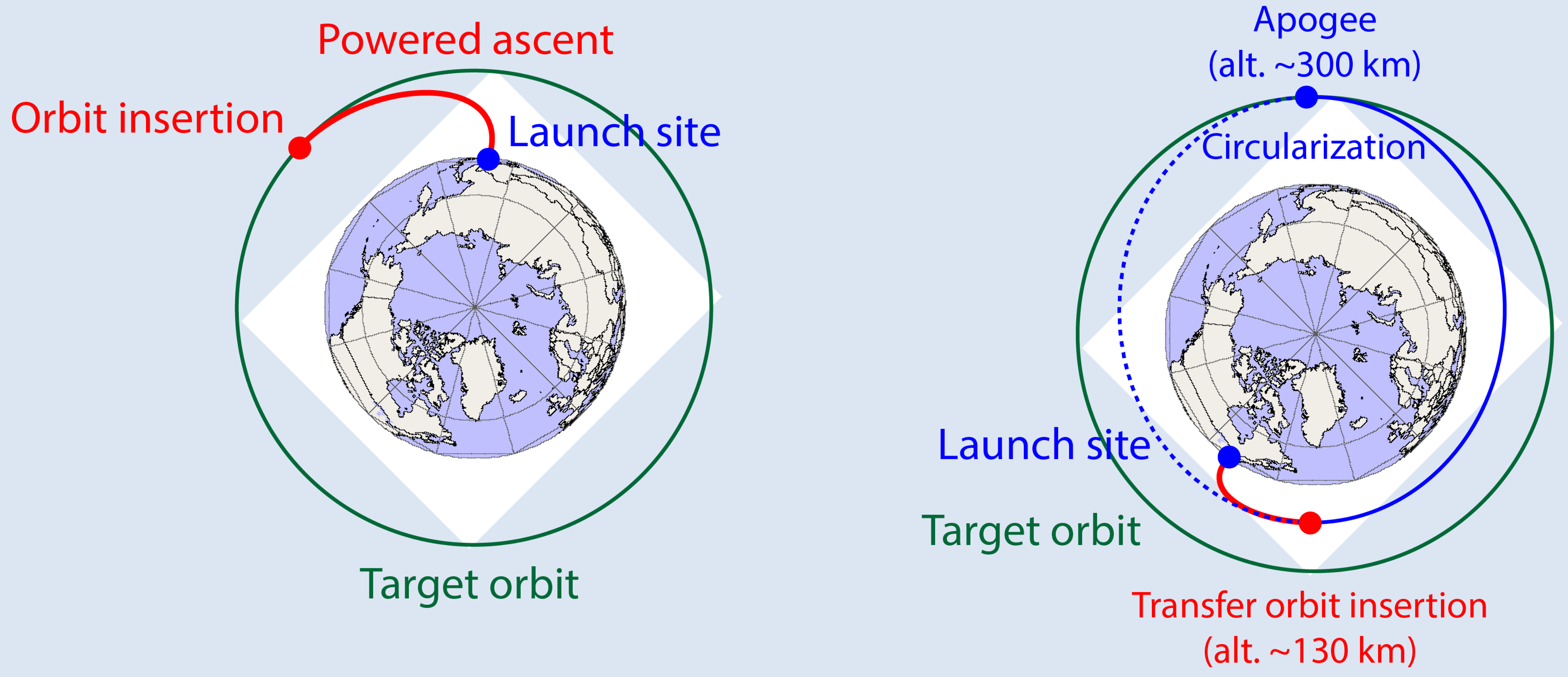
Ion engine with possibility to vary the thrust and the Isp.

It uses the properties of plasmas and confinement of plasma in an electromagnetic field.

This engine requires a lot of electrical energy, generated by solar panels or a nuclear reactor.



