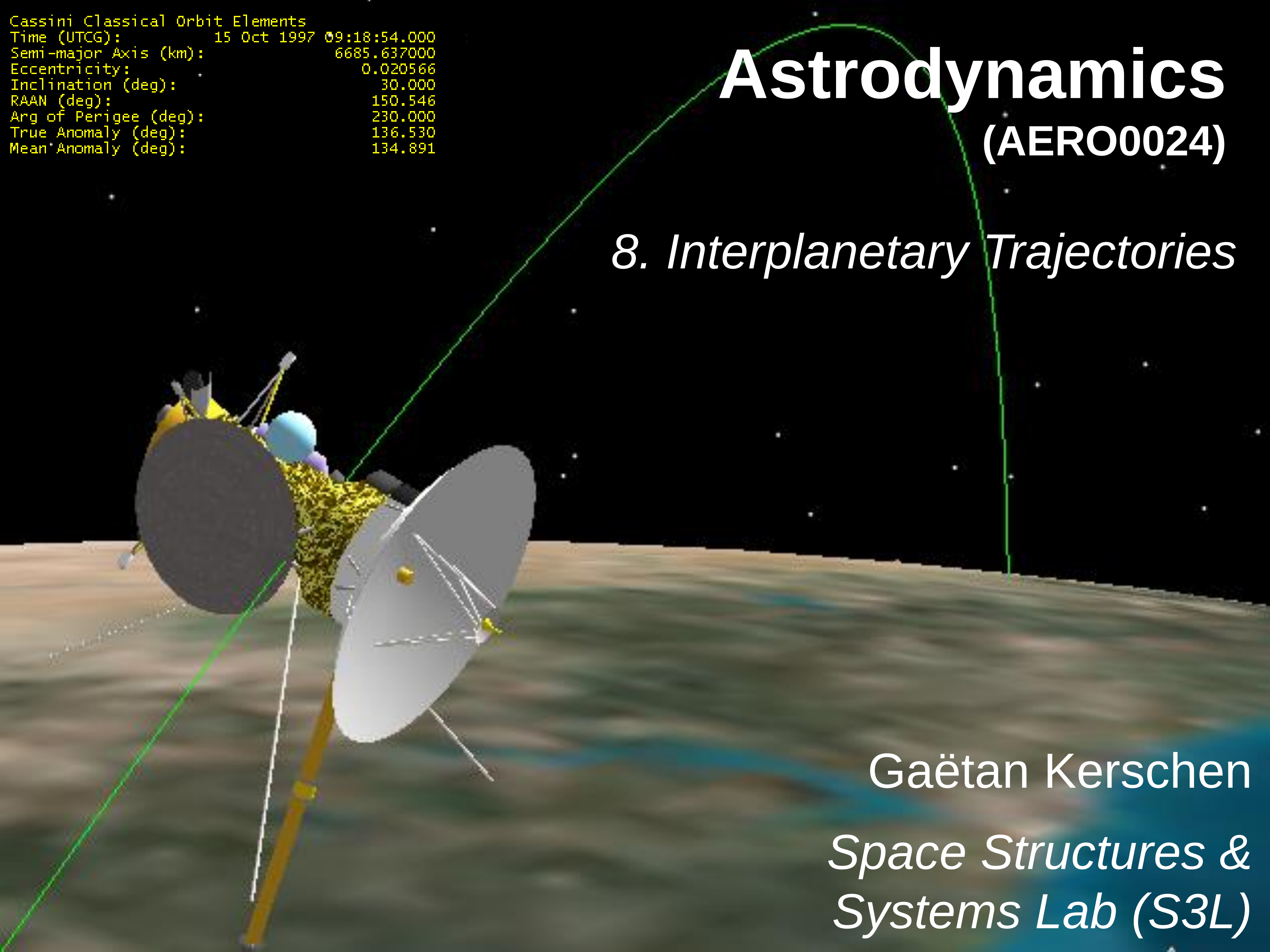
Astrodynamics
(AERO0024)

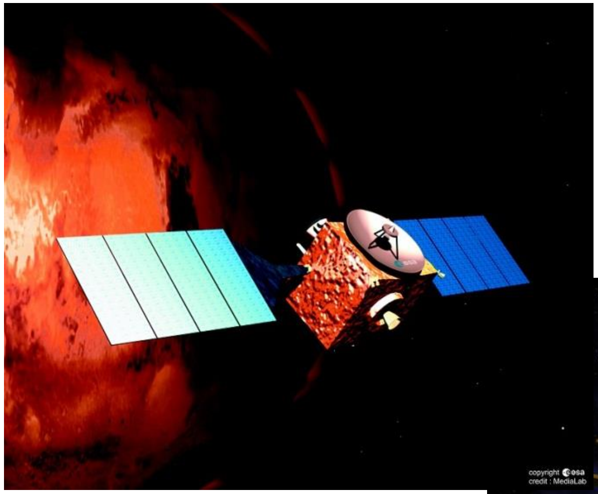
8. Interplanetary Trajectories

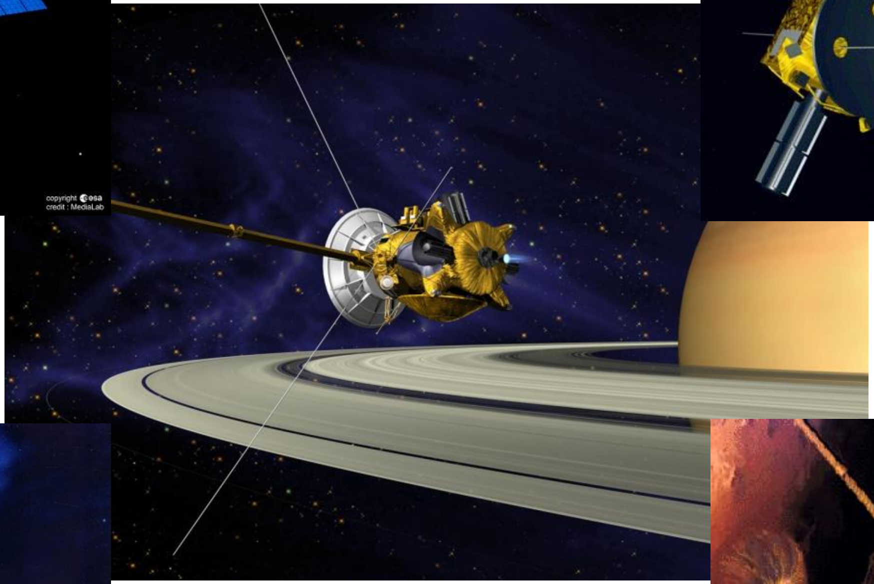


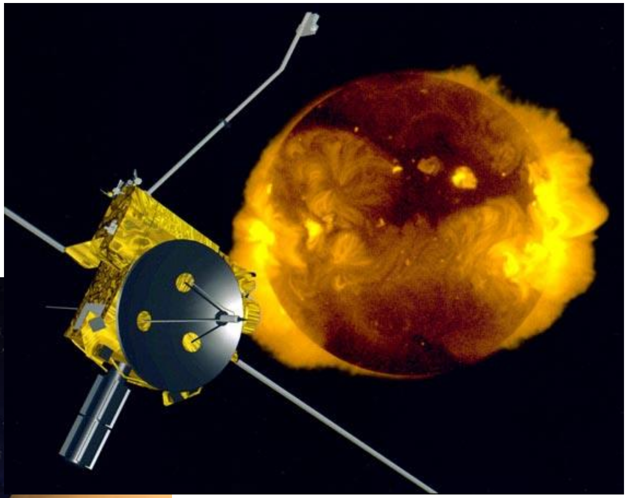
Gaëtan Kerschen

Space Structures & Systems Lab (S3L)

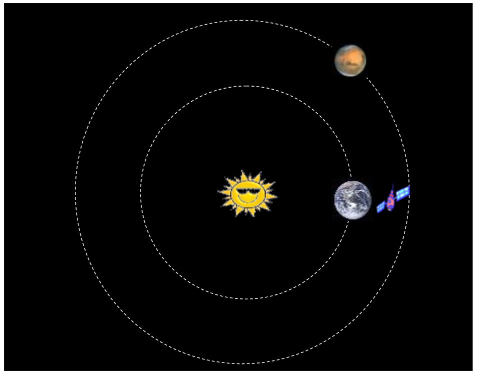
Motivation



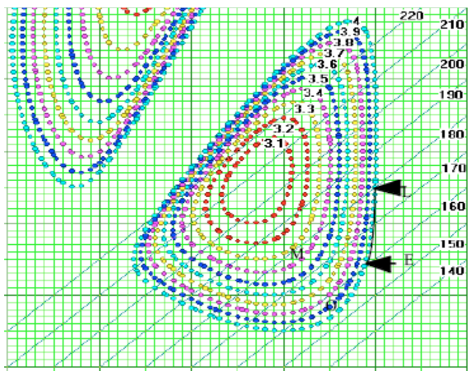




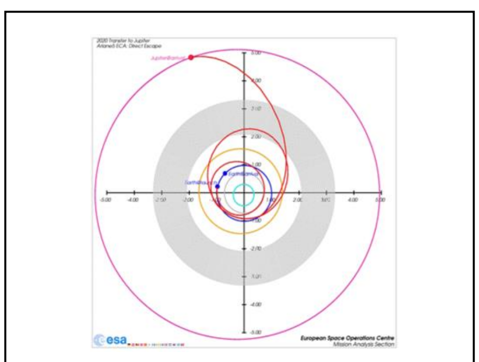
8. Interplanetary Trajectories



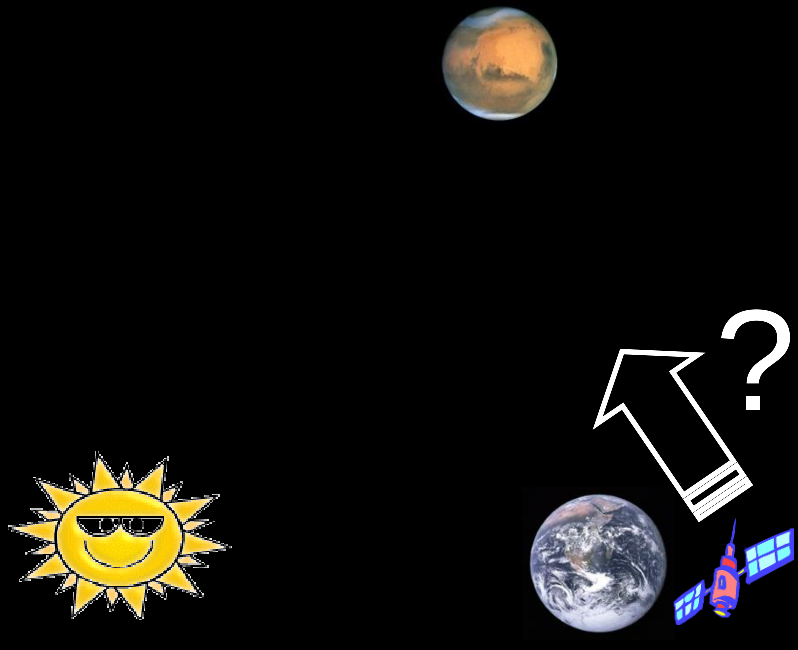
Patched conic method



Lambert's problem



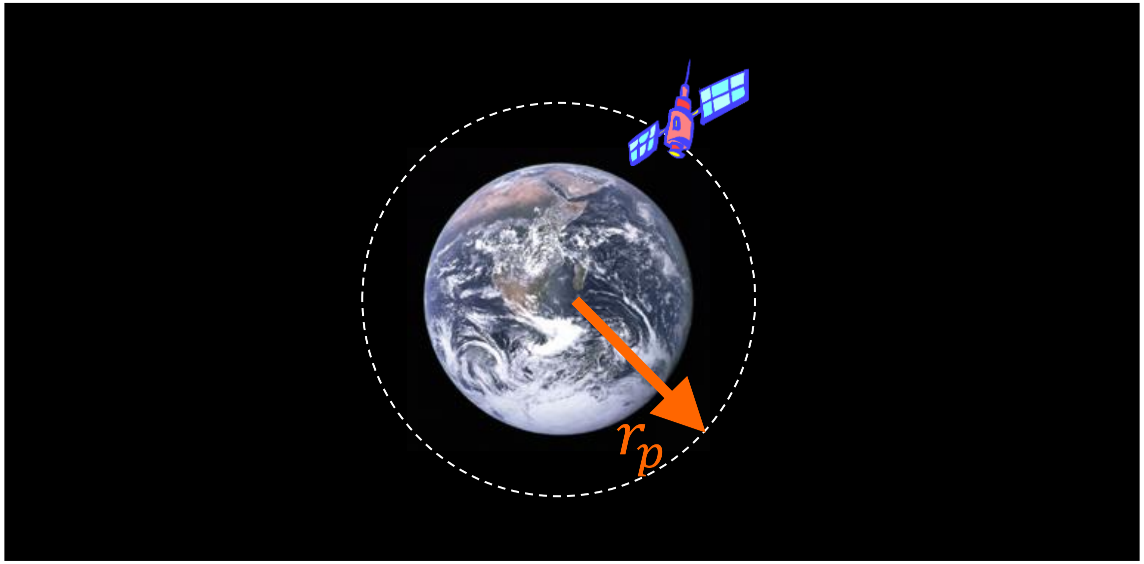
Gravity assist



What is the first step ?

The spacecraft is in parking orbit (circular) around the Earth.

What should be achieved first ?



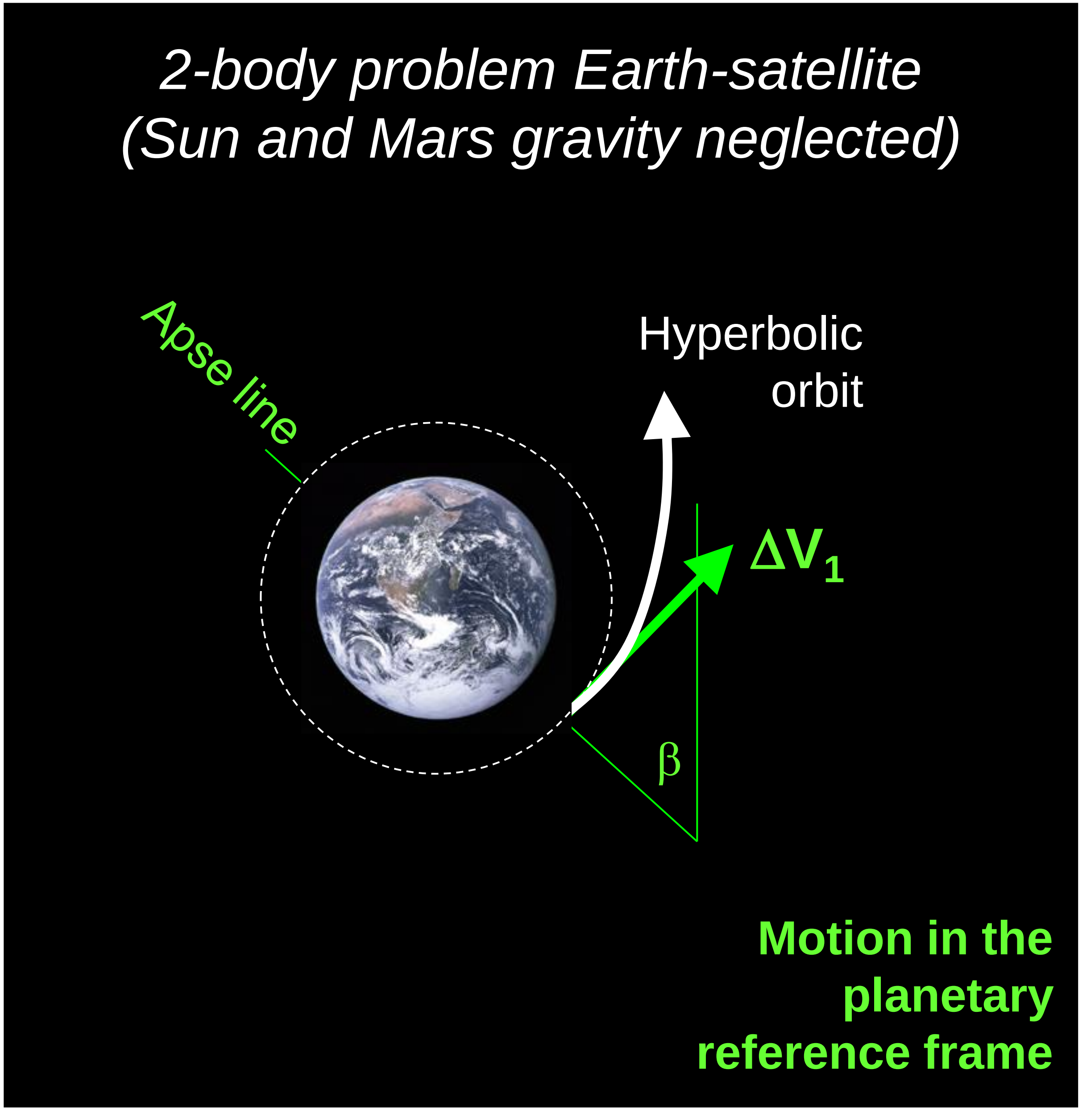
Reminder: 2-body problem

$$\varepsilon = \frac{\mu}{2a} = \frac{v^2}{2} - \frac{\mu}{r} \quad \Rightarrow \quad v_\infty = \sqrt{\frac{\mu}{a}}$$

Hyperbolic excess speed

$$\frac{v_\infty^2}{2} = \frac{v^2}{2} - \frac{\mu}{r} \quad \Rightarrow \quad v^2 = v_\infty^2 + v_{esc}^2 = C_3 + v_{esc}^2$$

Step 1: hyperbolic trajectory (escape the Earth)



How far can we go with this 2-body problem?

Sphere of influence (SOI): a spacecraft is within the Earth's SOI if the gravitational force due to Earth is greater than the gravitational force due to the sun.

