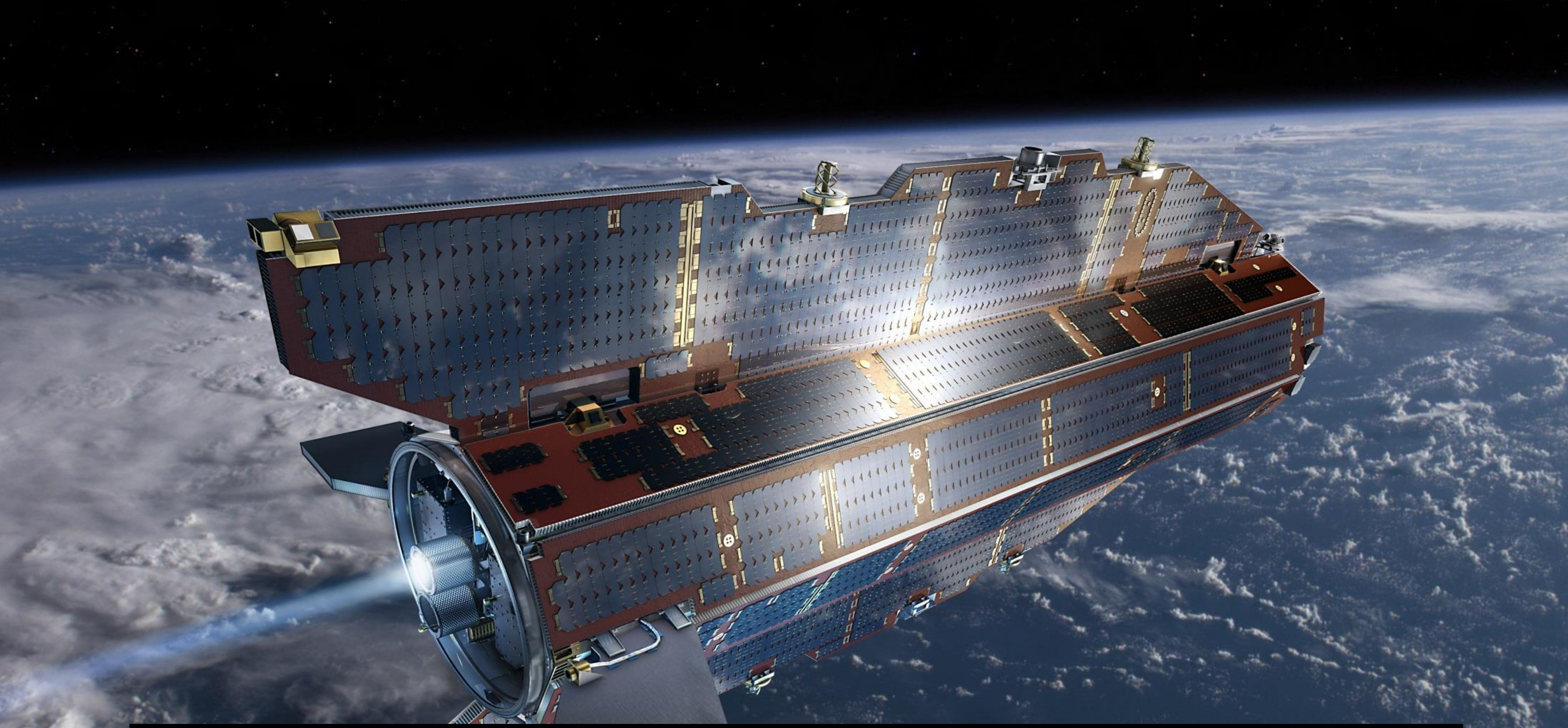
Flight and Orbital Mechanics

Lecture slides



Flight and Orbital Mechanics

AE2-104, lecture hours 17+18: Perturbations

Ron Noomen

October 25, 2012

Example: solar sail spacecraft

Questions:

- what is the purpose of this mission?
- where is the satellite located?
- why does it use a solar sail?
- ….