Credits: Nasa, JPL-Caltech



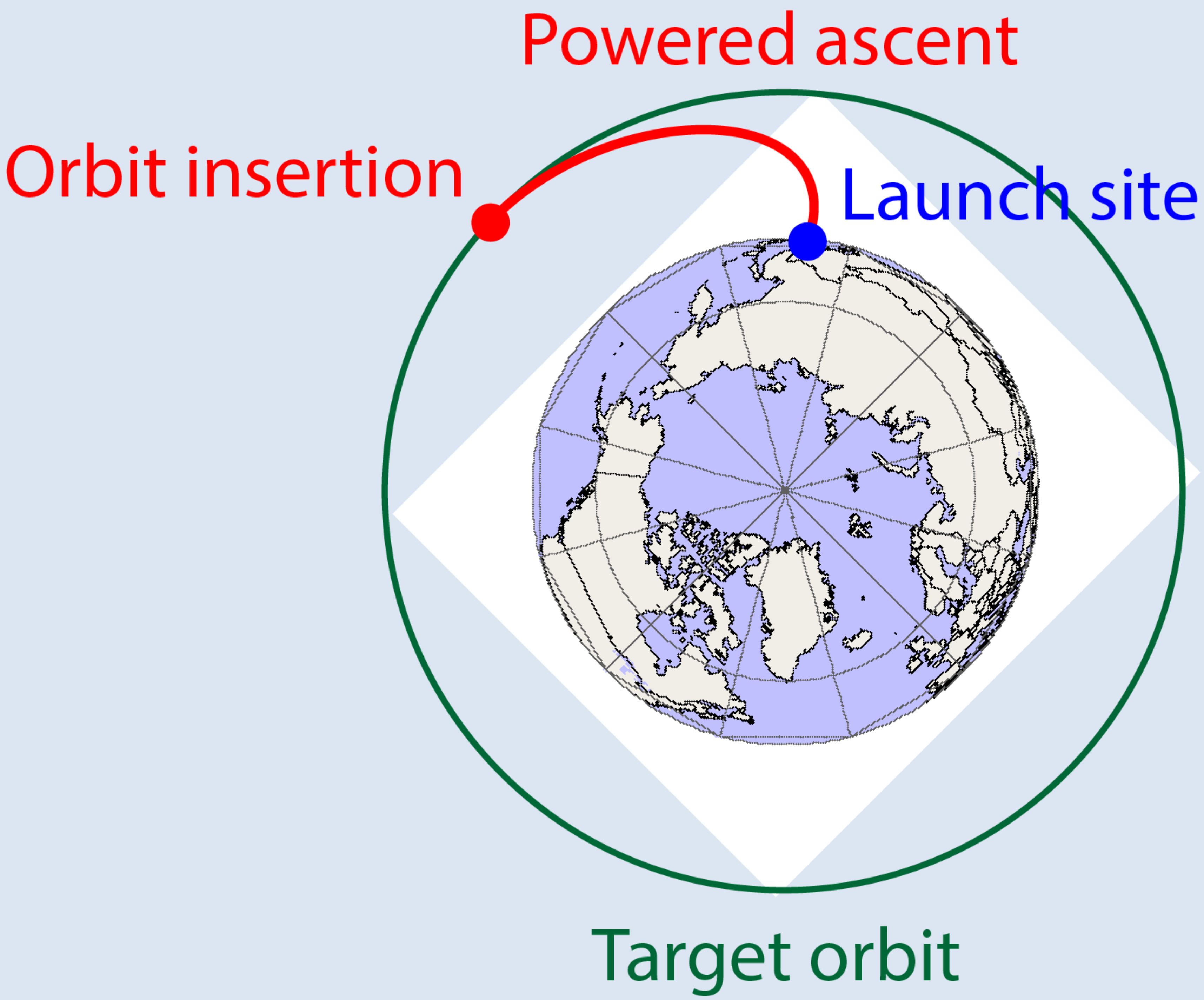
4.5.1 Ascent to space and re-entry

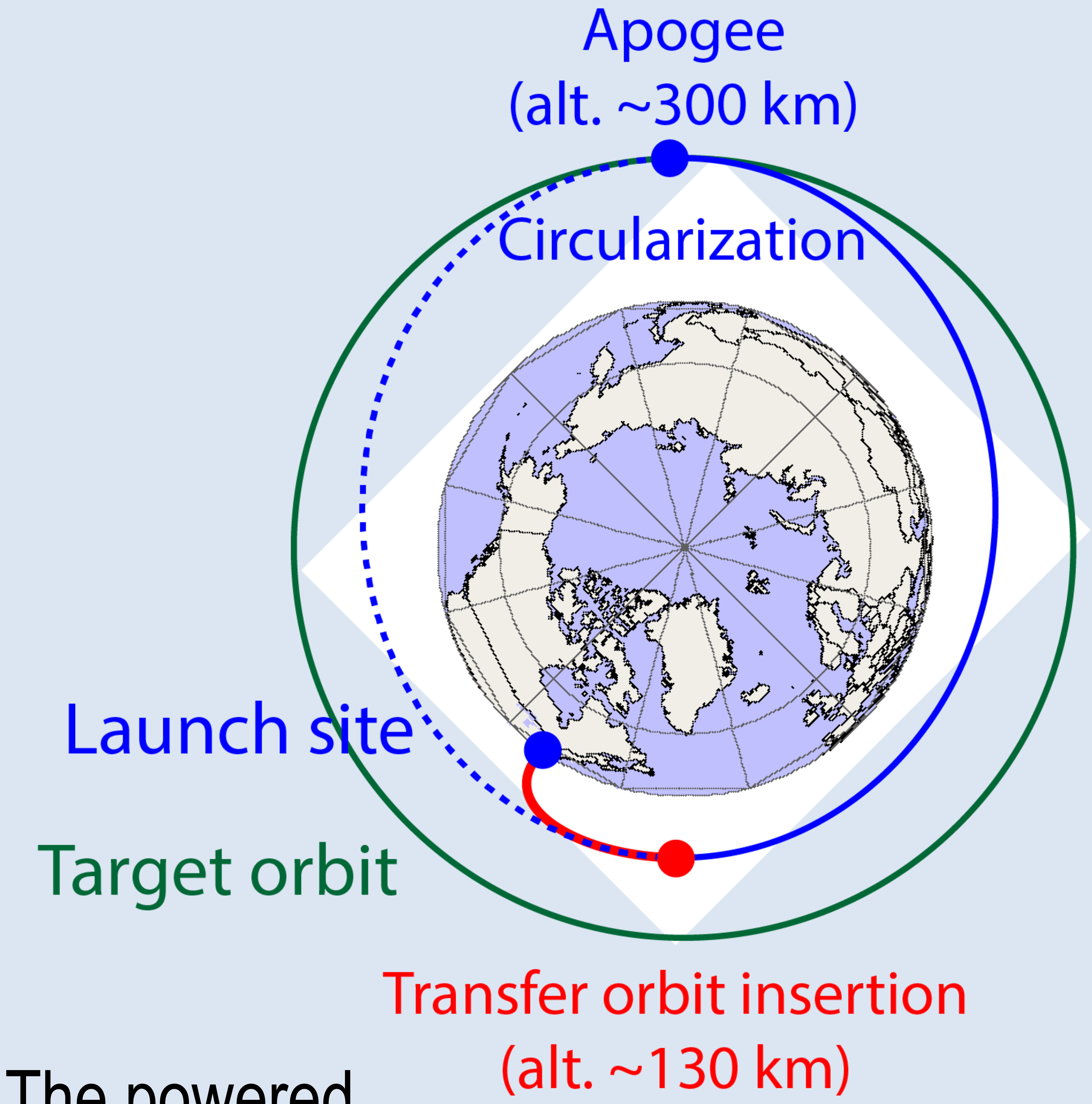
Space Mission Design and Operations

Prof. Claude Nicollier

Orbit insertion

Orbit insertion consists in bringing a