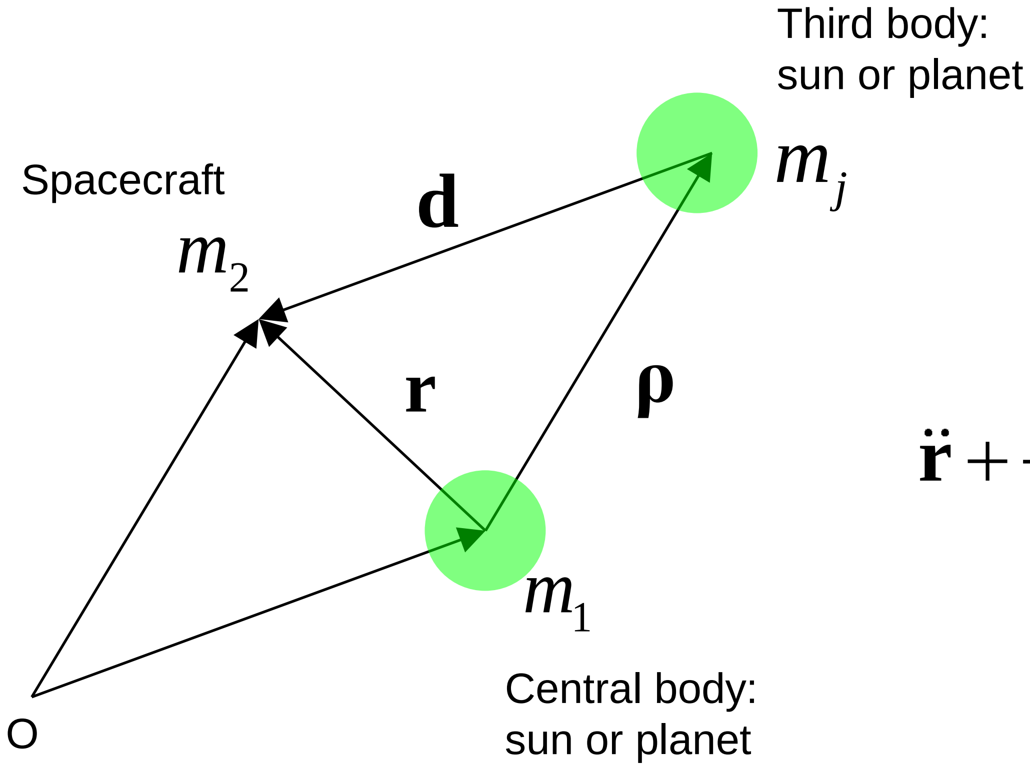
$$\frac{G m_E m_{sat}}{r_{E,sat}^2} > \frac{G m_S m_{sat}}{r_{S,sat}^2}$$

⇩

$$r_{E,sat} < 2.5 \times 10^5 \, \text{km}$$

What's wrong with this value ?

The 3-body problem



$$\ddot{\mathbf{r}} + \frac{\mu}{r^3}\mathbf{r} = -Gm_j\underbrace{\left(\frac{\mathbf{d}}{d^3} + \frac{\boldsymbol{\rho}}{\rho^3}\right)}_{\text{Disturbing function}}$$

The spacecraft orbits the planet or the Sun

$$\ddot{\mathbf{r}}_{pv} + \frac{G\left(m_p + m_v\right)}{r_{pv}^3}\mathbf{r}_{pv} = -Gm_s\left(\frac{\mathbf{r}_{sv}}{r_{sv}^3} + \frac{\mathbf{r}_{sp}}{r_{sp}^3}\right)$$

$$\ddot{\mathbf{r}}_{pv} - \mathbf{A}_p = \mathbf{P}_s$$

$\mathbf{P}_s$ → Perturbation acceleration due to the sun

$\mathbf{A}_p$ ↓ Primary gravitational acceleration due to the planet

p:planet
v: vehicle
s:sun

Orbit the planet

$$\ddot{\mathbf{r}}_{sv} + \frac{G\left(m_s + m_v\right)}{r_{sv}^3}\mathbf{r}_{sv} = -Gm_p\left(\frac{\mathbf{r}_{pv}}{r_{pv}^3} + \frac{\mathbf{r}_{sp}}{r_{sp}^3}\right)$$

$$\ddot{\mathbf{r}}_{sv} - \mathbf{A}_s = \mathbf{P}_p$$

$\mathbf{P}_p$ → Perturbation acceleration due to the planet

$\mathbf{A}_s$ ↓ Primary gravitational acceleration due to the sun

Orbit the sun

SOI: Correct Definition due to Laplace

It is the surface along which:

$$\frac{P_p}{A_s} = \frac{P_s}{A_p}$$

Measure of the planet's influence on the orbit of the vehicle relative to the sun ← $\frac{P_p}{A_s}$

$\frac{P_s}{A_p}$ → Measure of the deviation of the vehicle's orbit from the Keplerian orbit arising from the planet acting by itself

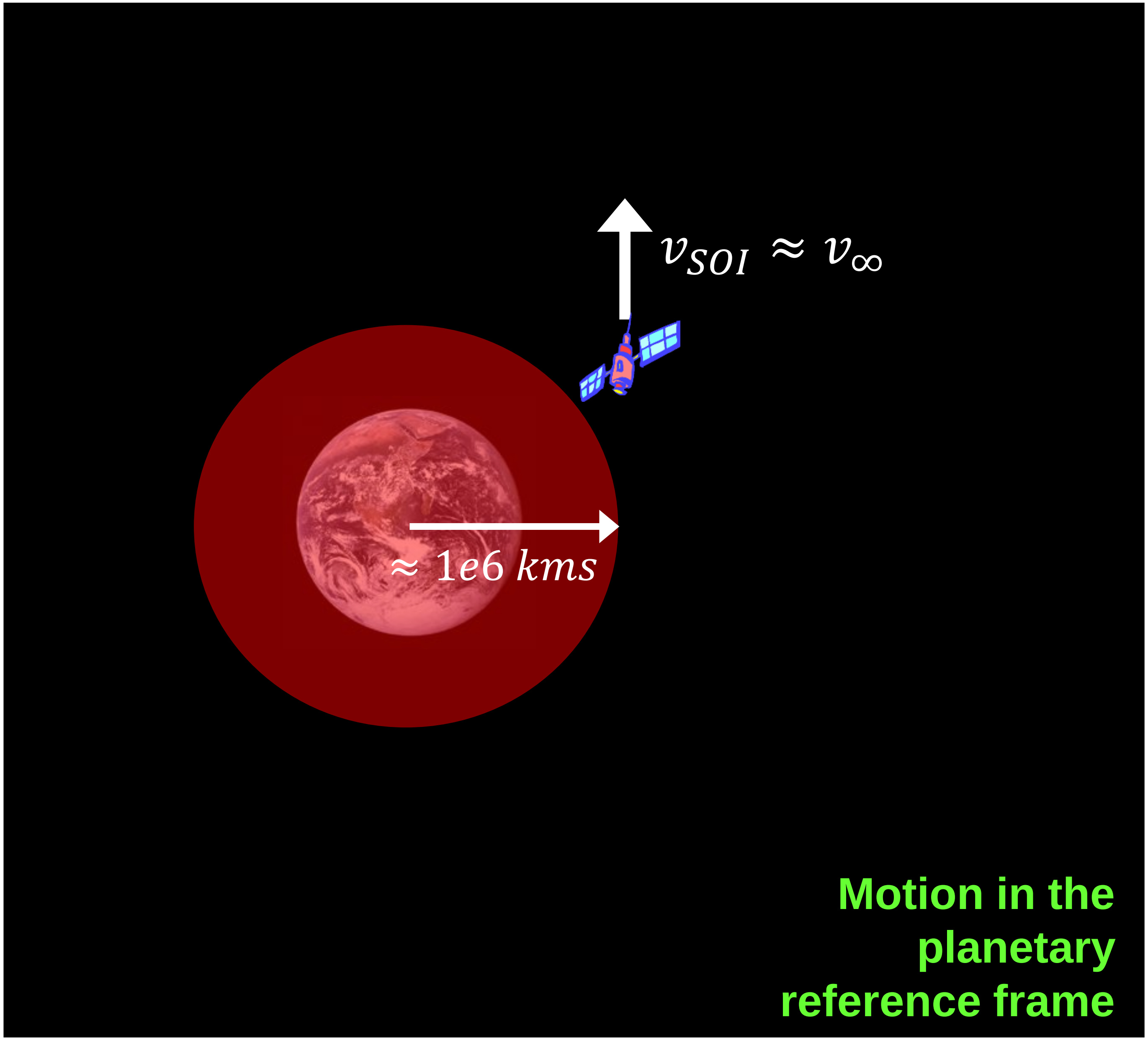
If $\frac{P_p}{A_s} > \frac{P_s}{A_p}$ the spacecraft is inside the SOI of the planet.

SOI Radii

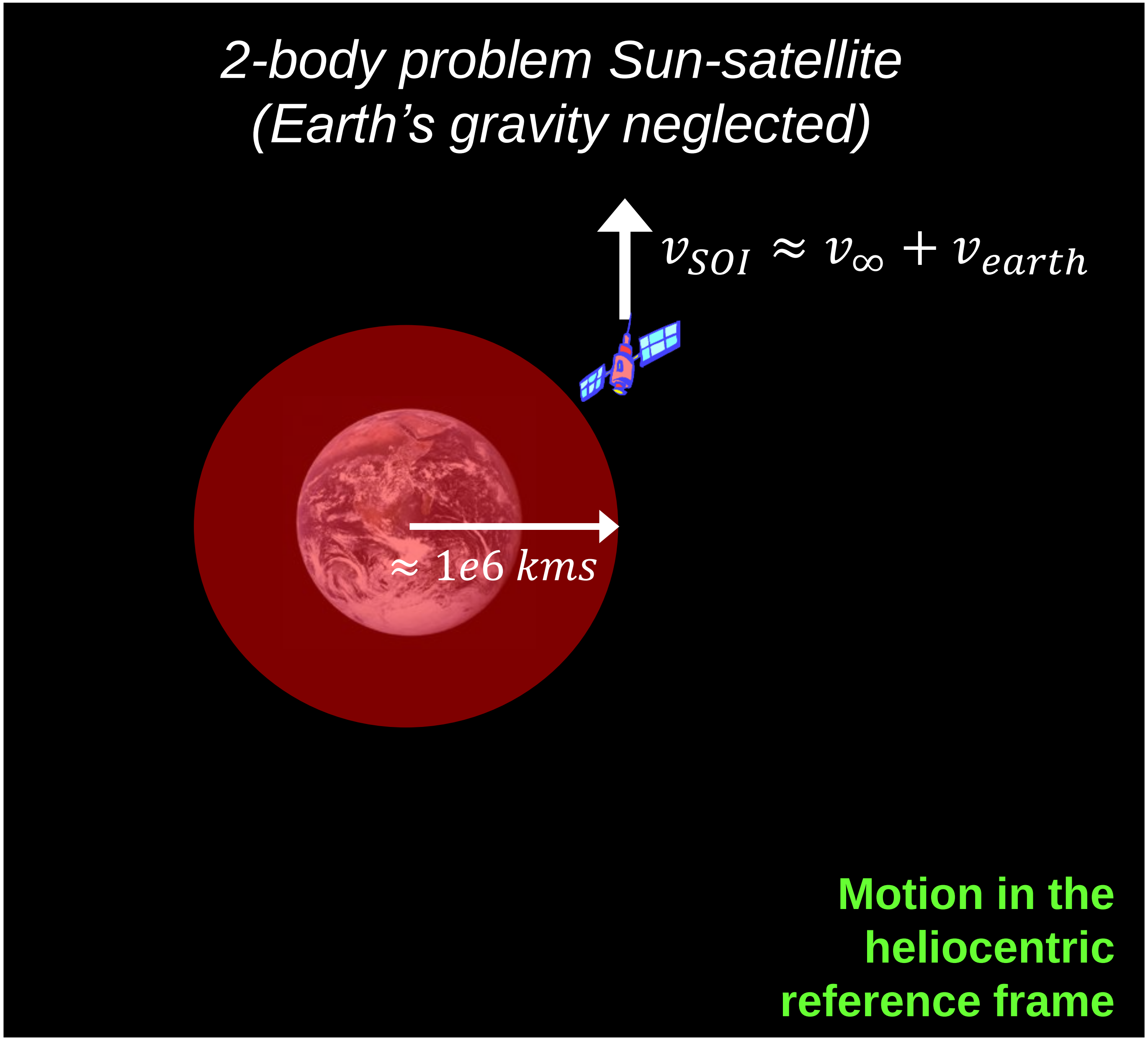
$$r_{SOI} \approx \left(\frac{m_p}{m_s}\right)^{\frac{2}{5}} r_{sp}$$

Planet	SOI Radius (km)	SOI radius (body radii)
Mercury	1.13×10^5	45
Venus	6.17×10^5	100
Earth	9.24×10^5 **OK !**	145
Mars	5.74×10^5	170
Jupiter	4.83×10^7	677
Neptune	8.67×10^7	3886

At the sphere of influence, far from the Earth

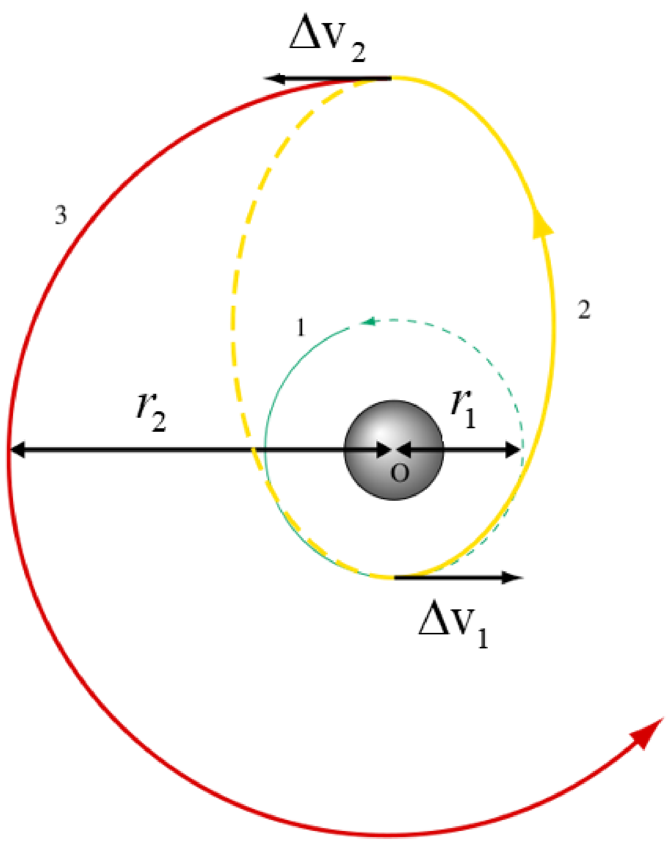


Planetary to heliocentric frame



Reminder: Hohmann Transfer

The minimum-fuel impulsive transfer orbit is the elliptic orbit that is tangent to both orbits at its apse line.

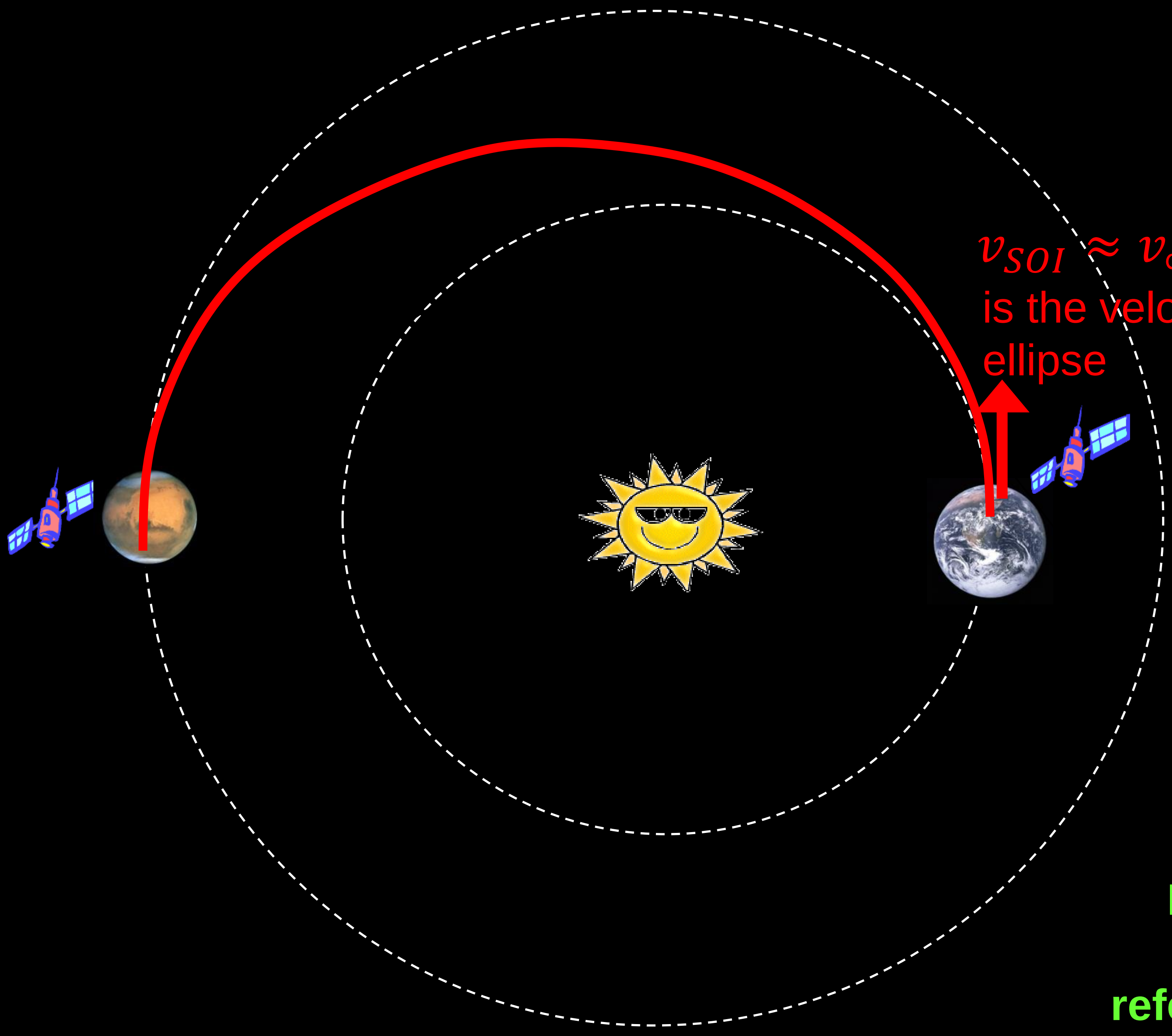


$$v_{circ} = \sqrt{\frac{\mu}{r}} \quad v_{ellip} = \sqrt{\mu\left(\frac{2}{r} - \frac{1}{a}\right)}$$

$$\Delta v_1 = \sqrt{\frac{2\mu r_2}{r_1(r_1 + r_2)}} - \sqrt{\frac{\mu}{r_1}}$$

$$\Delta v_2 = -\sqrt{\frac{2\mu r_1}{r_2(r_1 + r_2)}} + \sqrt{\frac{\mu}{r_2}}$$