[Wikipedia, 2010]

Overview

- Orbital mechanics (recap)

- Irregularities gravity field
- Third-body perturbations
- Atmospheric drag
- Solar radiation pressure
- Thrust

- Relativistic effects
- Tidal forces
- Thermal forces ← Special missions only; not treated here
- Impacts debris/micrometeoroids

Learning goals

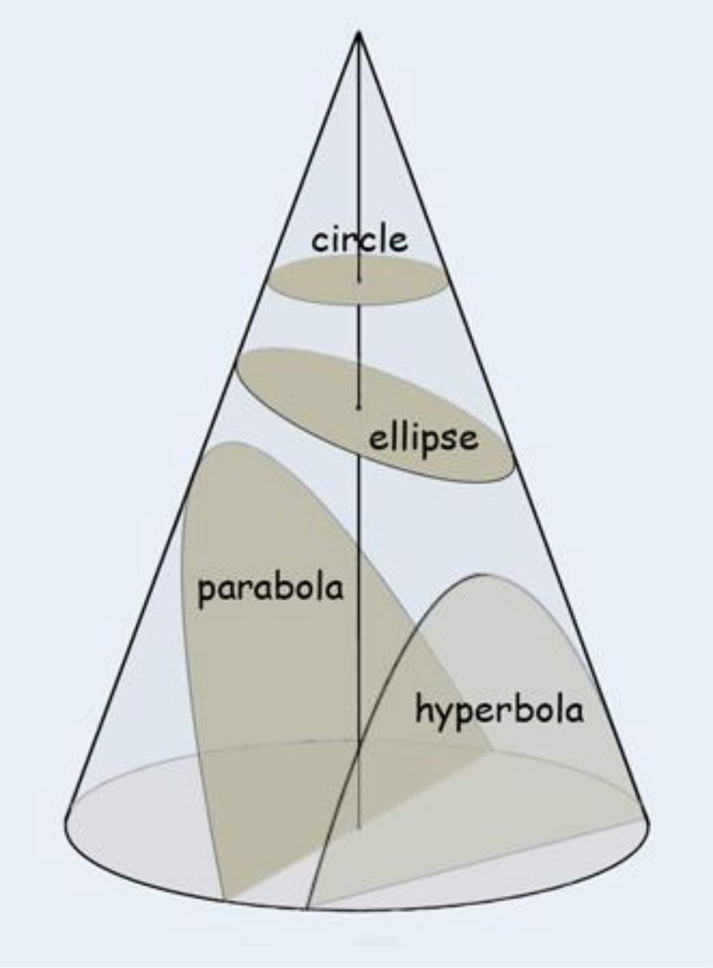
The student should be able to:

- mention and describe the various perturbing forces that may act on an arbitrary spacecraft;
- quantify the resulting accelerations;
- make an assessment of the importance of the different perturbing forces, depending on the specifics of the mission (phase) at hand.

Lecture material:

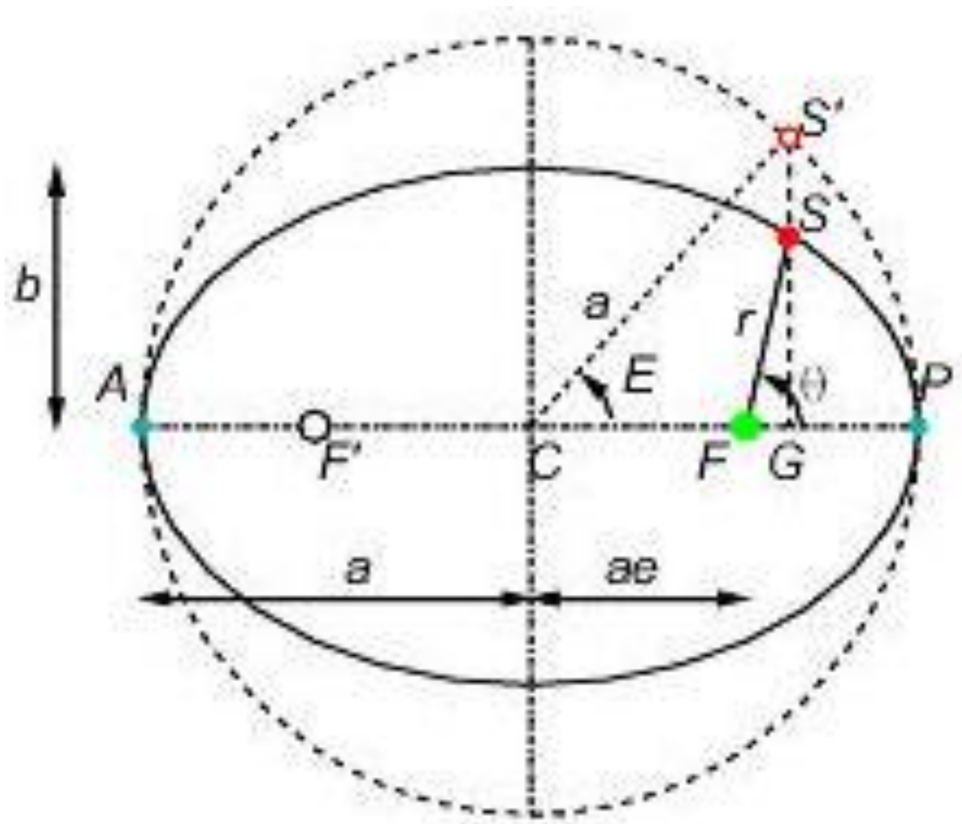
- these slides (incl. footnotes)