spacecraft to a desired stable orbit after a launch from the Earth surface.

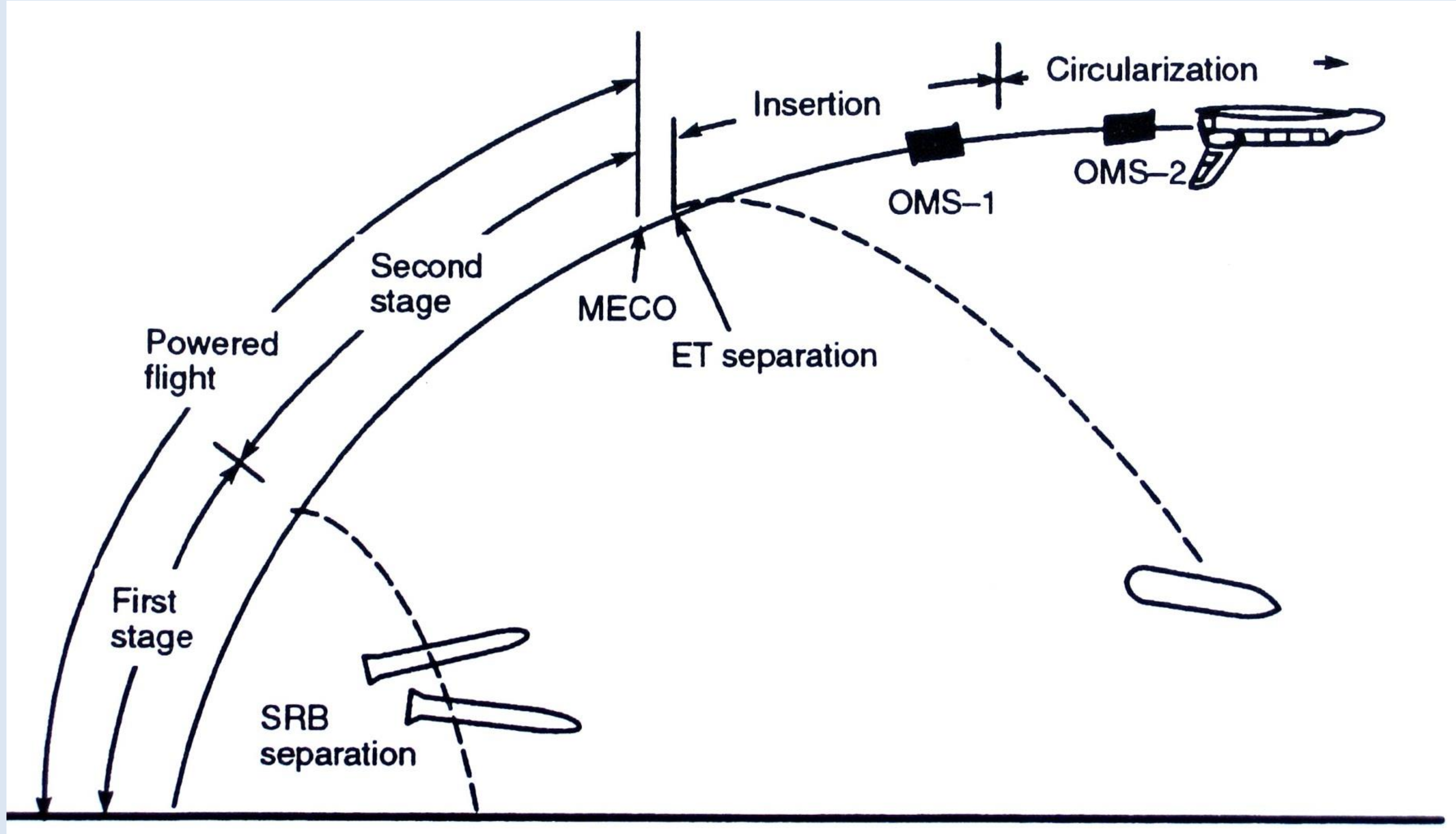




Direct insertion into orbit (left), or via a **transfer orbit** (right). The powered ascent uses either one, two or three stages until orbit insertion.