Step 2: Hohmann transfer



Motion in the heliocentric reference frame

Hohmann transfer design

$$\Delta V = \sqrt{\frac{2\mu_{sun}R_{mars}}{R_{earth}(R_{earth}+R_{mars})}} - \sqrt{\frac{\mu_{sun}}{R_{earth}}}$$

Velocity of the satellite on the elliptical orbit around the Sun

$v_\infty + v_{earth}$

Velocity of the Earth around the Sun

v_{earth}

$$\sqrt{\frac{2\mu_{sun}R_{mars}}{R_{earth}(R_{earth}+R_{mars})}} - \sqrt{\frac{\mu_{sun}}{R_{earth}}} \approx v_\infty + v_{earth} - v_{earth} \approx v_\infty$$

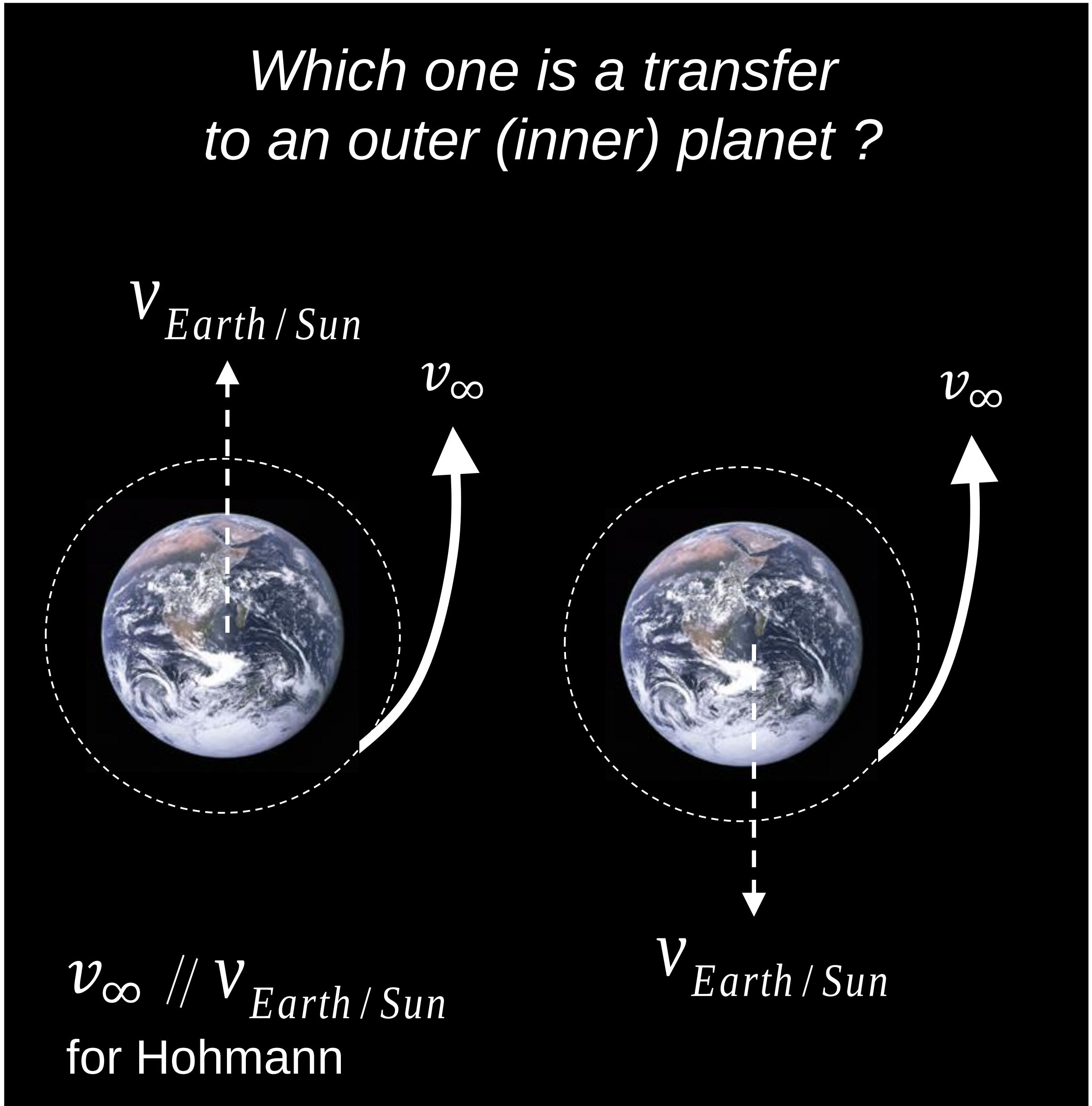
μ_{sun}=1.327e20

R_{earth}=149.6e9

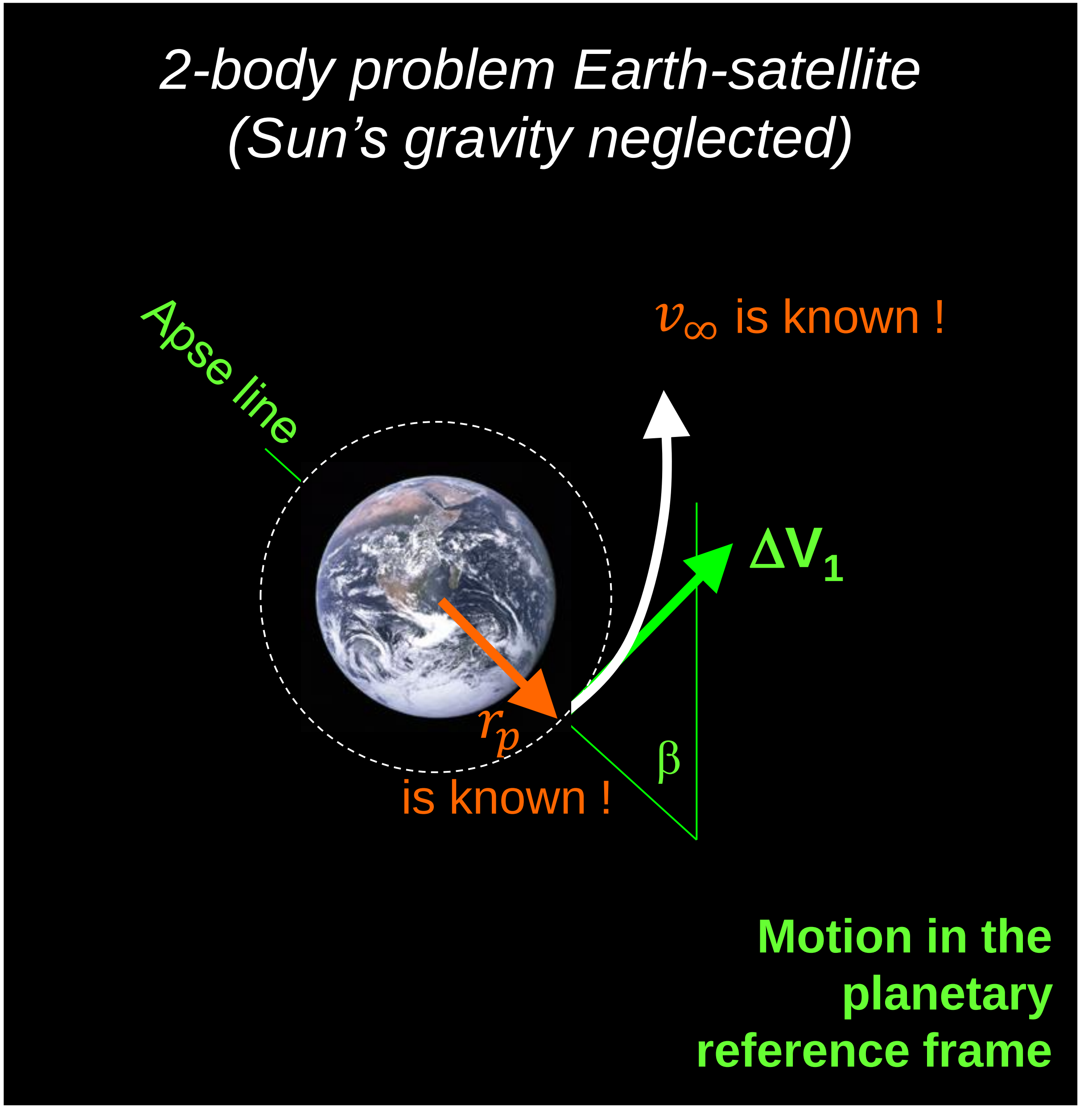
R_{mars}=228e9

We can calculate $v_\infty \approx 2.9 km/s$!

Let’s go back to our initial hyperbola



We can now design the initial hyperbola



We want to calculate ΔV1 and β

We know v_∞ and r_p

$$v_\infty = \sqrt{\frac{\mu}{a}}$$

$$r_p = \frac{h^2}{\mu(1+e)}$$

Orbit equation with θ=0

We have 3 unknowns, 2 equations, but

$$a = \frac{h^2}{\mu}\frac{1}{e^2-1}$$

https://en.wikipedia.org/wiki/Hyperbolic_trajectory

⇨

$$e = 1 + \frac{r_p v_\infty^2}{\mu}$$

$$h = r_p\sqrt{v_\infty^2 + \frac{2\mu}{r_p}}$$

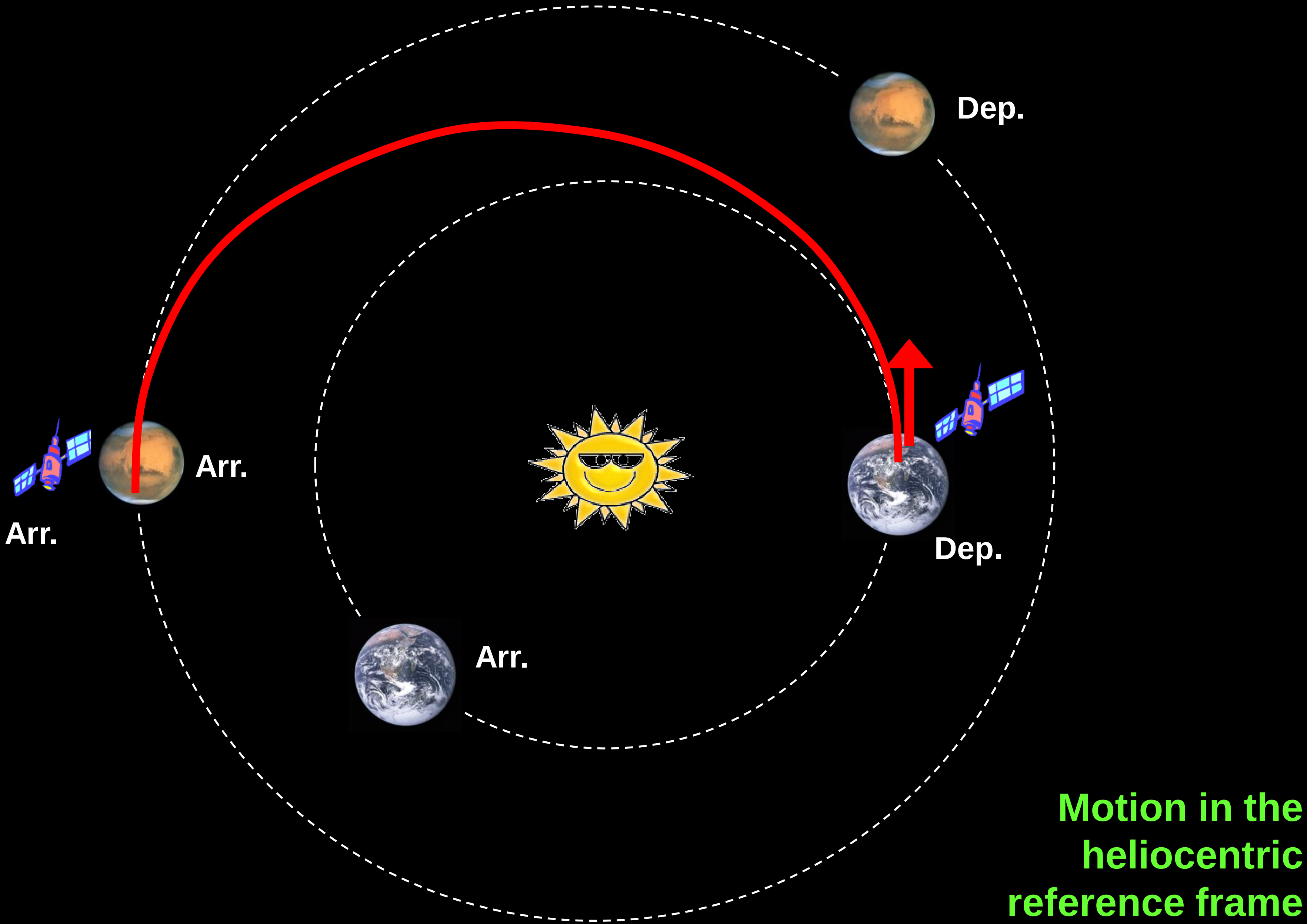
⇨

Hyper. Circ.

$$\Delta v = v_p - v_c = \frac{h}{r_p} - \sqrt{\frac{\mu}{r_p}}$$

$$\beta = \cos^{-1}\frac{1}{e}$$

Existence of launch windows: Mars should arrive at the apogee of the transfer ellipse at the same time the spacecraft does.

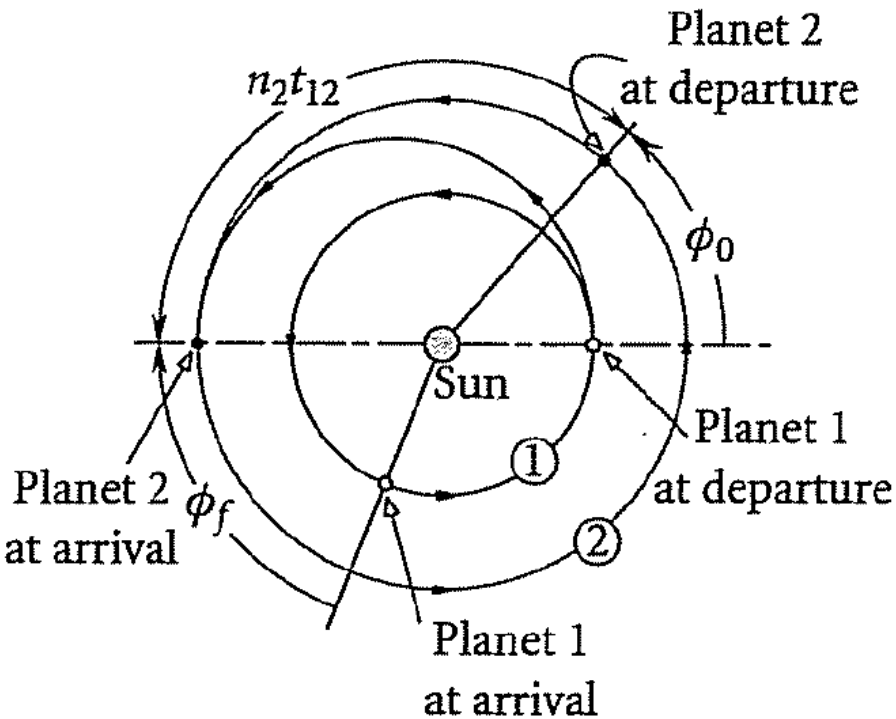


Existence of Launch Windows

Phasing maneuvers are not practical due to the large periods of the heliocentric orbits.

The planet should arrive at the apse line of the transfer ellipse at the same time the spacecraft does.

Transfer time



$$t_{12} = \frac{\pi}{\sqrt{\mu_{sun}}}\left(\frac{R_1 + R_2}{2}\right)^{3/2}$$

Hohmann

True anomaly

Mean motion

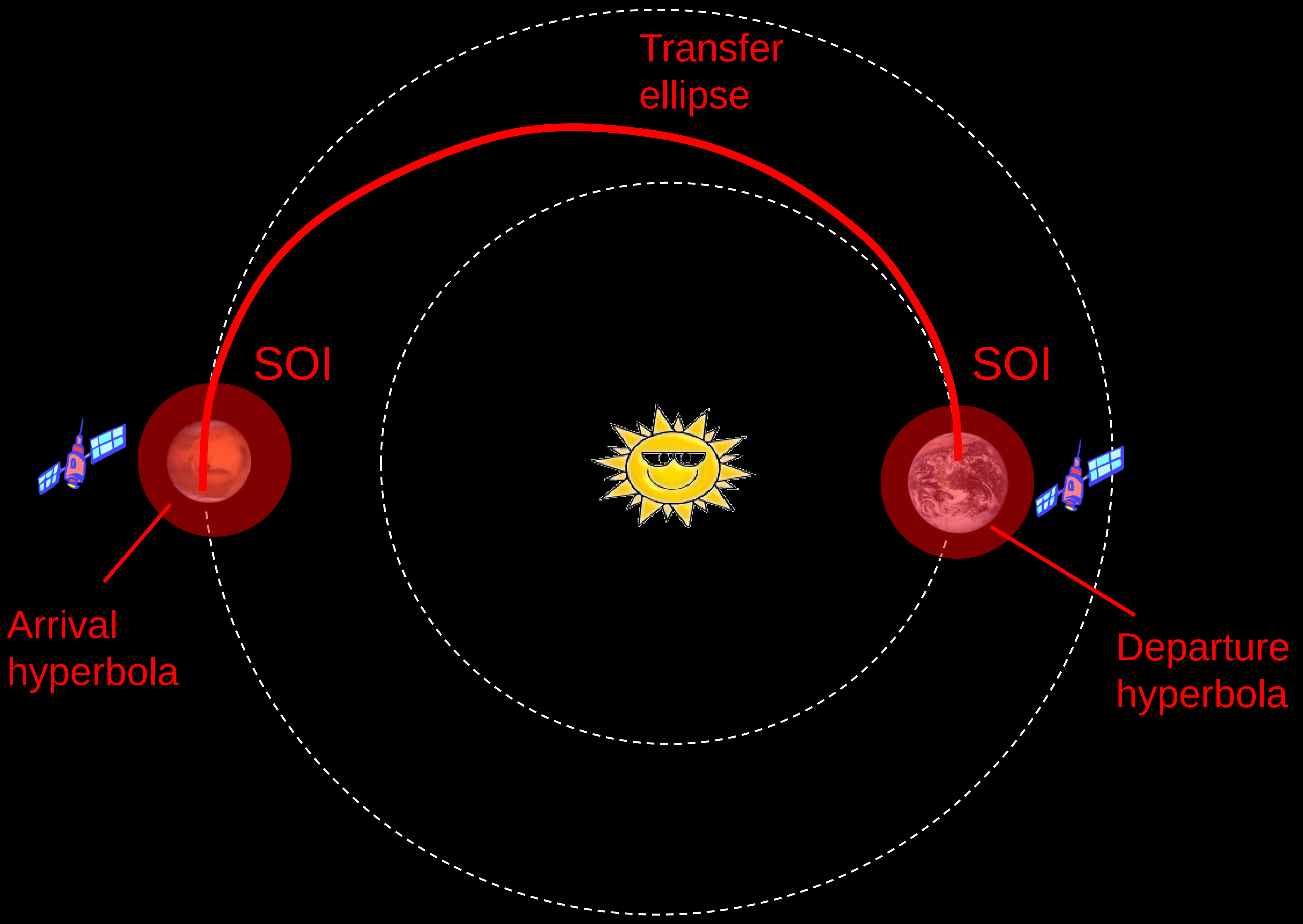
$$\theta_{Mars,2} = \theta_{Mars,1} + n_2 t_{12}$$

$$\theta_{Mars,1} = \emptyset_0 = \pi - n_2 t_{12}$$

$$t_{12} = 2.2362\times10^{7}\, s = 258.8 \text{ days}$$

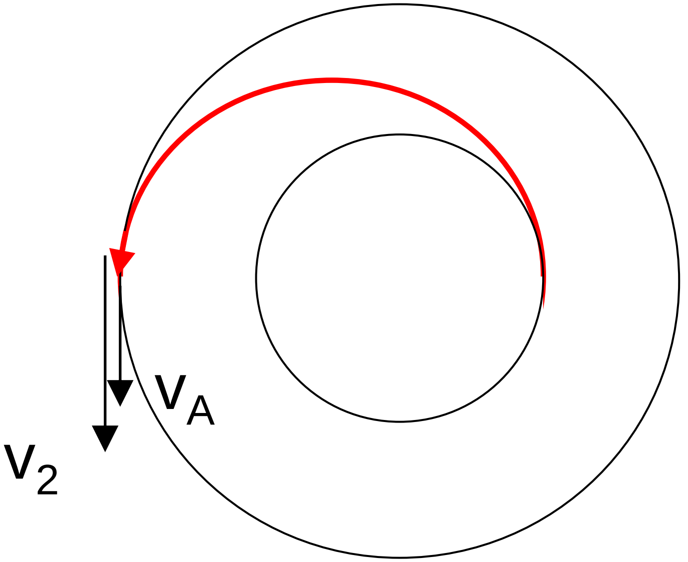
$$\phi_0 = 44^\circ$$

Step 3: Planetary arrival → similar reasoning



Governing Equations

$$v_2 - v_A = \sqrt{\frac{\mu_{sun}}{R_2}}\left(1 - \sqrt{\frac{2R_1}{(R_1 + R_2)}}\right)$$