2-dimensional Kepler orbits



[Seligman, 2010]

2-dimensional Kepler orbits



[Cornellise, Schöyer and Wakker, 1979]

a: **semi-major axis [m]**

e: **eccentricity [-]**

θ: **true anomaly [deg]**

E: **eccentric anomaly [deg]**

2-dimensional Kepler orbits: general equations

$$r = \frac{a\,(1-e^2)}{1+e\cos\theta} = \frac{p}{1+e\cos\theta} \;;\; r_p = a\,(1-e) \;;\; r_a = a\,(1+e)$$

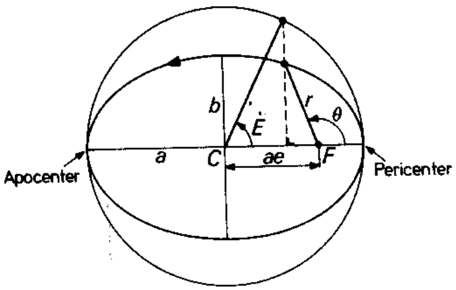
$$E_{tot} = E_{kin} + E_{pot} = \frac{V^2}{2} - \frac{\mu}{r} = -\frac{\mu}{2a}$$

$$V^2 = \mu\left(\frac{2}{r} - \frac{1}{a}\right) \;;\; V_{circ} = \sqrt{\frac{\mu}{r}} = \sqrt{\frac{\mu}{a}} \;;\; V_{esc} = \sqrt{\frac{2\mu}{r}}$$

$$T = 2\pi\sqrt{\frac{a^3}{\mu}}$$

2-dimensional Kepler orbits: equations (cnt'd)

ellips: $0 \le e < 1$ $\quad a > 0 \quad$ $E_{tot} < 0$



$$n = \sqrt{\frac{\mu}{a^3}}$$

$$\tan\frac{\theta}{2} = \sqrt{\frac{1+e}{1-e}}\tan\frac{E}{2}$$

Kepler's Equation

➔ $$M = E - e\sin E$$

$$M = n(t - t_0)$$

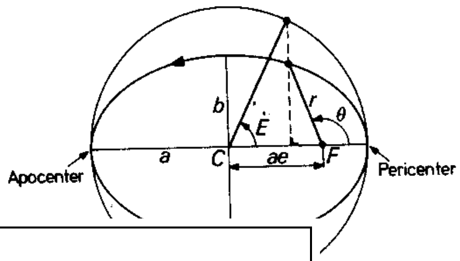
$$E_{i+1} = E_i + \frac{M - E_i + e\sin E_i}{1 - e\cos E_i}$$