Shuttle ascent to orbit

Shuttle mission STS 41G, 1984



SRB: Solid Rocket Booster MECO: Main Engine Cut Off
ET: External tank OMS-2: Posigrade burn at the apogee of the transfer orbit to circularize the trajectory. OMS-1 optional and only if needed to reach desired apogee altitude.

Credits: Documention of the training division for NASA astronauts in the 90's.

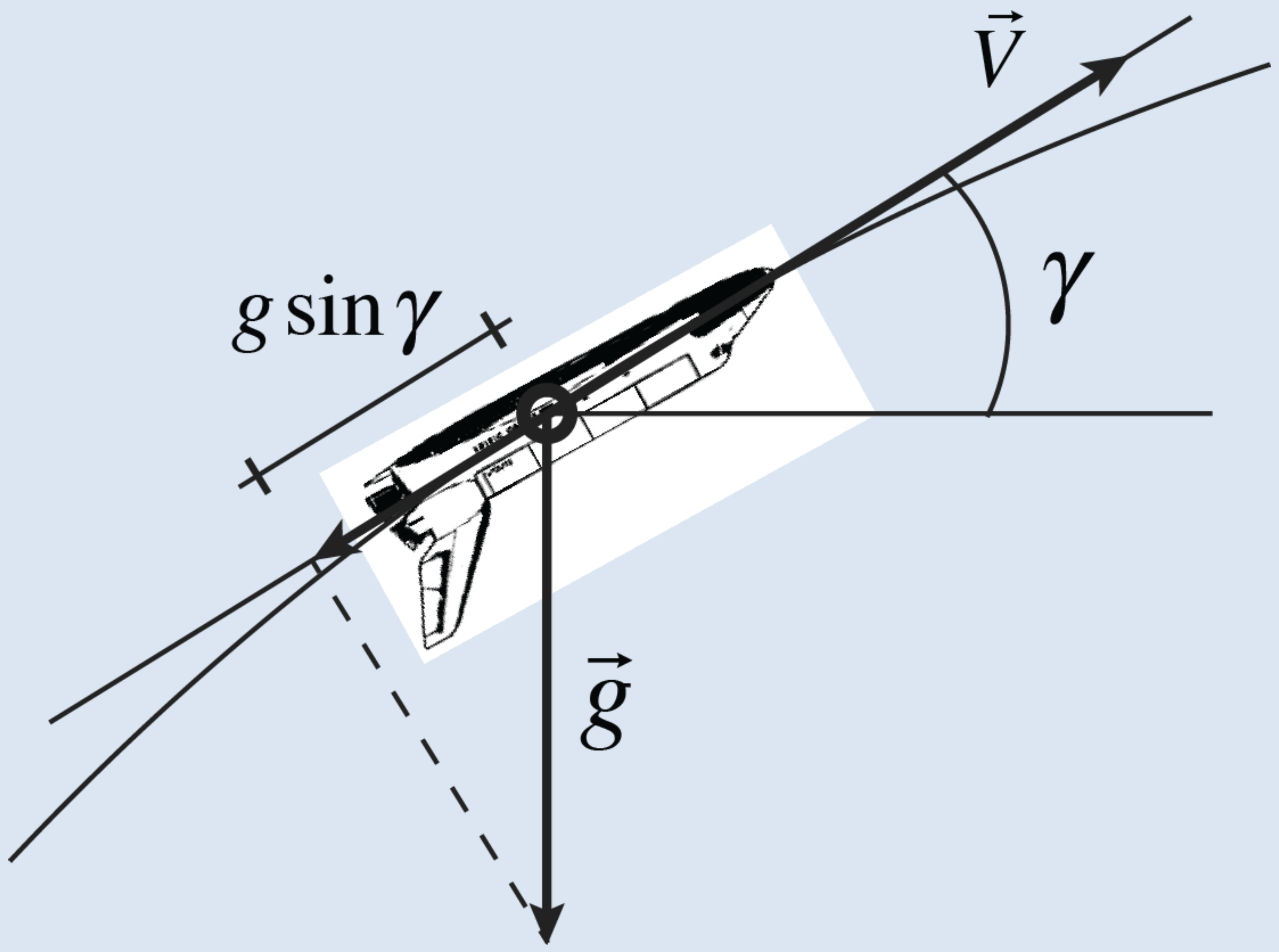
Losses during ascent to orbit

ΔV achieved by a rocket **in reality**, in case of an ascent to orbit from the Earth surface:

$$\boxed{\Delta V = g I_{sp} \log_e \left(\frac{m_i}{m_f} \right) - \left(\int_{t_0}^{t_f} g \sin \gamma dt + \int_{t_0}^{t_f} \frac{D}{m} dt \right)}$$

Losses during ascent to orbit: **gravity loss** and **drag loss**

The planned and actual ascent trajectory is shaped to minimize these losses.