Patched conic method: in summary

Sequence of 2-body problems: outbound hyperbola (departure), Hohmann transfer ellipse (interplanetary travel) and inbound hyperbola (arrival) with one body always being the spacecraft.

Approximate method: if the spacecraft is close enough to one celestial body, the gravitational forces due to other planets are neglected.

Very useful for preliminary mission design (delta-v requirements and flight times). But actual mission design employs the accurate numerical integration techniques.

Assumption of Circular, Coplanar Orbits

Planet	Inclination of the orbit to the ecliptic plane	Eccentricity	
Mercury	7.00º	0.206	KO !
Venus	3.39º	0.007	
Earth	0.00º	0.017	
Mars	1.85º	0.094	
Jupiter	1.30º	0.049	
Saturn	2.48º	0.056	
Uranus	0.77º	0.046	
Neptune	1.77º	0.011	
Pluto	17.16º	0.244	KO !

Earth-Jupiter Example: Hohmann

Galileo’s original mission was designed to use a direct Hohmann transfer, but following the loss of Challenger Galileo's intended Centaur booster rocket was no longer allowed to fly on Shuttles. Using a less-powerful solid booster rocket instead, Galileo used gravity assists instead.

Earth-Jupiter Example: Hohmann

Velocity when leaving Earth's SOI:

$$v_D - v_1 = v_\infty^E = \sqrt{\frac{\mu_{sun}}{R_1}}\left(\sqrt{\frac{2R_2}{(R_1+R_2)}}-1\right) = 8.792\text{km/s}$$

Velocity relative to Jupiter at Jupiter's SOI:

$$v_2 - v_A = v_\infty^J = \sqrt{\frac{\mu_{sun}}{R_2}}\left(1-\sqrt{\frac{2R_1}{(R_1+R_2)}}\right) = 5.643\text{km/s}$$

Transfer time: 2.732 years

Earth-Jupiter Example: Departure

Velocity on a circular parking orbit (300km):

$$v_c = \sqrt{\frac{\mu_E}{R_E + h}} = 7.726\text{km/s}$$

$$\Delta v = \sqrt{v_\infty^2 + \frac{2\mu}{r_p}} - 7.726\,\text{km/s} = 6.298\,\text{km/s}$$

$$e = 1 + \frac{r_p v_\infty^2}{\mu} = 2.295$$

Earth-Jupiter Example: Arrival

Final orbit is circular with radius=6R$_J$

$$\Delta v = \sqrt{v_\infty^2 + \frac{2\mu}{r_p}} - \sqrt{\frac{\mu(1+e)}{r_p}} = 24.95 - 17.18 = 7.77\text{km/s}$$

e=1.108

Rendez-vous opportunities: synodic period

$$\theta_1 = \theta_{10} + n_1 t$$

$$\theta_2 = \theta_{20} + n_2 t$$

$$\phi = \theta_2 - \theta_1$$

$$\Rightarrow \quad \phi = \phi_0 + \left(n_2 - n_1\right) t$$

$$\phi_0 - 2\pi = \phi_0 + \left(n_2 - n_1\right) T_{syn} \quad \Rightarrow \quad T_{syn} = \frac{2\pi}{\left|n_1 - n_2\right|}$$

$$\Downarrow$$

$$T_{syn} = \frac{T_1 T_2}{\left|T_1 - T_2\right|}$$

$$T_{syn} = \frac{365.26 \times 687.99}{\left|365.26 - 687.99\right|} = 777.9 \text{ days}$$

Earth-Mars mission

The total time for a manned Mars mission is

$$258.8 + 453.8 + 258.8 = 971.4 \text{ days} = 2.66 \text{ years}$$

1. In 258 days, Mars travels 258/688*360=135 degrees. Mars should be ahead of 45 degrees.

2. In 258 days, the Earth travels 258/365*360=255 degrees. At Mars arrival, the Earth is 75 degrees ahead of Mars.

3. At Mars departure, the Earth should be behind Mars of 75 degrees.

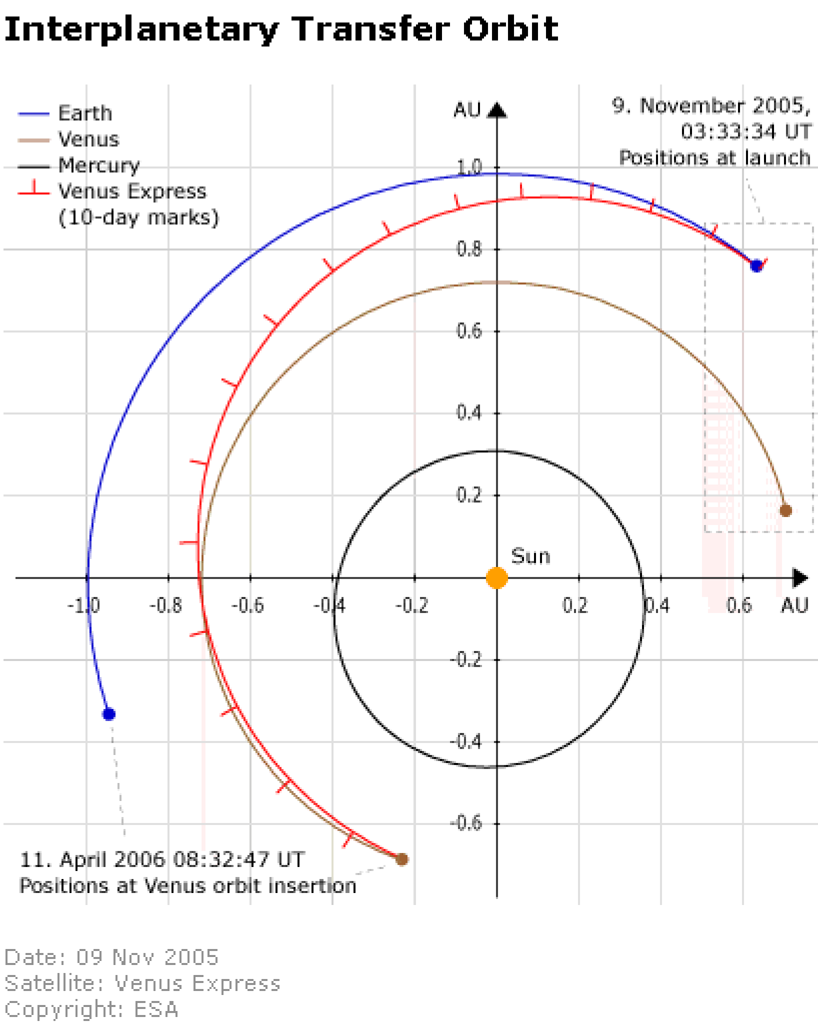
4. A return is possible if the Earth wins 360-75-75=210 degrees w.r.t. Mars. The Earth wins 360/365-360/688=0.463 degrees per day. So one has to wait 210/0.46=453 days.

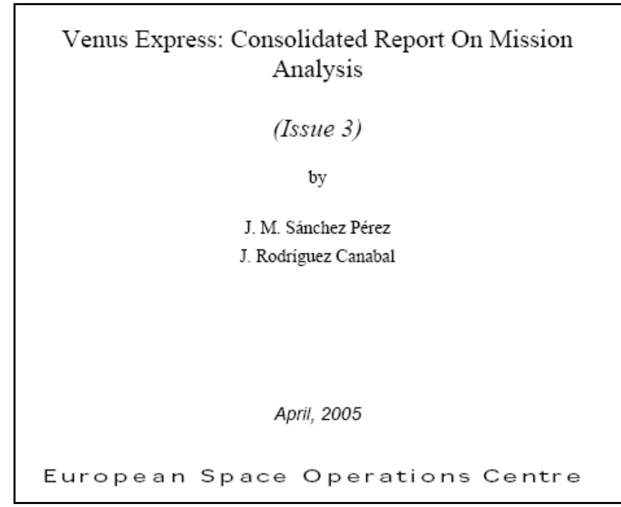
Hohmann Transfer: Other Planets

Planet	v_∞ departure (km/s)	Transfer time (days)
Mercury	7.5	105
Venus	2.5	146
Mars	2.9	259
Jupiter	8.8	998
Saturn	10.3	2222
Pluto	11.8	16482

Assumption of circular, co-planar orbits and tangential burns

Venus Express: A Hohmann-Like Transfer




Venus Express: Consolidated Report On Mission Analysis

(Issue 3)

by

J. M. Sánchez Pérez

J. Rodríguez Canabal

April, 2005

European Space Operations Centre

$C_3 = 7.8\ km^2/s^2$

Time: 154 days

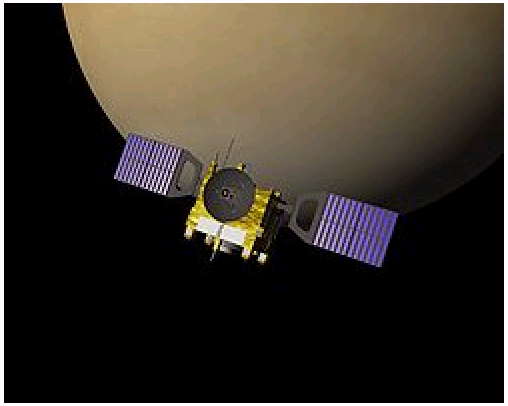
Real data

Why ?

$C_3 = 6.25\ km^2/s^2$

Time: 146 days

Hohmann



Organization	ESA
Major contractors	EADS Astrium, Toulouse, France, leading a team of 25 subcontractors from 14 European countries.
Mission type	Orbiter
Satellite of	Venus
Launch date	9 November 2005 03:33:34 UTC
Launch vehicle	Soyuz-FG/Fregat
Mission duration	150 days enroute; 1,000 days in orbit 4 years and 5 months elapsed
COSPAR ID	2005-045A
Home page	www.esa.int/SPECIALS/Venus_Express
Mass	1,270 kg

SOYUZ

from the Guiana Space Centre

User's Manual

Issue 1 – Revision 0 – June 06

* - transition between sub-orbital and direct injection profile curves is simplified

Sensitivity Analysis: Departure

The maneuver occurs well within the SOI, which is just a point on the scale of the solar system.

One may therefore ask what effects small errors in position and velocity (r_p and v_p) at the maneuver point have on the trajectory (target radius R_2 of the heliocentric Hohmann transfer ellipse).

$$\frac{\delta R_2}{R_2} = \frac{2}{1 - \frac{R_1 v_D^2}{2\mu_{sun}}} \left(\frac{\mu_1}{v_D v_\infty r_p} \frac{\delta r_p}{r_p} + \frac{v_\infty + \frac{2\mu_1}{r_p}}{v_D} \frac{\delta v_p}{v_p} \right)$$

Sensitivity Analysis: Earth-Mars, 300km Orbit

$$\mu_{sun} = 1.327 \times 10^{11} \text{km}^3 / s^2, \mu_1 = 398600 \text{ km}^3 / s^2$$

$$R_1 = 149.6 \times 10^6 \text{km}, R_2 = 227.9 \times 10^6 \text{km}, r_p = 6678 \text{ km}$$

$$v_D = 32.73 \text{ km} / s, v_\infty = 2.943 \text{ km} / s$$

$$\Rightarrow \quad \frac{\delta R_2}{R_2} = 3.127 \frac{\delta r_p}{r_p} + 6.708 \frac{\delta v_p}{v_p}$$

A 0.01% variation in the burnout speed v_p changes the target radius by 0.067% or 153000 km.

A 0.01% variation in burnout radius r_p (670 m !) produces an error over 70000 km.

Sensitivity Analysis: Launch Errors

Standard GTO

a	semi-major axis (km)	40
e	eccentricity	$4.5\ 10^{-4}$
i	inclination (deg)	0.02
ω_p	argument of perigee (deg)	0.2
Ω	ascending node (deg)	0.2

Ariane V

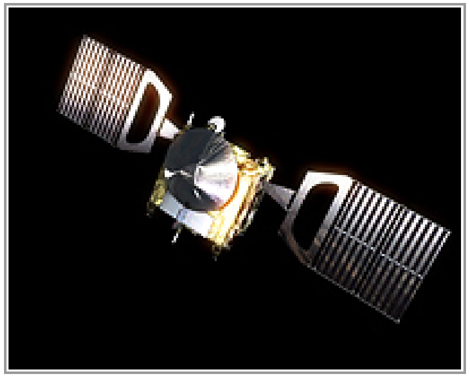
Trajectory correction maneuvers are clearly mandatory.

Sensitivity Analysis: Arrival

The heliocentric velocity of Mars in its orbit is roughly 24km/s.

If an orbit injection were planned to occur at a 500 km periapsis height, a spacecraft arriving even 10s late at Mars would likely enter the atmosphere.

News



Artist's impression of Venus Express spacecraft

Venus Express mission operations update

10 November 2005

At 11:30 CET, 10 November 2005, Venus Express Ground Segment Manager Manfred Warhaut reported from ESOC's Main Control Room that both the Venus Express spacecraft and ground segment continue to perform excellently.

The Venus Express Launch and Early Orbit (LEOP) operations continue to run very smoothly. However, the highlight of this period was the successful planning and testing of the Trajectory Correction Manoeuvre (TCM-0).

Given the slight over-performance of the Soyuz-Fregat launcher, it was decided to do the TCM-0 in direction of Earth in order to make best use of fuel. The movement (slew) of the spacecraft was enabled at 06:20 CET, started 06:43 and was completed 07:13.

Subsequently, the TCM-0 started at 07:38:52, had a manoeuvre duration of 48 seconds and a magnitude of 0.5 metres per second. Assessment of the manoeuvre afterwards based on Doppler data indicated that the manoeuvre duration was about 1 second less than commanded with negligible error in performance.

At 08:33 the spacecraft was turned back to the starting attitude. This completed the foreseen activities for this period.

The support from the ESA and NASA Deep Space Network ground stations has been very good throughout the LEOP.