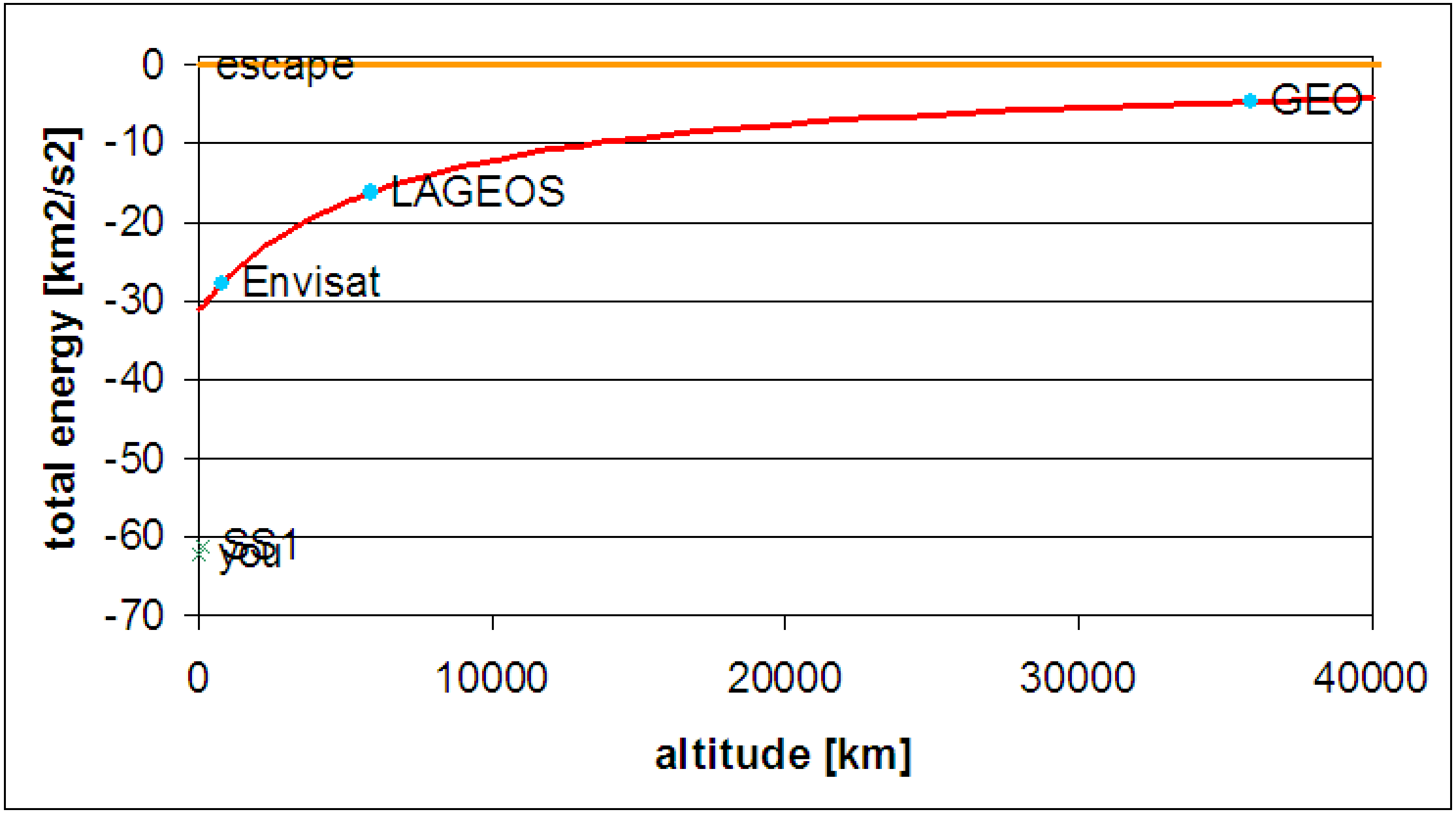
$$r = a(1 - e\cos E)$$

satellite	altitude [km]	specific energy [km²/s²/kg]
launch platform	0	-62.4
SpaceShipOne	100+ (culmination)	-61.4
imaginary sat	100	-30.8
Envisat	800	-27.8
LAGEOS	5900	-16.2
GEO	35900	-4.7