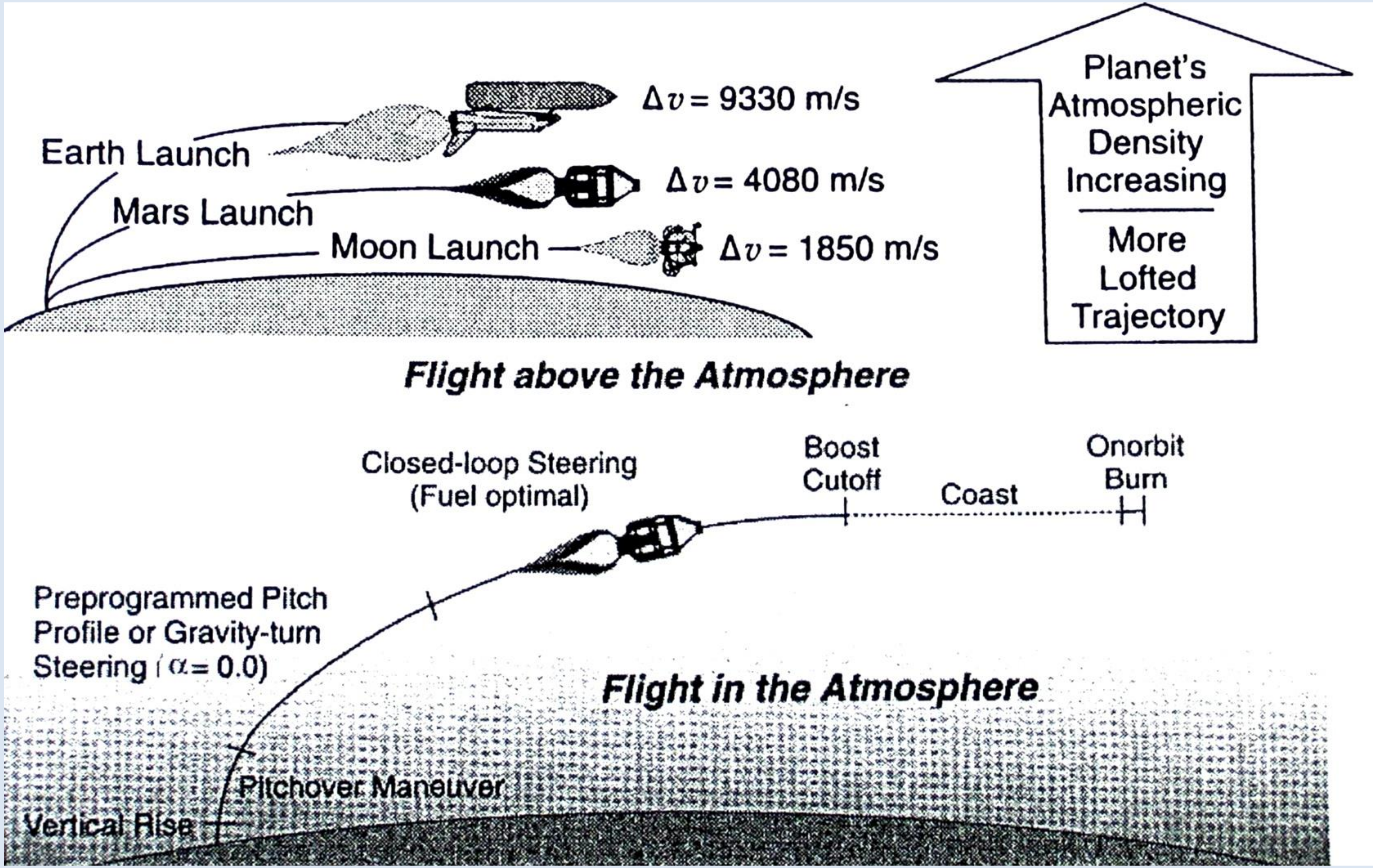
- D: drag force in Newton
- Y: flight path angle

Different cases of orbit insertion

For Earth launch, the ascent trajectory is significantly lofted because of the atmosphere.

On a planet with thinner atmosphere like Mars, loft is less necessary.

The case of the Moon: no atmosphere, only gravity loss during ascent to orbit. After a very short period of vertical launch the spacecraft tilts toward the desired direction.