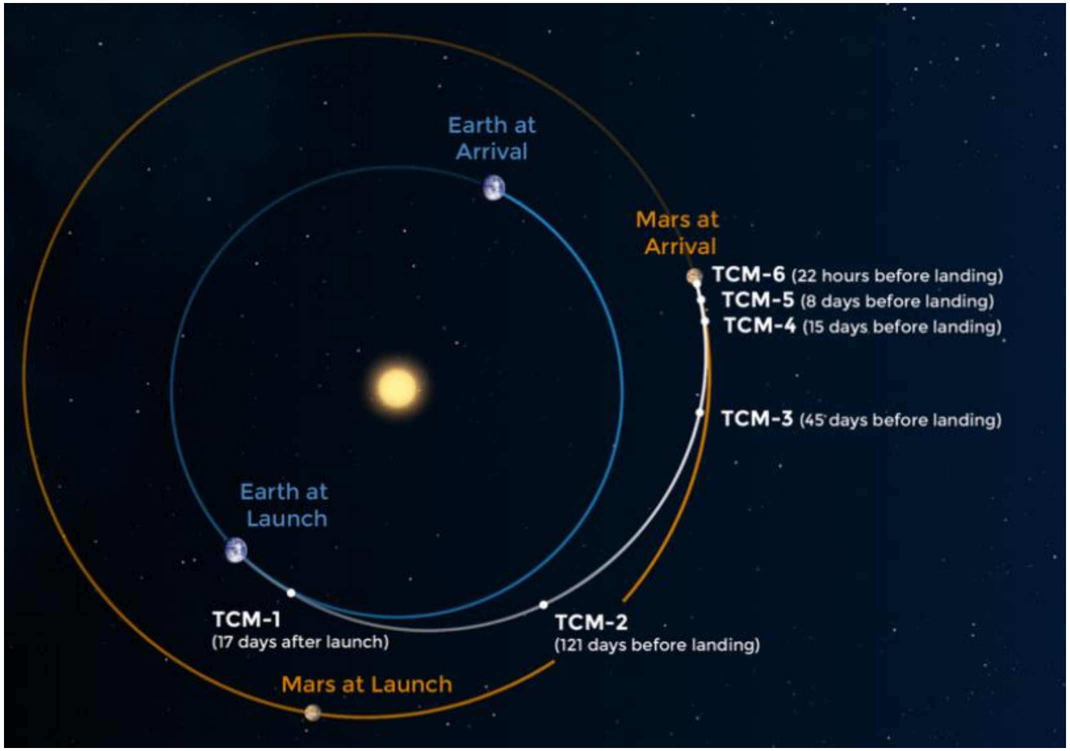
TCM/ OTM	Date	Event	Duration [s]	Delta v [m/s] Actual		P
				Bi	Mono	
(1)	(2)	(3)	(4)	(5)	(6)	
1	09.11.97	V1-Launch	34,13	2,70		
2	25.02.98	V1			0,18	
3	Canceled	V1				
4	Canceled	V2-CA				
5	03.12.98	V2-DSM	5.275,23	450,00		
6	04.02.99	V2	125,21	11,55		
7	18.05.99	V2			0,23	
8	Canceled	V2				
9	06.07.99	Earth	466,91	43,49		
10	19.07.99	Earth	54,63	5,13		
11	02.08.99	Earth	383,78	36,29		
12	11.08.99	Earth	128,46	12,25		
13	31.08.99	Earth-CA	69,90	6,69		
14	14.06.00	Flush	5,74	0,55		
15	Canceled	Jupiter				
16	Canceled	Jupiter				
17	28.02.01	Flush	5,32	0,51		
18	01.04.02	Flush	9,85	0,89		
19	01.05.03	Flush	17,53	1,58		
20	27.05.04	Phoebe	362,00	34,70		
21	17.06.04	Phoebe-CA	38,38	3,68		
22	Canceled	Pre SOI				
Cruise				609,99	0,40	

Cassini-Huygens

MARS InSight Mission

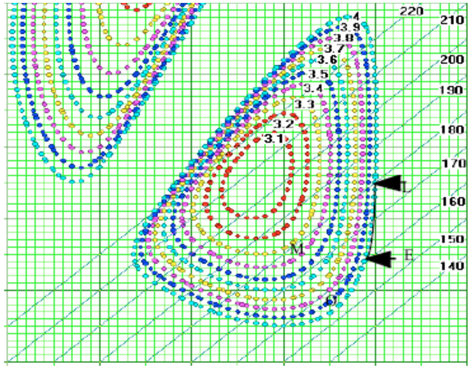
Contrairement à ce que l'on pourrait penser, la fusée utilisée pour InSight n'est pas pointée directement vers Mars, bien au contraire. Les règles de protection planétaire, qui stipulent que dans l'exploration martienne, tout doit être fait pour éviter de contaminer la planète rouge avec des germes terrestres, ont ici une conséquence étonnante. Les engins robotiques martiens sont effectivement lancés de manière à rater leur cible, ceci pour empêcher l'étage supérieur du lanceur, qui suit les sondes sur leur lancée, de s'écraser sur Mars.

InSight n'étant pas tiré précisément en direction de Mars, des manoeuvres de correction de trajectoire sont programmées tout au long de son voyage pour éliminer la dérive placée volontairement au départ, et ramener la sonde sur le droit chemin.

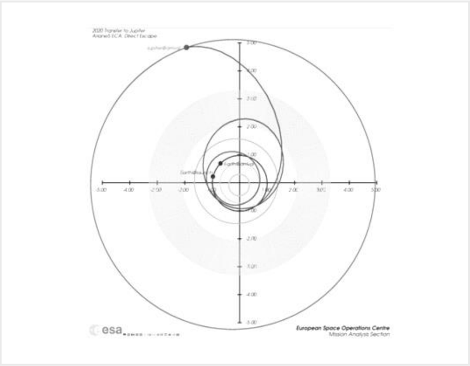


MARS InSight Mission

Date (subject to change)	Trajectory Correction Maneuvers	Activity
May 22, 2018 17 days after launch	TCM 1	To point InSight towards Mars and fine-tune its flight path after launch.
July 28, 2018 121 days before landing	TCM 2	To point InSight towards Mars.
Oct. 12, 2018 45 days before landing	TCM 3	To make sure InSight travels at the right speed and direction to arrive at correct location at the top of the Martian atmosphere before its planned landing.
Nov. 11, 2018 15 days before landing	TCM 4	
Nov. 18, 2018 8 days before landing	TCM 5	
Nov. 25, 2018 22 hours before landing	TCM 6	



6.2 Lambert's problem



Nontangential Burns

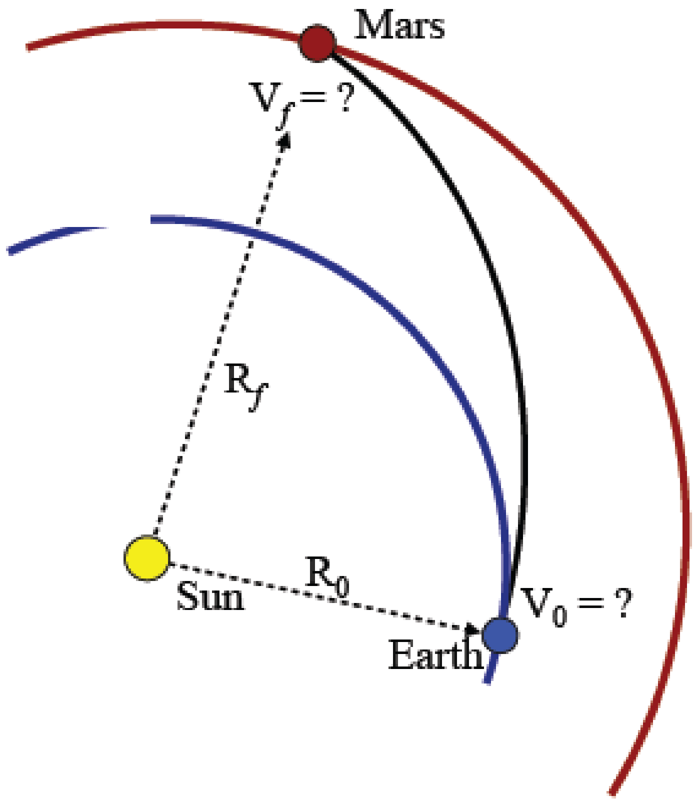
Section 6.1 discussed Hohmann interplanetary transfers, which are optimal with respect to fuel consumption.

Why should we consider nontangential burns (i.e., non-Hohmann transfer) ?

	Initial Alt (km)	Final Alt (km)	ν_{trans_b}	Bi-elliptic Transfer Alt (km)	Δv (km/s)	τ_{trans} (h)
			Transfer to Geosynchronous			
Hohmann	191.344 11	35,781.35			3.935	5.256
One-tangent	191.344 11	35,781.35	160°		4.699	3.457
Bi-elliptic	191.344 11	35,781.35		47,836.00	4.076	21.944
			Transfer to the Moon			
Hohmann	191.344 11	376,310.00			3.966	118.683
One-tangent	191.344 11	376,310.00	175°		4.099	83.061
Bi-elliptic	191.344 11	376,310.00		503,873.00	3.904	593.919

L05

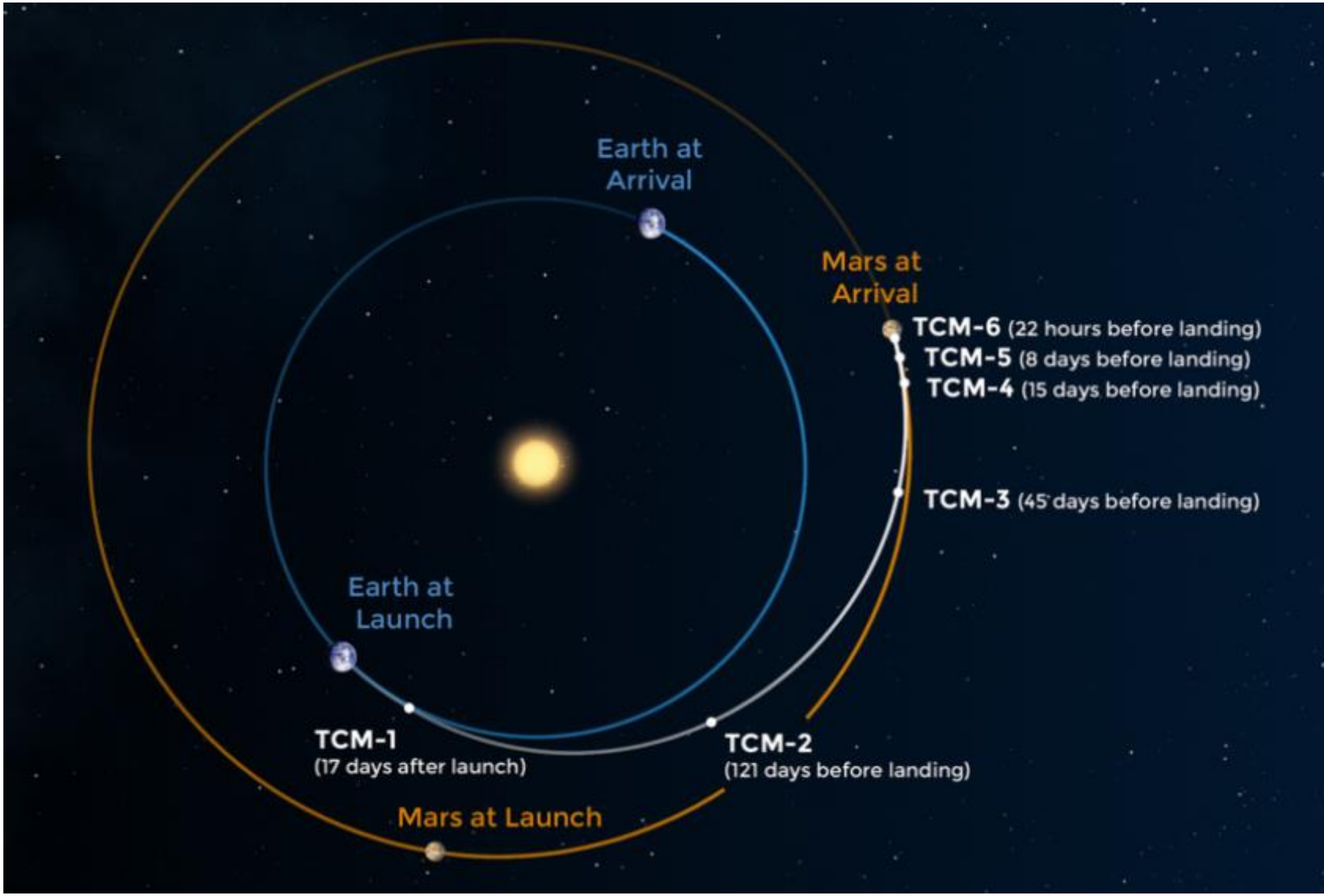
Non-Hohmann Trajectories



Solution using Lambert's theorem (Lecture 05):

If two position vectors and the time of flight are known, then the orbit can be fully determined.

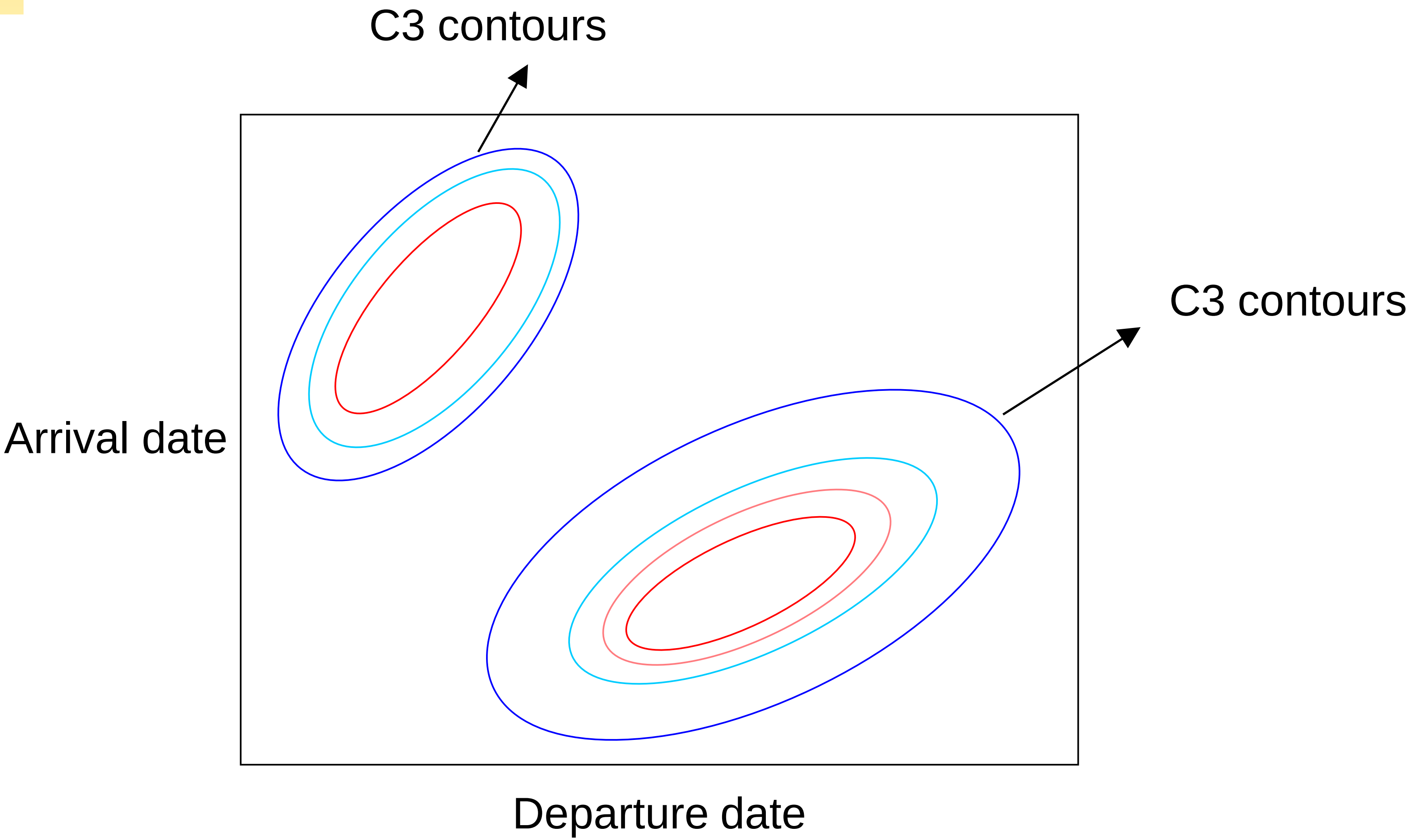
NASA Insight: 205 days vs. 258 days



Venus Express Example

Earth Departure				Venus Arrival					FP
Date	Lift Off	V_∞ Km/s	δ_∞ deg	Date	Hour	V_∞ Km/s	ξ Km	η Km	
26.10.05	04:43:38.7	2.7855	-25.614	06.04.06	21:16:27	4.6215	8815.3	12826.5	1
27.10.05	04:37:42.4	2.7855	-25.614	07.04.06	02:12:56	4.6192	8824.2	12828.4	
28.10.05	04:31:46.4	2.7855	-25.614	07.04.06	07:02:54	4.6171	8832.2	12829.9	
29.10.05	04:25:36.0	2.7855	-25.613	07.04.06	11:46:26	4.6153	8839.3	12830.9	
30.10.05	04:19:25.9	2.7855	-25.613	07.04.06	16:24:56	4.6139	8845.4	12831.5	
31.10.05	04:13:10.7	2.7855	-25.613	07.04.06	20:57:02	4.6128	8850.7	12831.5	
01.11.05	04:06:50.1	2.7855	-25.613	08.04.06	01:22:50	4.6121	8854.9	12830.9	
02.11.05	04:00:23.6	2.7855	-25.613	08.04.06	05:41:07	4.6119	8858.2	12829.6	
03.11.05	03:53:50.4	2.7855	-25.612	08.04.06	09:52:23	4.6120	8860.5	12827.5	
04.11.05	03:47:09.4	2.7855	-25.612	08.04.06	13:53:36	4.6127	8861.9	12824.2	
05.11.05	04:03:21.1	2.7904	-21.052	10.04.06	17:27:06	4.6059	8769.9	12910.2	2
06.11.05	03:57:04.2	2.7904	-21.051	10.04.06	18:29:06	4.6036	8767.5	12919.5	
07.11.05	03:44:32.1	2.7904	-21.051	10.04.06	12:10:22	4.6033	8790.0	12905.4	
08.11.05	03:39:30.3	2.7904	-21.051	11.04.06	04:26:18	4.5999	8733.8	12955.0	
09.11.05	03:33:34.5	2.7904	-21.050	11.04.06	08:16:25	4.5990	8715.3	12970.4	
10.11.05	03:26:40.7	2.7904	-21.050	11.04.06	11:26:04	4.5986	8697.4	12983.7	
11.11.05	03:19:19.0	2.7904	-21.050	11.04.06	14:27:44	4.5987	8677.8	12996.4	
12.11.05	03:19:32.8	2.8560	-19.502	12.04.06	09:12:37	4.5983	8582.6	13061.0	3
13.11.05	03:12:43.3	2.8560	-19.502	12.04.06	11:55:55	4.5984	8552.8	13080.1	
14.11.05	03:04:40.2	2.8560	-19.502	12.04.06	14:12:26	4.5990	8523.5	13097.1	

Porkchop Plot: Visual Design Tool



In porkchop plots, orbits are considered to be non-coplanar and elliptic.