2-dimensional Kepler orbits: equations (cnt'd)

ellips: $0 \leq e < 1$ $\quad a > 0 \quad$ $E_{tot} < 0$





2-dimensional Kepler orbits: equations (cnt'd)

parabola: *e* = 1 　　 *a* = ∞ 　　 E_{tot} = 0

$$r = \frac{p}{1 + \cos\theta}$$

$$M = \frac{1}{2}\tan\frac{\theta}{2} + \frac{1}{6}\tan^3\frac{\theta}{2}$$

$$M = n\,(t - t_0)$$

$$n = \sqrt{\frac{\mu}{p^3}}$$

$$V^2 = V_{esc}^2 = \frac{2\mu}{r}$$

2-dimensional Kepler orbits: equations (cnt'd)

hyperbola: $e > 1$ $\quad a < 0$ $\quad E_{tot} > 0$

$$\tan\frac{\theta}{2} = \sqrt{\frac{e+1}{e-1}}\,\tanh\frac{F}{2}$$

$$M = e\sinh F - F$$

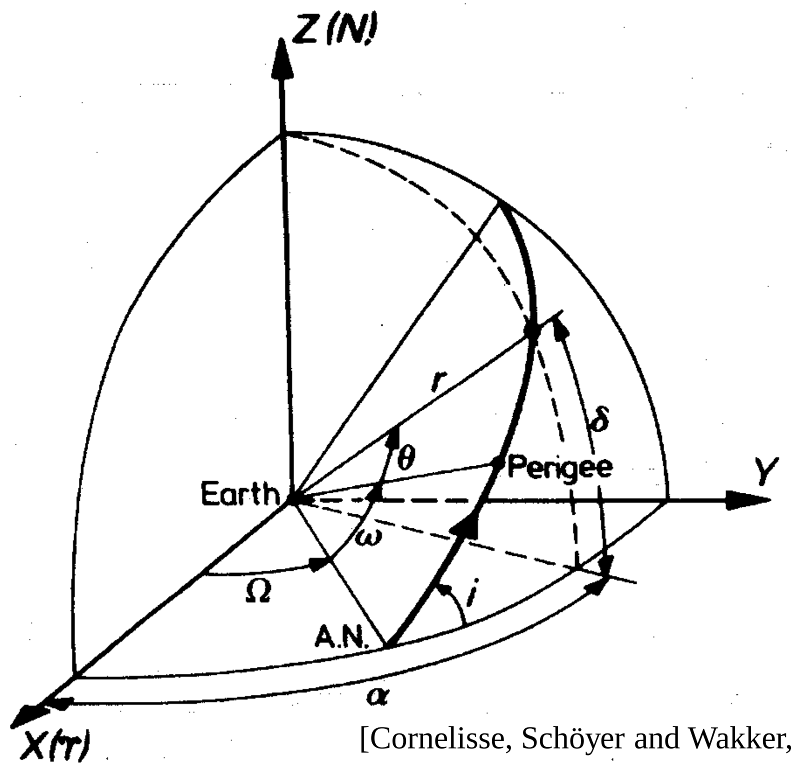
$$M = n\,(t - t_0)$$

$$n = \sqrt{\frac{\mu}{(-a)^3}}$$

$$r = a\,(1 - e\cosh F)$$

$$V^2 = V_{esc}^2 + V_\infty^2 = \frac{2\mu}{r} + V_\infty^2$$

3-dimensional Kepler orbits



[Cornelisse, Schöyer and Wakker, 1979]

i: **inclination [deg]**

Ω: **right ascension of the ascending node, or longitude of the ascending node [deg]**

ω: **argument of pericenter [deg]**

$u = \omega + \theta$ **: argument of latitude [deg]**

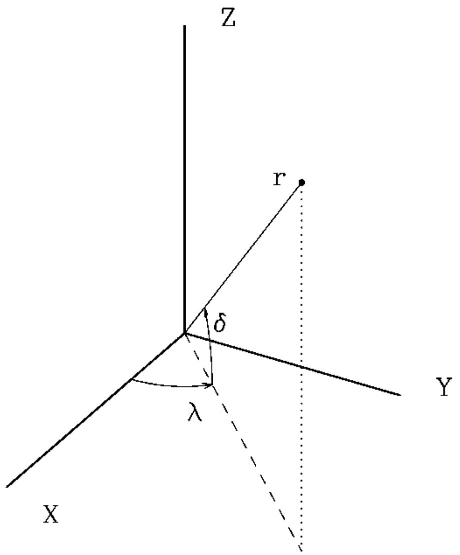
coordinates transformations

1) from spherical (r,λ,δ) (λ in X-Y plane; δ w.r.t. X-Y plane) to cartesian (x,y,z):

$$x = r \cos \delta \cos \lambda$$

$$y = r \cos \delta \sin \lambda$$

$$z = r \sin \delta$$



2) from cartesian (x,y,z) to spherical (r,λ,δ):

$$r = \mathrm{sqrt}\,(x^2 + y^2 + z^2)$$

$$r_{xy} = \mathrm{sqrt}\,(x^2 + y^2)$$

$$\lambda = \mathrm{atan2}\,(y / r_{xy} , x / r_{xy})$$

$$\delta = \mathrm{asin}\,(z / r)$$

Orbital perturbations: introduction

Questions:

- **what forces?**
- **magnitude of forces?**
- **how to model/compute in satellite orbit?**
- **analytically? numerically?**
- **accuracy?**
- **efficiency?**

Inclusion in orbit modeling

Option 1: include directly in equation of motion

$$\frac{d\mathbf{x}^2}{dt^2} = \mathbf{a}_{main} + \mathbf{a}_{\Delta grav} + \mathbf{a}_{drag} + \mathbf{a}_{solrad} + \mathbf{a}_{3rdbody} + etcetera$$

Option 2: express as variation of orbital elements

$$e.g. \quad \frac{da}{dt} = \frac{2a^2}{\sqrt{\mu p}}\left[Se\sin\theta + N\frac{p}{r} \right]$$

(not further treated here; cf. ae4-874 and ae4-878)

Perturbations:

- **Irregularities gravity field**
- **Third body**
- **Atmospheric drag**
- **Solar radiation pressure**
- **Thrust**

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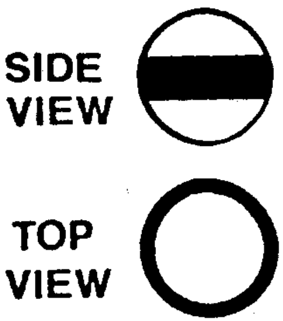
Irregularities gravity field

Already treated in lecture hours 15+16



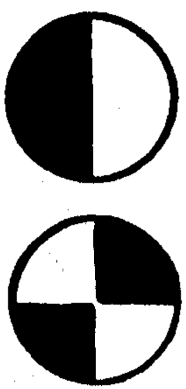
2,0

e.g., North-South acceleration due to J_2:

$$acc = -3\mu J_2 R_e^2 r^{-4} \sin(\delta)\cos(\delta)$$

e.g., East-West acceleration due to $J_{2,2}$:

$$acc = -6\mu J_{2,2} R_e^2 r^{-4} 3\cos(\delta)\sin(2(\lambda - \lambda_{2,2}))$$



2,2

Perturbations:

- **Irregularities gravity field**
- **Third body**
- **Atmospheric drag**
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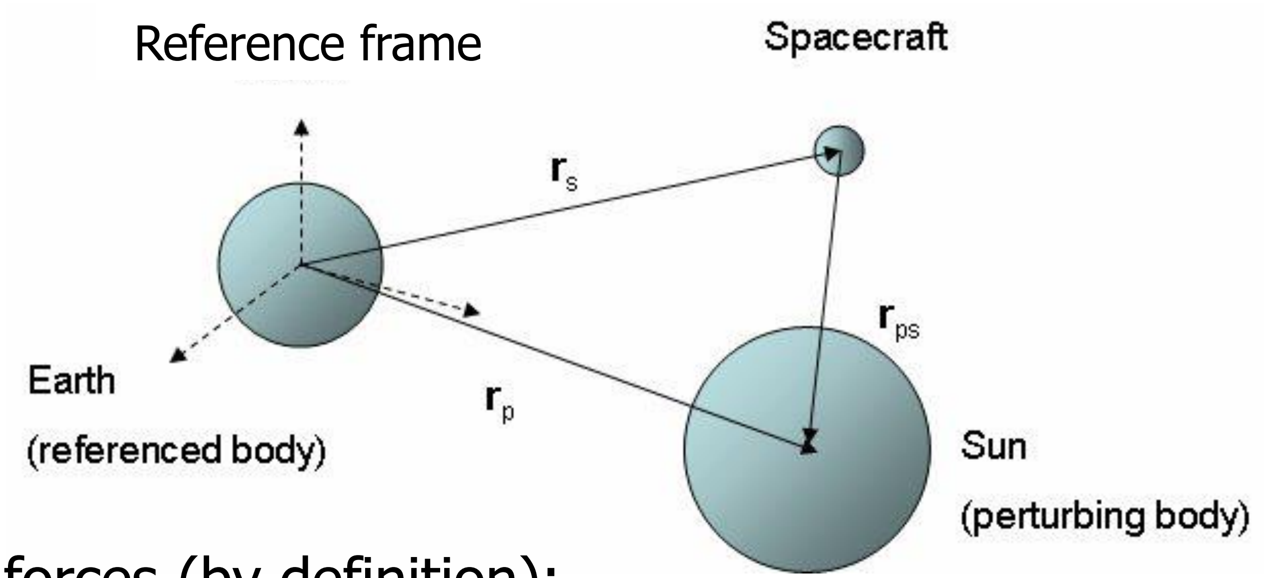
- **Thrust**