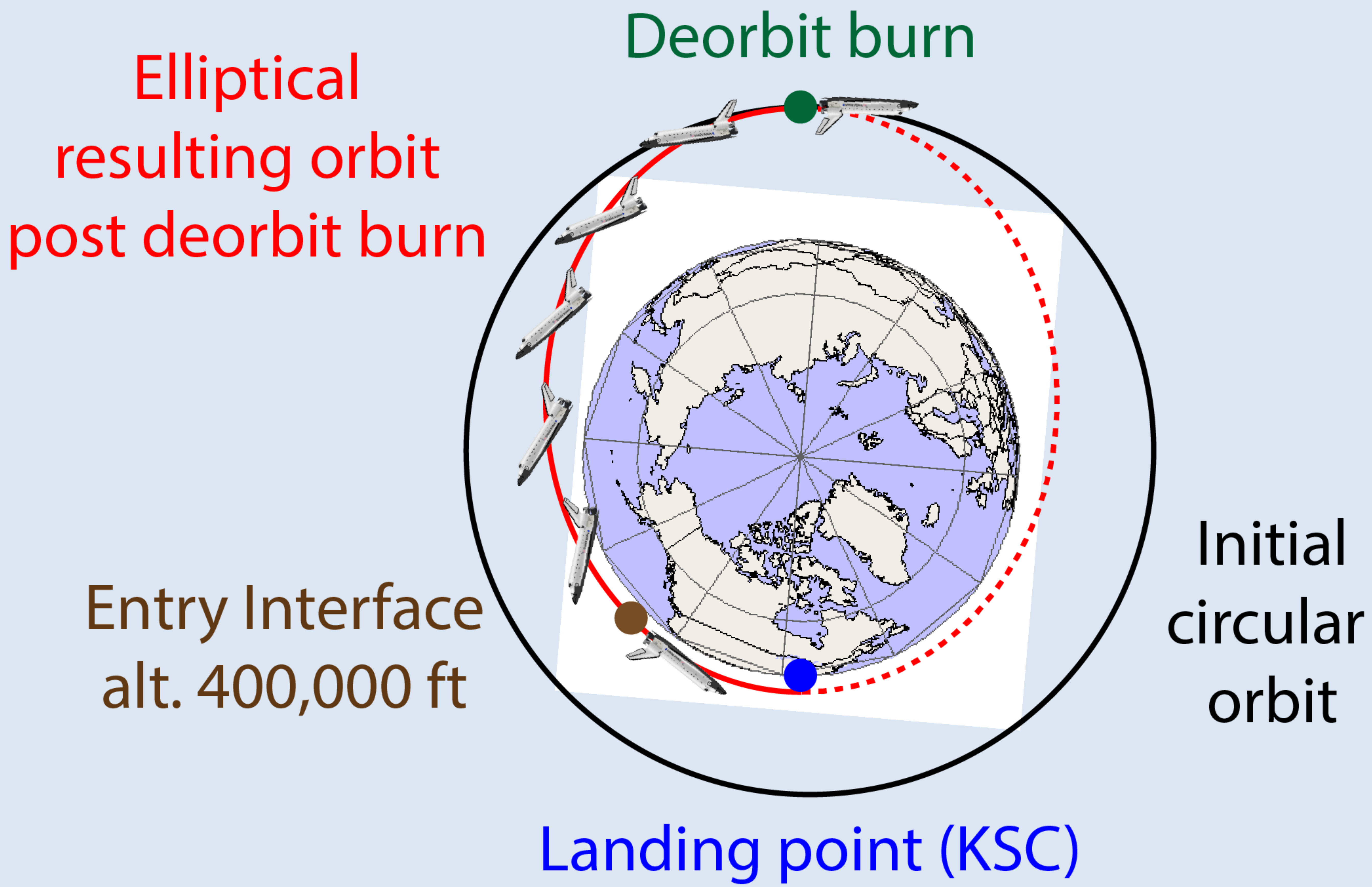


Credits: Documention of the training division for NASA astronauts in the 90's.