NASA/TM—1998–208533

Interplanetary Mission Design Handbook:

Earth-to-Mars Mission Opportunities and Mars-to-Earth Return Opportunities 2009–2024

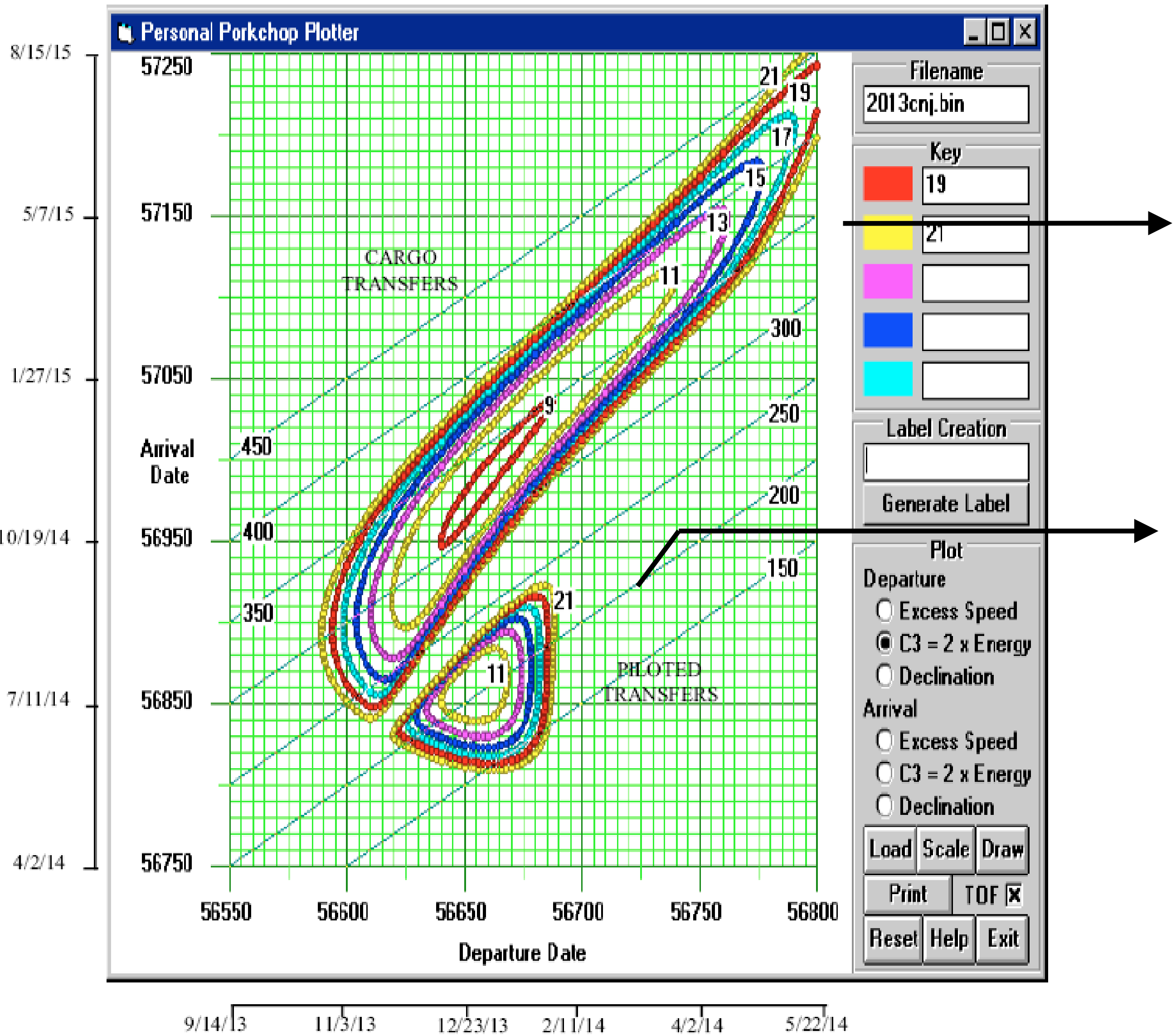
L.E. George
U.S. Air Force Academy, Colorado Springs, Colorado

L.D. Kos
Marshall Space Flight Center, Marshall Space Flight Center, Alabama

HUMAN MARS DESIGN REFERENCE MISSION OVERVIEW

The design reference mission (DRM) is currently envisioned to consist of three trans-Mars injection (TMI)/flights: two cargo missions in 2011, followed by a piloted mission in 2014. The cargo missions will be on slow (near Hohmann-transfer) trajectories with an in-flight time of 193–383 days. The crew will be on higher energy, faster trajectories lasting no longer than 180 days each way in order to limit the crew's exposure to radiation and other hazards. Their time spent on the surface of Mars will be approximately 535–651 days (figure 1). A summary of the primary cargo and piloted trajectories is summarized in table 1.

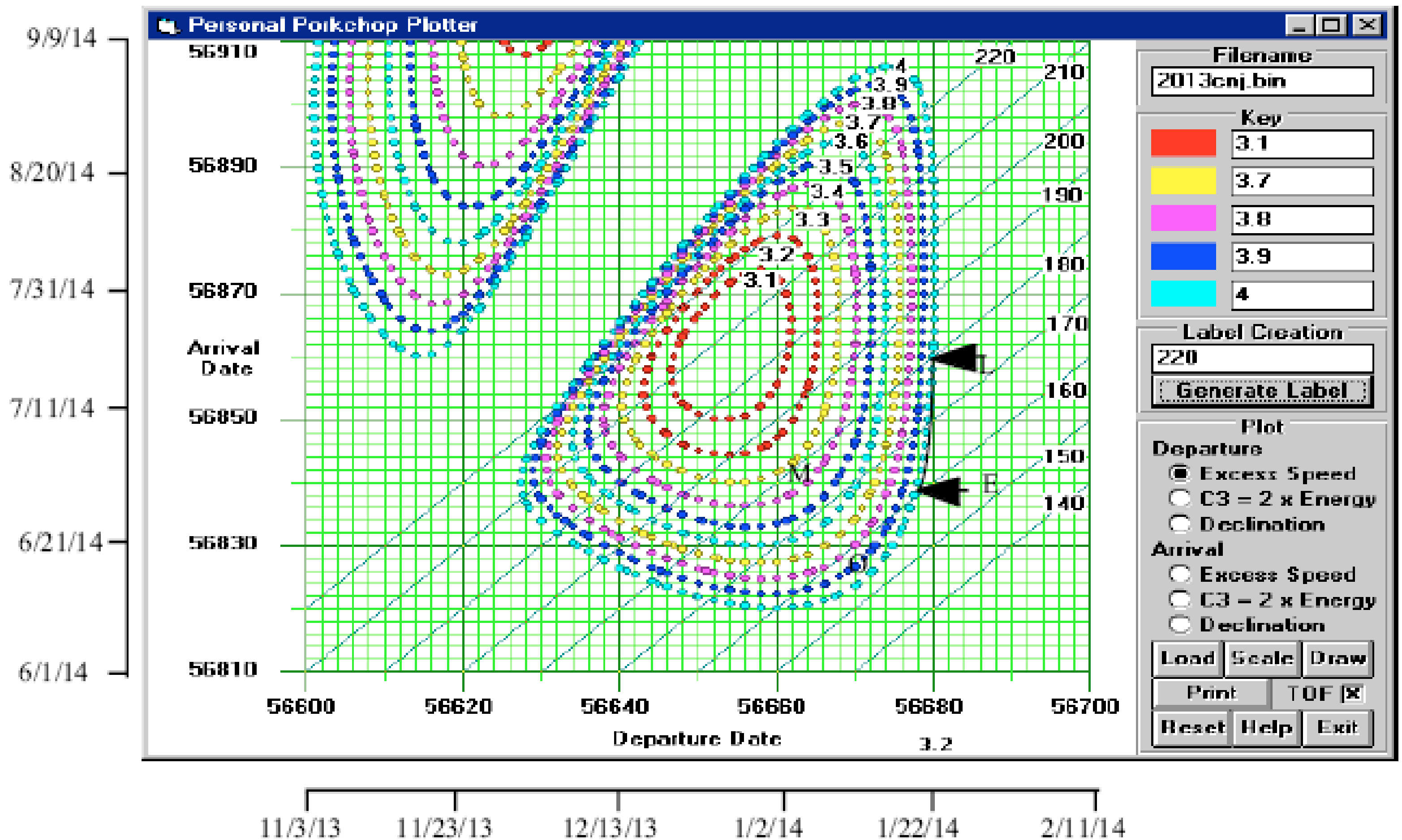
Earth-Mars Trajectories
2013/14 Conjunction Class
C_3 (Departure Energy) km^2/sec^2



Type II transfer for cargo: the spacecraft travels more than a 180° true anomaly

Type I transfer for piloted: the spacecraft travels less than a 180° true anomaly

Earth-Mars Trajectories
2013/14 Piloted Missions



E=Minimum flight time trajectory using 2011 Piloted Mission Departure Excess Speed (3.99 km/sec) and while maintaining acceptable Mars entry velocity needed for aerobraking.

Departure: 1/20/14 (56678J) Arrival: 6/30/14 (56839J)

L=Latest possible trajectory to keep flight time limited to 180 days. The acceptable window of opportunity for launch will be along the arc from E to L.

Latest Departure: 1/22/14 (56679J) Arrival: 7/21/14 (56859J)

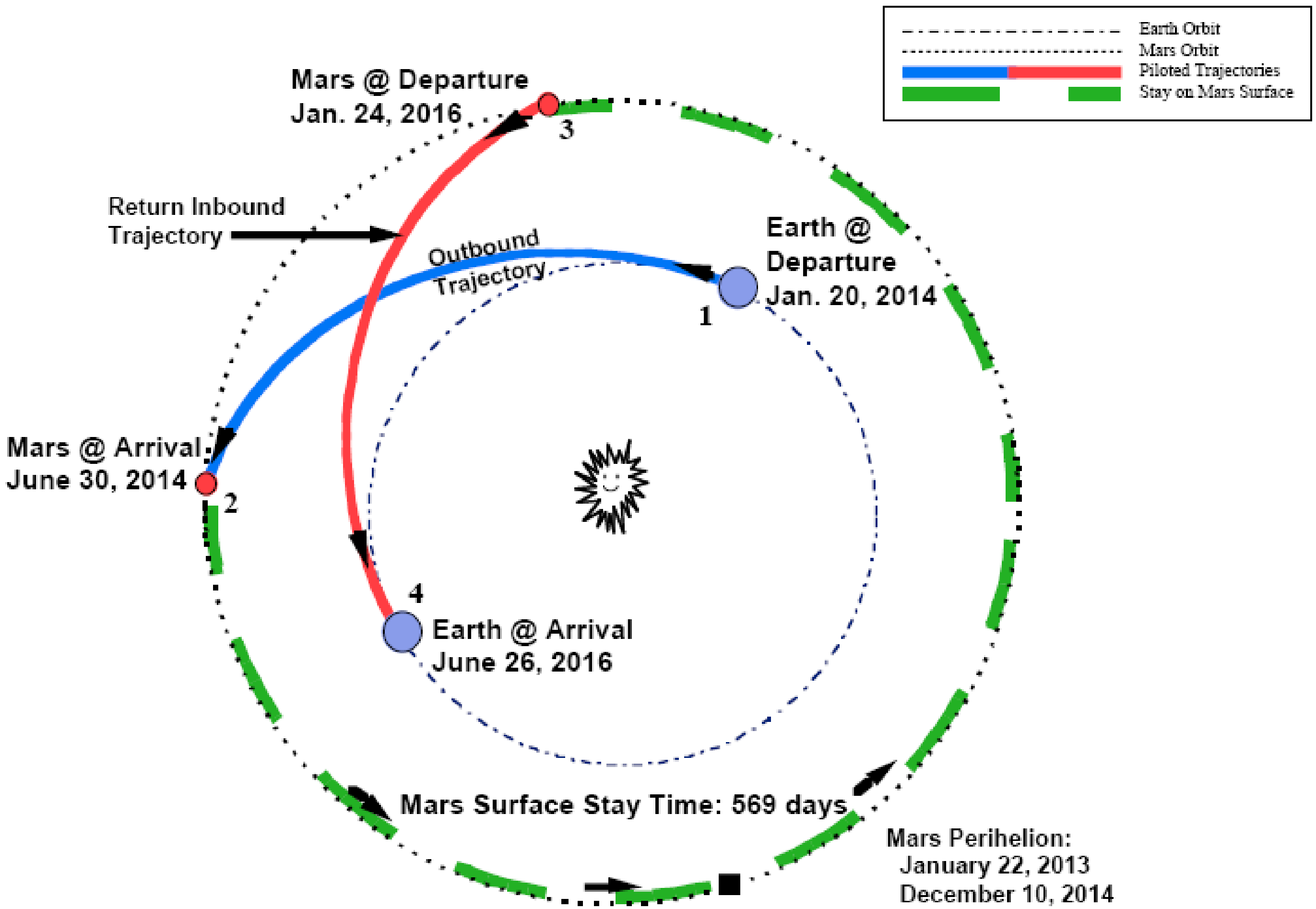


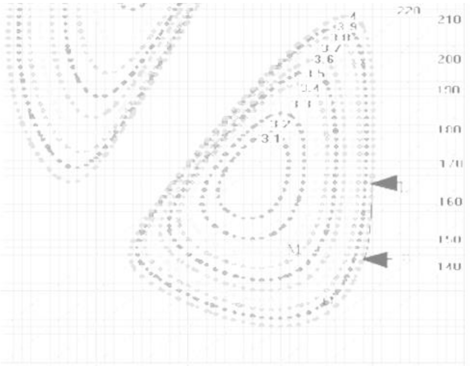
Figure 1. 2014 primary piloted opportunity.

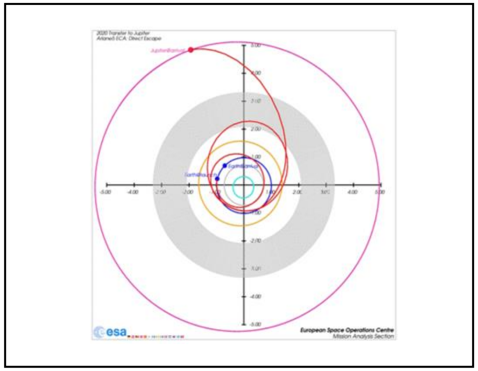
Mission	Launch Date (m/d/yr)	TMI ΔV (m/sec)	Velocity Losses (m/sec)	C_3 (km^2/sec^2)	Mars Arrival Date	Transfer Time (days)
Cargo 1	11/8/11	3,673	92	8.95	8/31/12	297
Cargo 2	11/8/11	3,695	113	8.95	8/31/12	297

Primary Piloted Mission Opportunity 2014

Launch Date	TMI ΔV (m/sec)	Velocity Losses (m/sec)	C_3 (km^2/sec^2)	Outbound TOF (days)	Mars Arrival Date	Mars Stay (days)	Mars Depart Date	TEI ΔV (m/sec)	TOF (days)	Earth Arrival Date	Total TOF (days)
1/20/14	4,019	132	15.92	161	6/30/14	573	1/24/16	1,476	154	6/26/16	888
1/22/14	4,018	131	15.92	180	7/21/14	568	2/9/16	1,476	180	8/7/16	928

6. Interplanetary Trajectories





Gravity assist

ΔV Budget: Earth Departure

Planet	C_3 (km^2/s^2)
Mercury	[56.25]
Venus	6.25
Mars	8.41
Jupiter	77.44
Saturn	106.09
Pluto	[139.24]

Assumption of circular, co-planar orbits and tangential burns

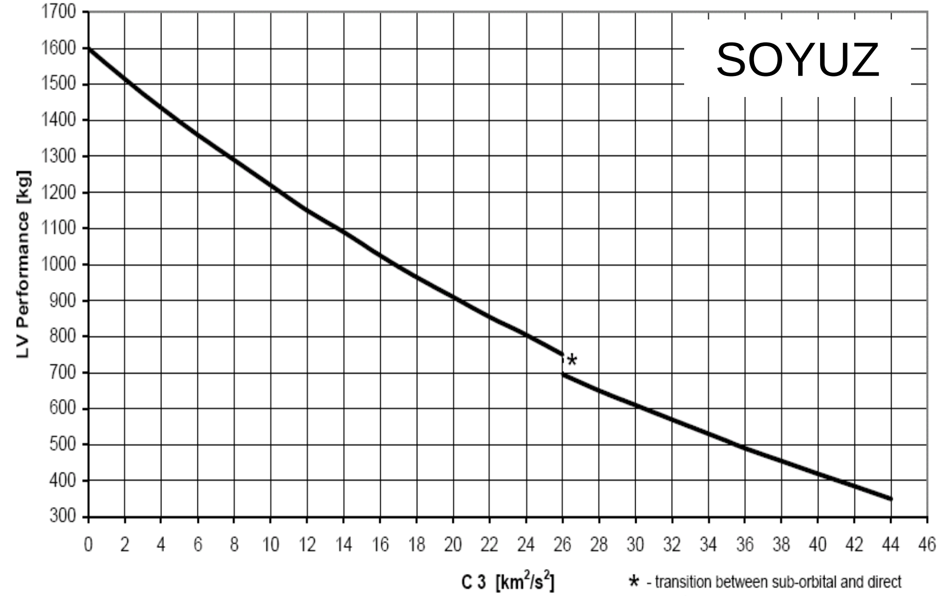


Table 2.9.1-1: Earth Escape Proton M Breeze M Missions

C3 Parameter (km^2/s^2)	Payload Systems Mass (kg)
-5	6270
-2	5890
0	5650
5	5090
10	4580
15	4110
20	3685
25	3295
30	2920
35	2575
40	2260
45	1990
50	1750
55	1525
60	1305
65	1120

ΔV Budget: Arrival at the Planet

A spacecraft traveling to an inner planet is accelerated by the Sun's gravity to a speed notably greater than the orbital speed of that destination planet.

If the spacecraft is to be inserted into orbit about that inner planet, then there must be a mechanism to slow the spacecraft.

Likewise, a spacecraft traveling to an outer planet is decelerated by the Sun's gravity to a speed far less than the orbital speed of that outer planet. Thus there must be a mechanism to accelerate the spacecraft.