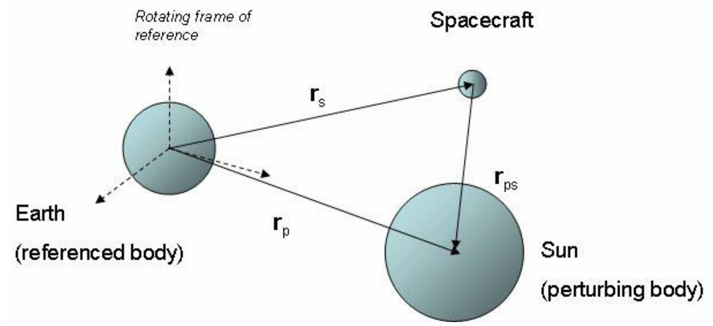
Third-body perturbations



Attractional forces (by definition):

- satellite <-> Earth
- satellite <-> perturbing body
- Earth <-> perturbing body

Third-body perturbations (cnt'd)



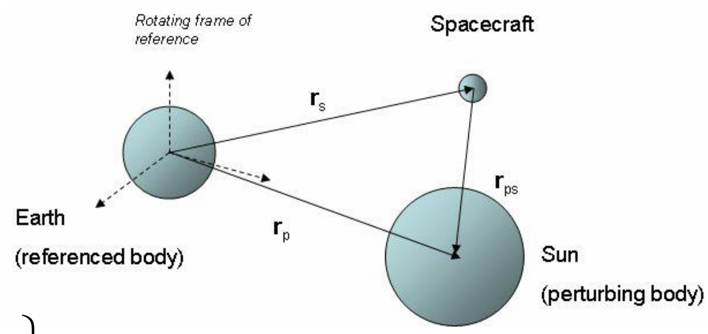
Attractional forces (practice):

- Earth attracts satellite
- perturbing body attracts satellite
- perturbing body attracts Earth
- net effect counts

$M_{sat} << M_E, M_S$

$$\ddot{\mathbf{r}}_s = -G \frac{M_{main}}{r_s^3} \mathbf{r}_s + G M_p \left(\frac{\mathbf{r}_{ps}}{r_{ps}^3} - \frac{\mathbf{r}_p}{r_p^3} \right)$$

Third-body perturbations (cnt'd)



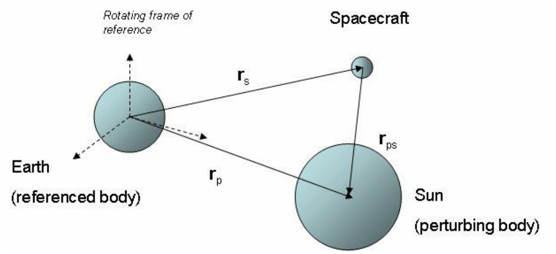
perturbing acceleration $\mathbf{a_p}$:

$$\mathbf{a_p} = \mu_p \left\{ \frac{\mathbf{r}_{ps}}{\|\mathbf{r}_{ps}\|^3} - \frac{\mathbf{r}_p}{\|\mathbf{r}_p\|^3} \right\}$$

maximum when on straight line:

$$\left(\frac{a_p}{a_{main}} \right)_{\max} = 2 \frac{m_p}{m_{main}} \left(\frac{r_s}{r_p} \right)^3$$