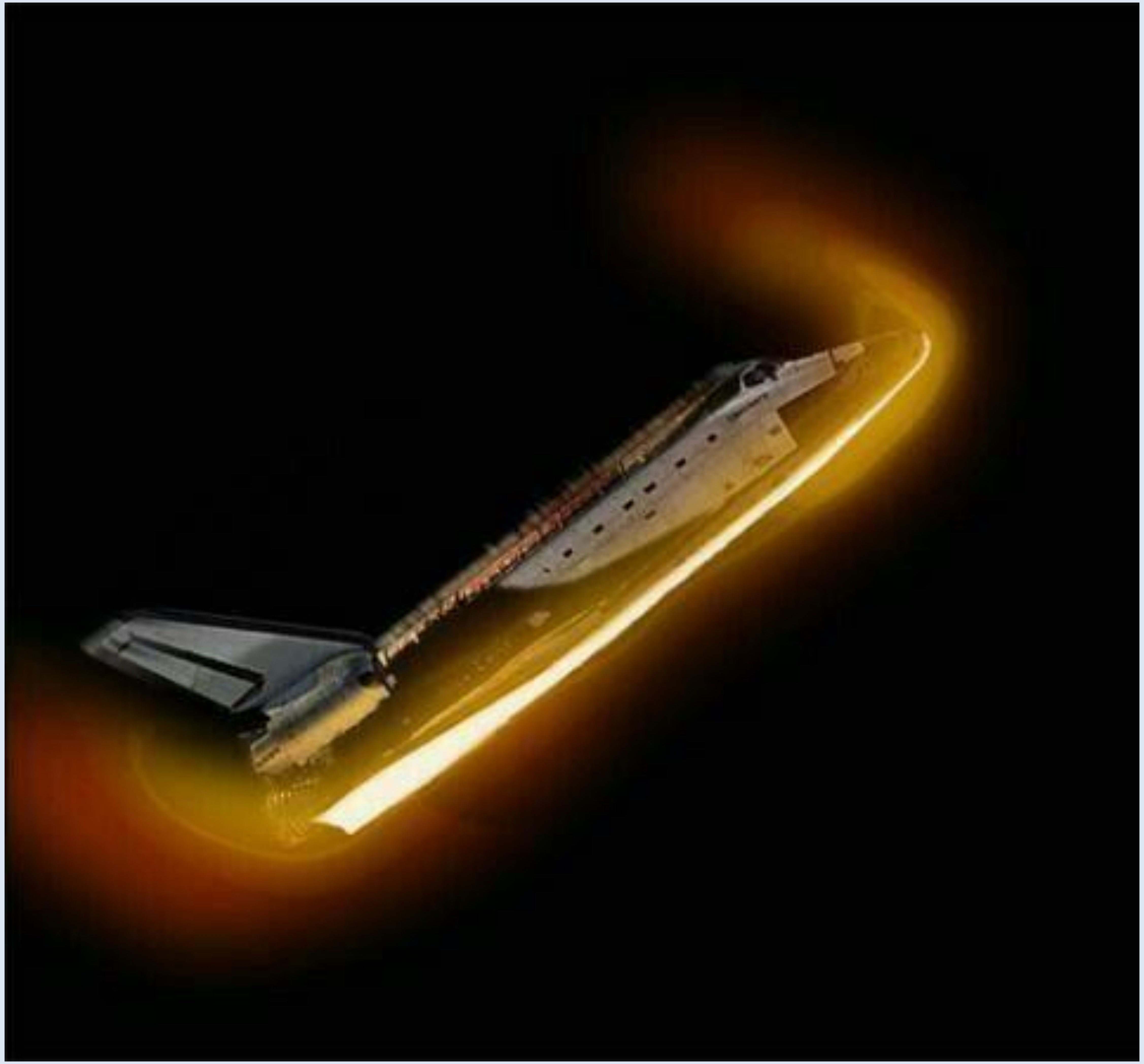
Shuttle re-entry

The deorbit burn was a braking maneuver, using the OMS engines to reduce the velocity and come to an elliptical orbit with the perigee at some height above the vicinity of the landing point.

During the early part of re-entry, the Orbiter angle of attack α was maintained at 40º in order to cause enough braking high in the atmosphere, and reduce the drag and deceleration when reaching the lower layers of the atmosphere.



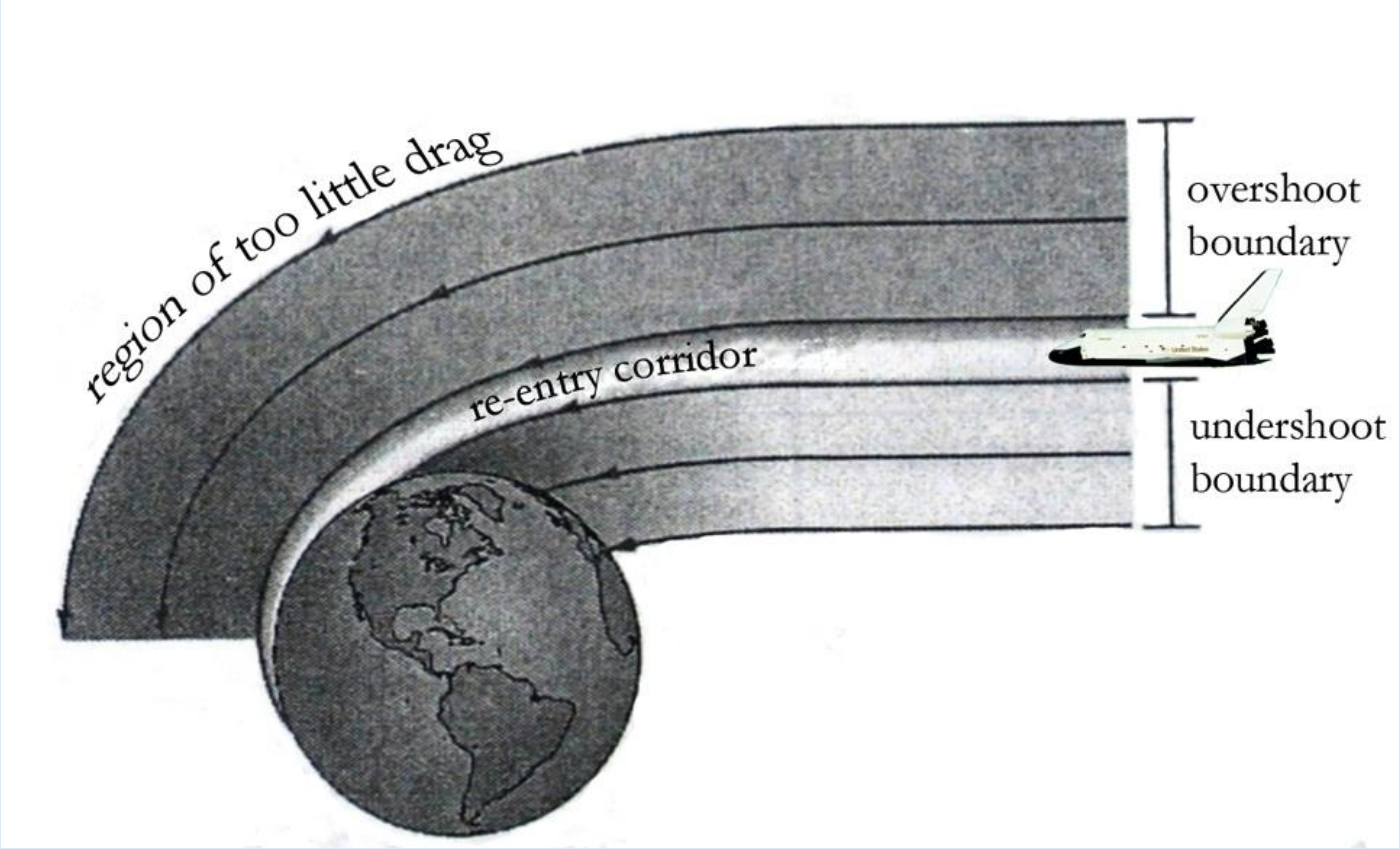
Shuttle re-entry



The Shuttle's angle of attack was precisely maintained at 40 degrees (+/- 1 degree) during the early part of re-entry, normally by the digital autopilot (DAP). It was then reduced when deceleration of 1.5 g was reached, with a gradual transition to aerodynamic flying.