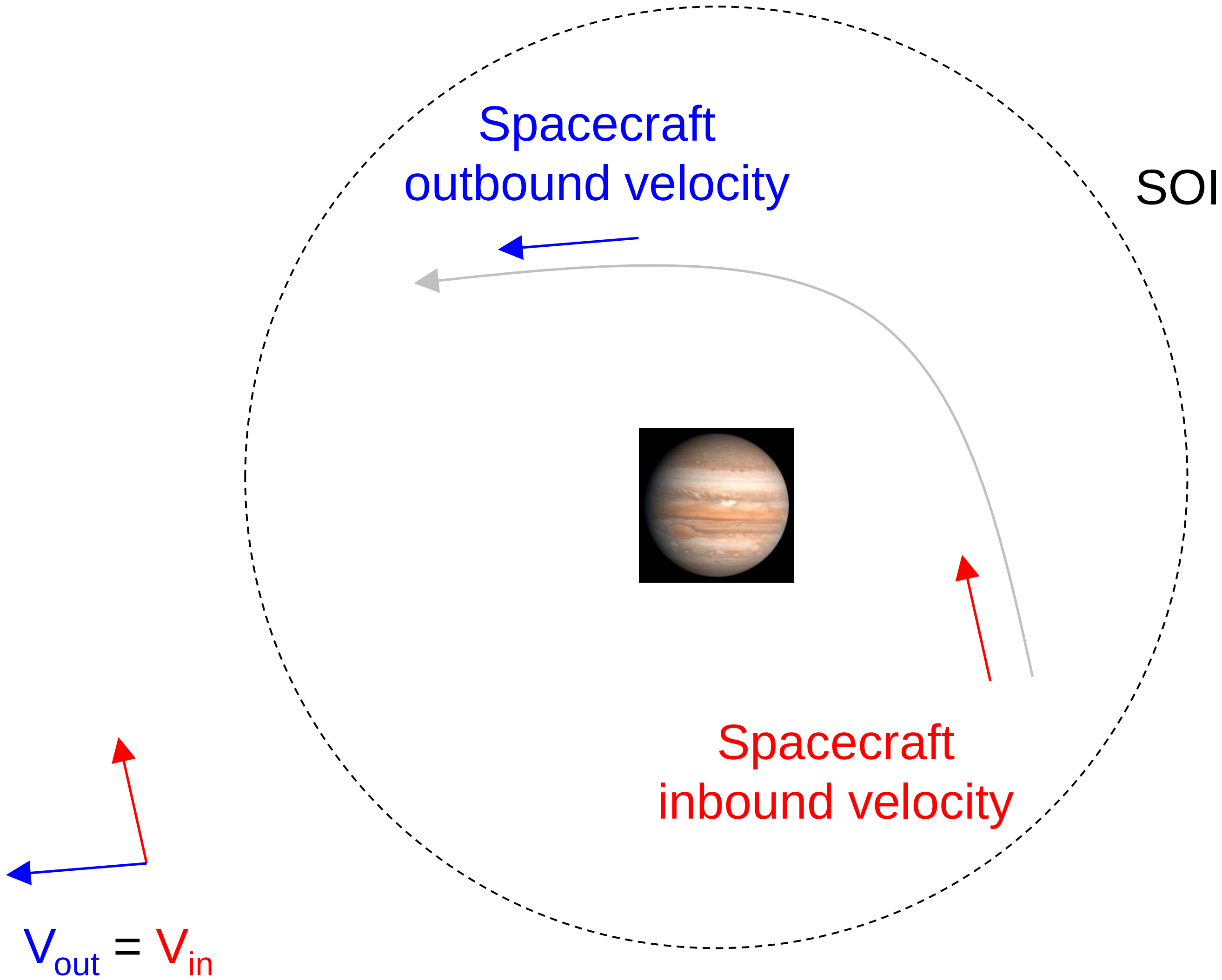
Prohibitive ΔV Budget ? Use Gravity Assist

Also known as planetary flyby trajectory, slingshot maneuver and swingby trajectory.

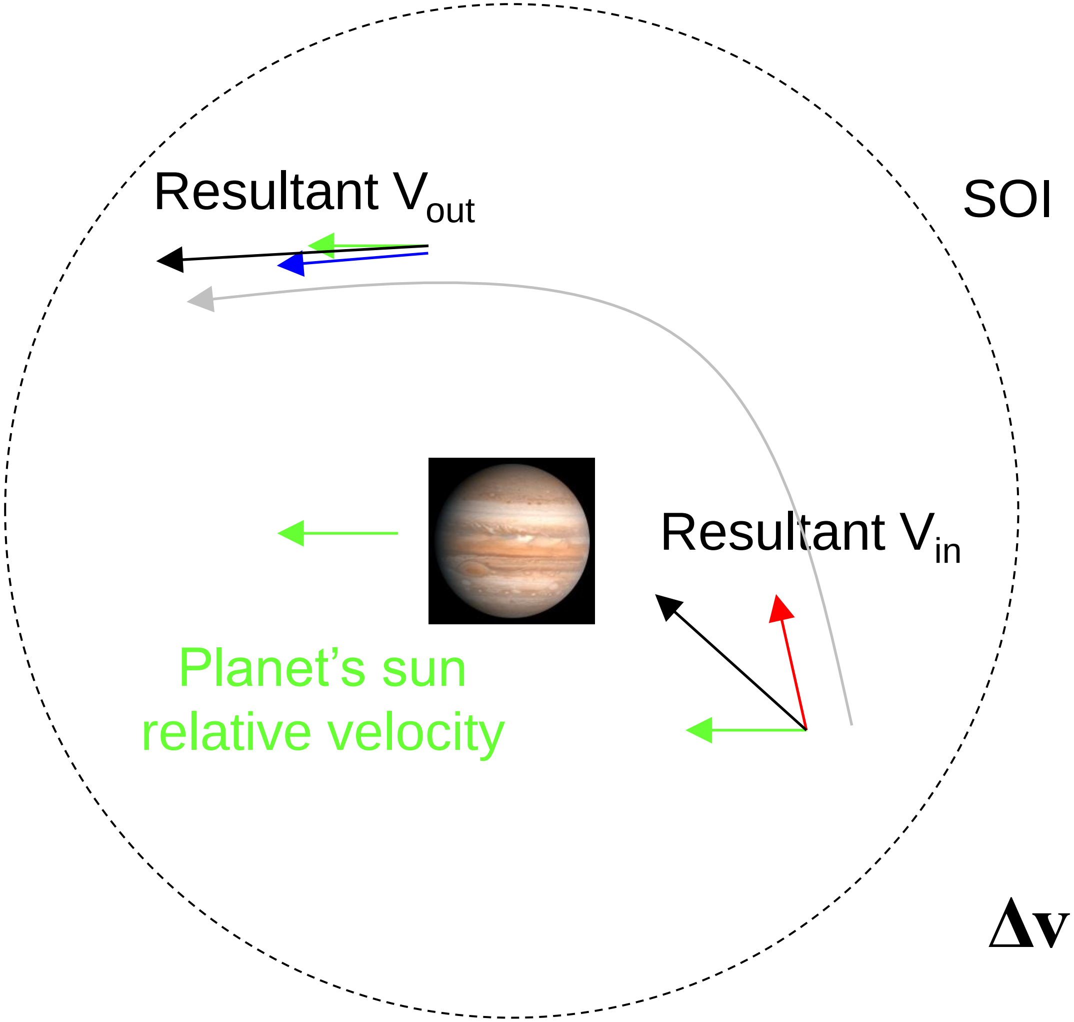
Useful in interplanetary missions to obtain a velocity change without expending propellant.

This free velocity change is provided by the gravitational field of the flyby planet and can be used to lower the Δv cost of a mission.

What Do We Gain ?

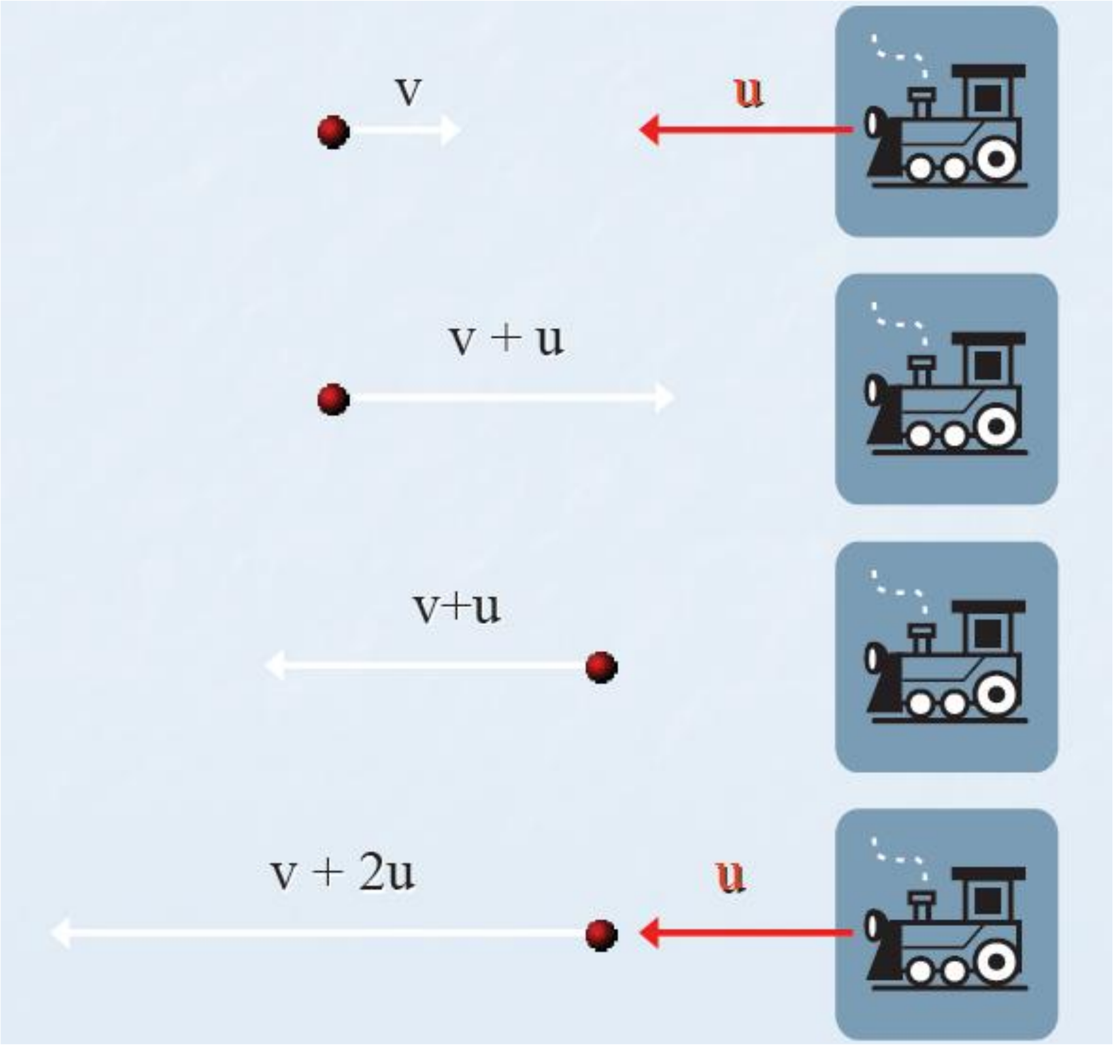


Gravity Assist in the Heliocentric Frame



$$\Delta \mathbf{v} = \mathbf{v}_{\infty,out} - \mathbf{v}_{\infty,in}$$

A Gravity Assist Looks Like an Elastic Collision



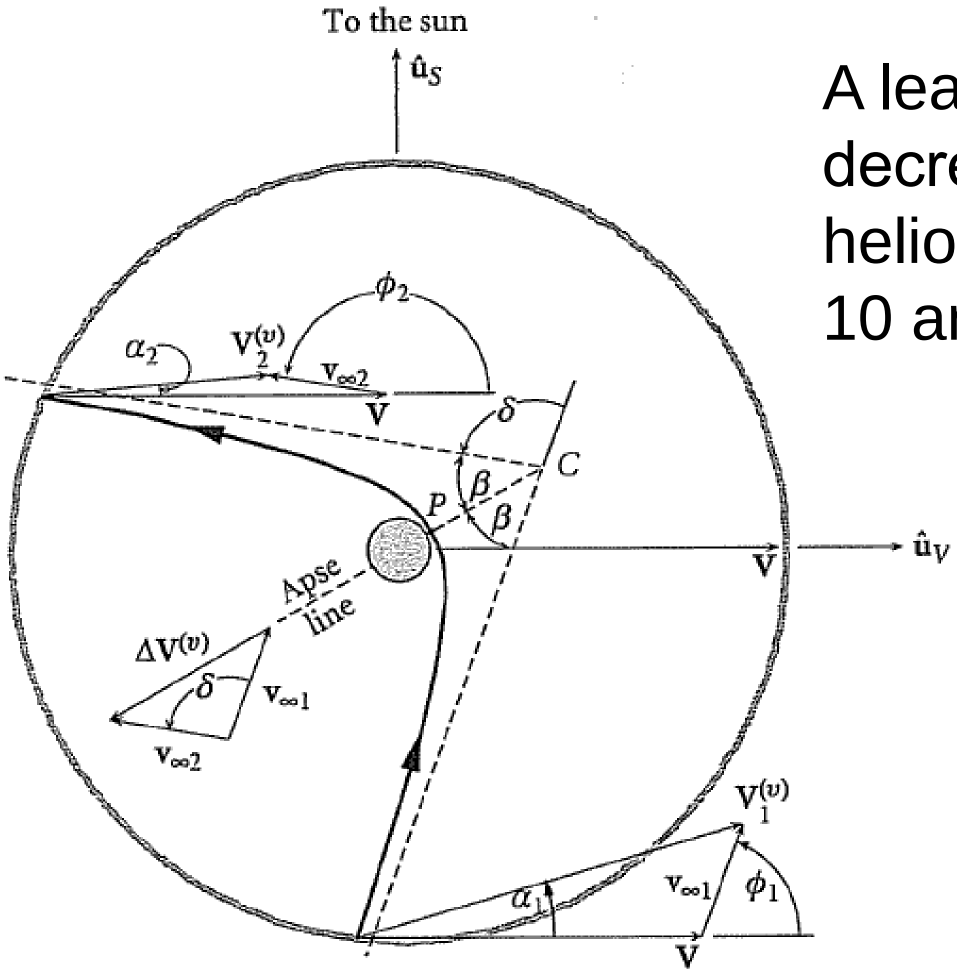
Inertial frame

Frame attached to the train

Frame attached to the train

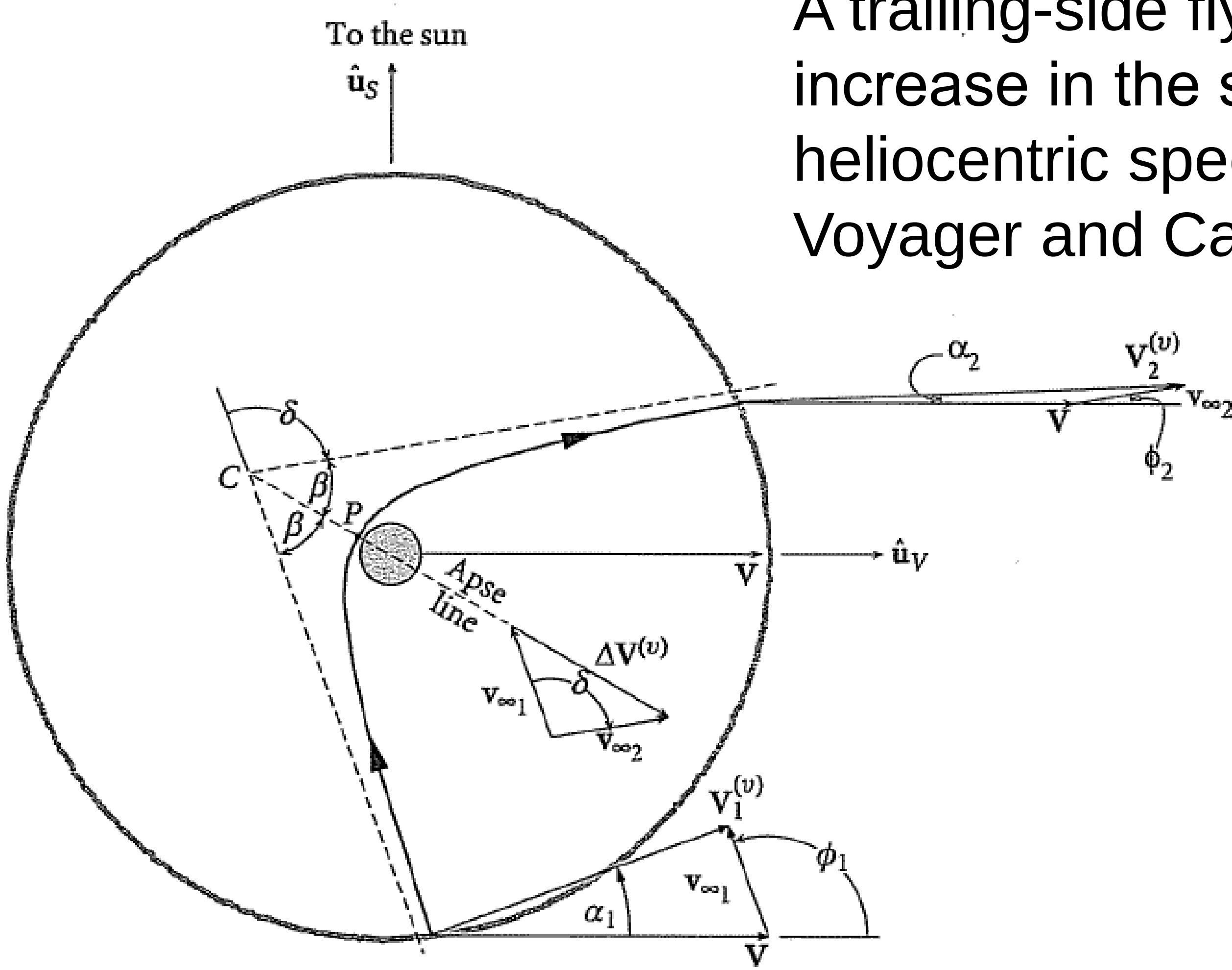
Inertial frame

Leading-Side Planetary Flyby



A leading-side flyby results in a decrease in the spacecraft's heliocentric speed (e.g., Mariner 10 and Messenger).

Trailing-Side Planetary Flyby



A trailing-side flyby results in an increase in the spacecraft's heliocentric speed (e.g., Voyager and Cassini-Huygens).