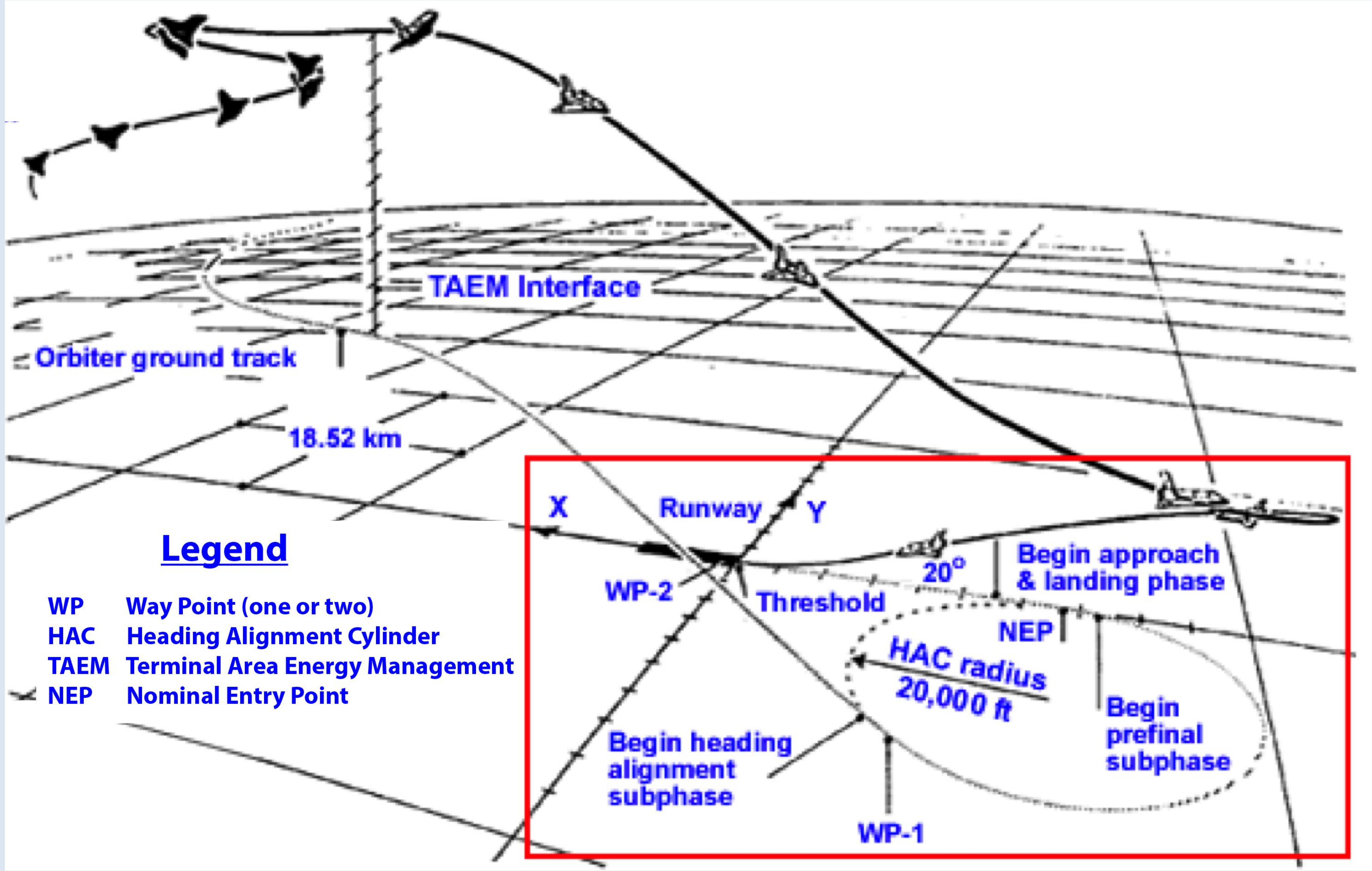
Re-entry through the atmosphere

- Entry requirements and constraints:
  - Deceleration: Human limit is about 12g's for short duration.
  - Heating: Must withstand both total heat load and peak heating rate.
  - Accuracy of landing or impact: Function primarily of trajectory and vehicle design.
  - Size of the entry corridor: The size of the corridor depends on three constraints (deceleration, heating and accuracy).



Credits: Documention of the training division for NASA astronauts in the 90's.

Shuttle approach to landing



TAEM interface: Terminal Area Energy Management interface.

HAC: Heading Alignment Cylinder

Shuttle final approach and landing

The view through the HUD (Head Up Display) in final approach, and manual landing