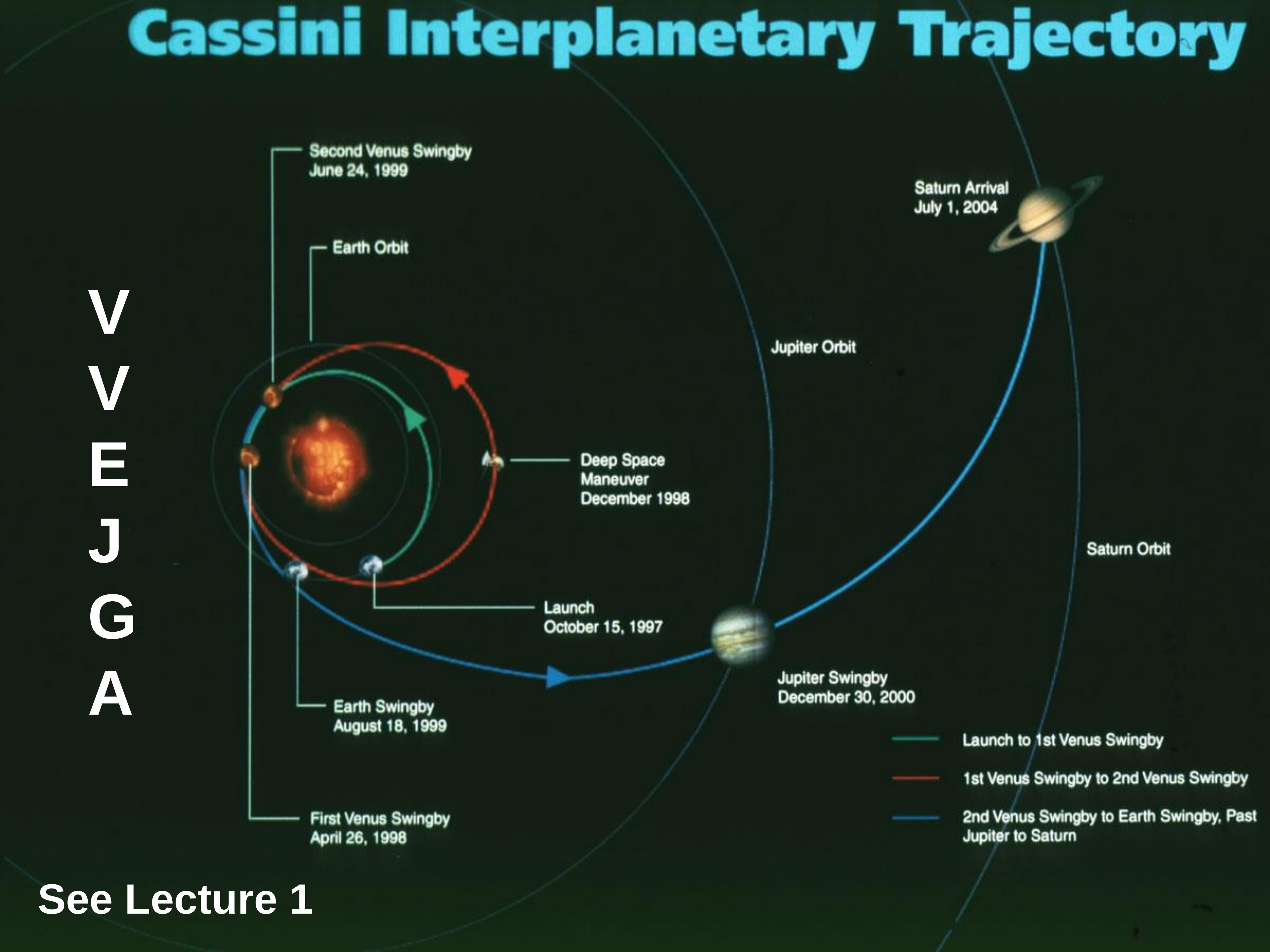
What Are the Limitations ?

Launch windows may be rare (e.g., Voyager).

Presence of an atmosphere (the closer the spacecraft can get, the more boost it gets).

Encounter different planets with different (possibly harsh) environments.

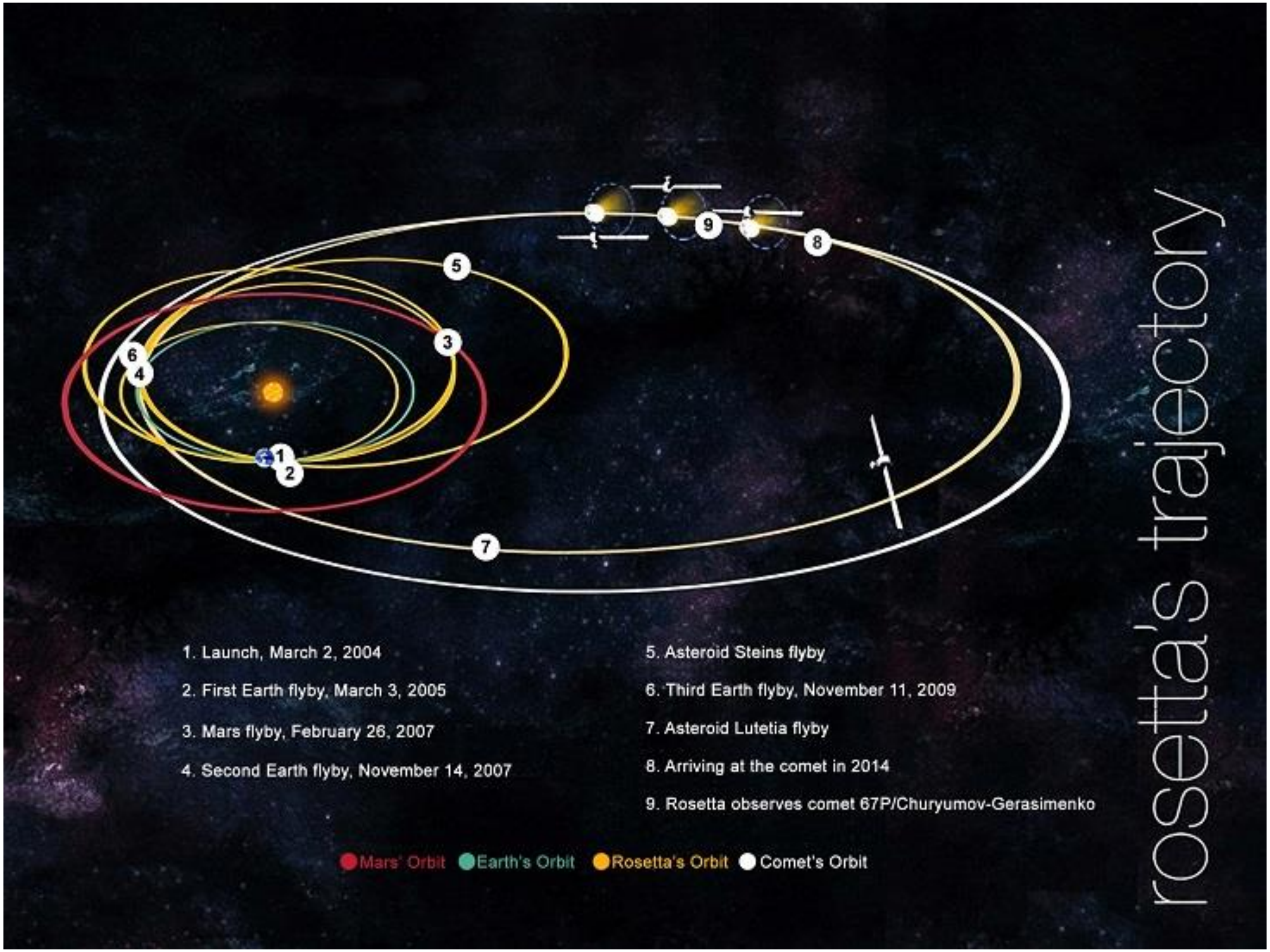
What about flight time ?



V
V
E
J
G
A

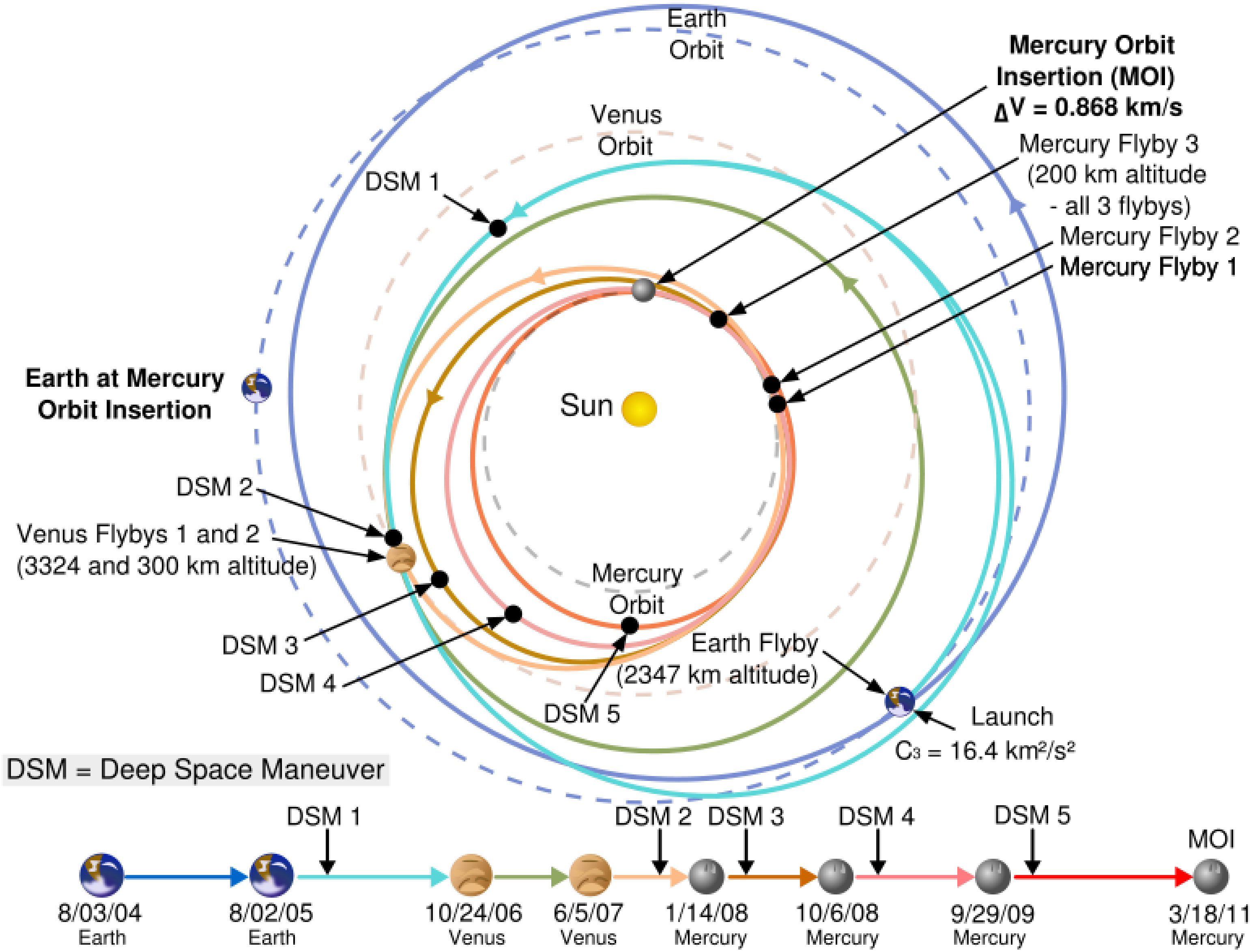
See Lecture 1

Rosetta



rosetta's trajectory
1. Launch, March 2, 2004
2. First Earth flyby, March 3, 2005
3. Mars flyby, February 26, 2007
4. Second Earth flyby, November 14, 2007
5. Asteroid Steins flyby
6. Third Earth flyby, November 11, 2009
7. Asteroid Lutetia flyby
8. Arriving at the comet in 2014
9. Rosetta observes comet 67P/Churyumov-Gerasimenko
Mars' Orbit
Earth's Orbit
Rosetta's Orbit
Comet's Orbit

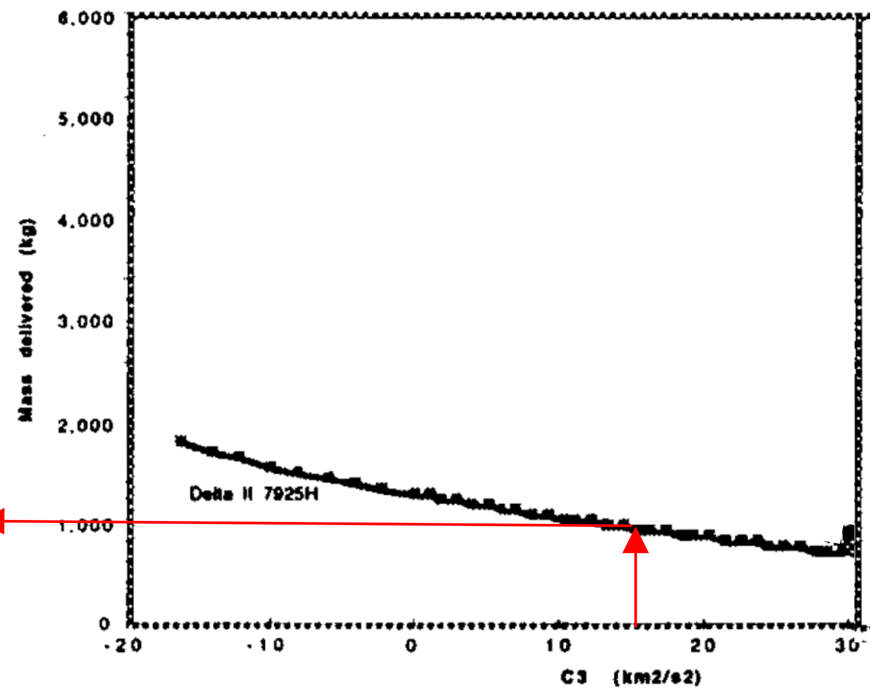
Messenger



Earth Orbit
Venus Orbit
Mercury Orbit Insertion (MOI)
ΔV = 0.868 km/s
Mercury Flyby 3
(200 km altitude
- all 3 flybys)
Mercury Flyby 2
Mercury Flyby 1
DSM 1
Earth at Mercury Orbit Insertion
Sun
DSM 2
Venus Flybys 1 and 2
(3324 and 300 km altitude)
Mercury Orbit
DSM 3
DSM 4
Earth Flyby
(2347 km altitude)
DSM 5
Launch
C3 = 16.4 km²/s²
DSM = Deep Space Maneuver
DSM 1
DSM 2
DSM 3
DSM 4
DSM 5
MOI
8/03/04
Earth
8/02/05
Earth
10/24/06
Venus
6/5/07
Venus
1/14/08
Mercury
10/6/08
Mercury
9/29/09
Mercury
3/18/11
Mercury

Technicians prepare MESSENGER for transfer to a hazardous processing facility prior to loading the spacecraft's complement of hypergolic propellants.

Organization	NASA
Major contractors	Johns Hopkins University Applied Physics Laboratory (JHUAPL)
Mission type	Fly-by(s)/orbit
Flyby of	Earth, Venus, Mercury
Satellite of	Mercury
Orbital insertion date	ETA: 2011-03-18 02:14:00 UTC
Launch date	2004-08-03 06:15:56 UTC elapsed: 5 years, 8 months, and 6 days
Launch vehicle	Delta II 7925H-9.5
Launch site	Space Launch Complex 17-A Cape Canaveral Air Force Station
COSPAR ID	2004-030A
Home page	messenger.jhuapl.edu
Mass	1,093 kg (2,410 lb)

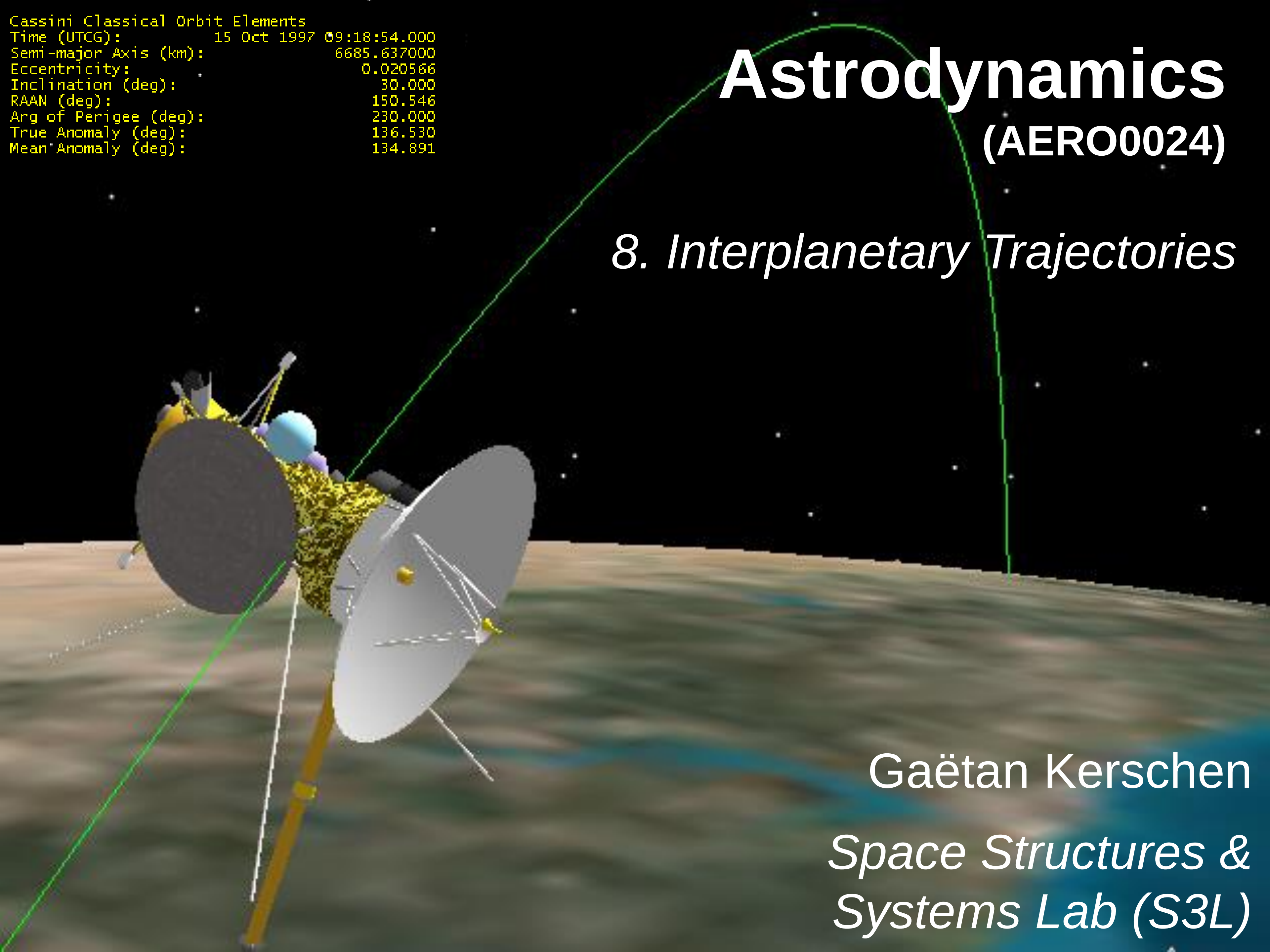


Hohmann Transfer vs. Gravity Assist

Gravity assist

Planet	C3 (km^2/s^2)	Transfer time (days)	Real mission	C3 (km^2/s^2)	Transfer time (days)
Mercury	[56.25]	105	Messenger	16.4	2400
Saturn	106.09	2222	Cassini Huygens	16.6	2500

Remark: the comparison between the transfer times is difficult, because it depends on the target orbit. The transfer time for gravity assist mission is the time elapsed between departure at the Earth and first arrival at the planet.



Astrodynamics
(AERO0024)

8. Interplanetary Trajectories

Gaëtan Kerschen

Space Structures & Systems Lab (S3L)