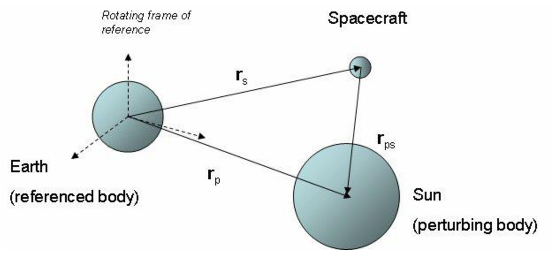
Third-body perturbations (cnt'd)

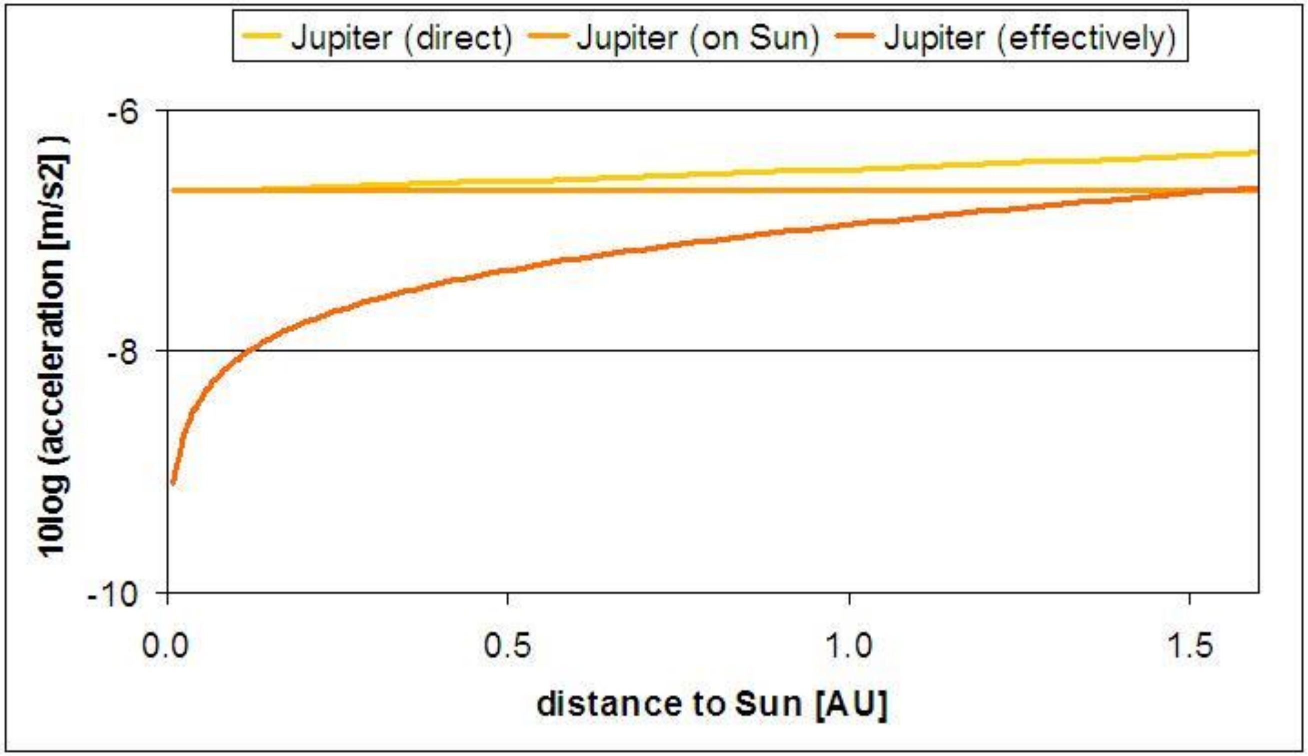


2 situations:

1. heliocentric (*i.e.*, orbits around Sun)
2. planetocentric (*i.e.*, orbits around Earth)

Third-body perturbations (cnt'd)

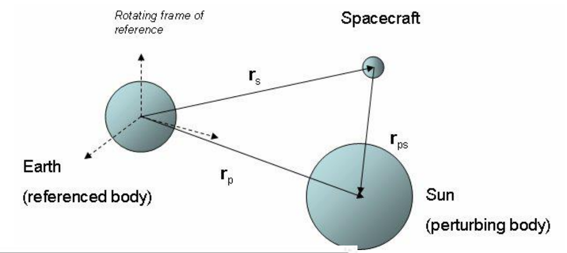


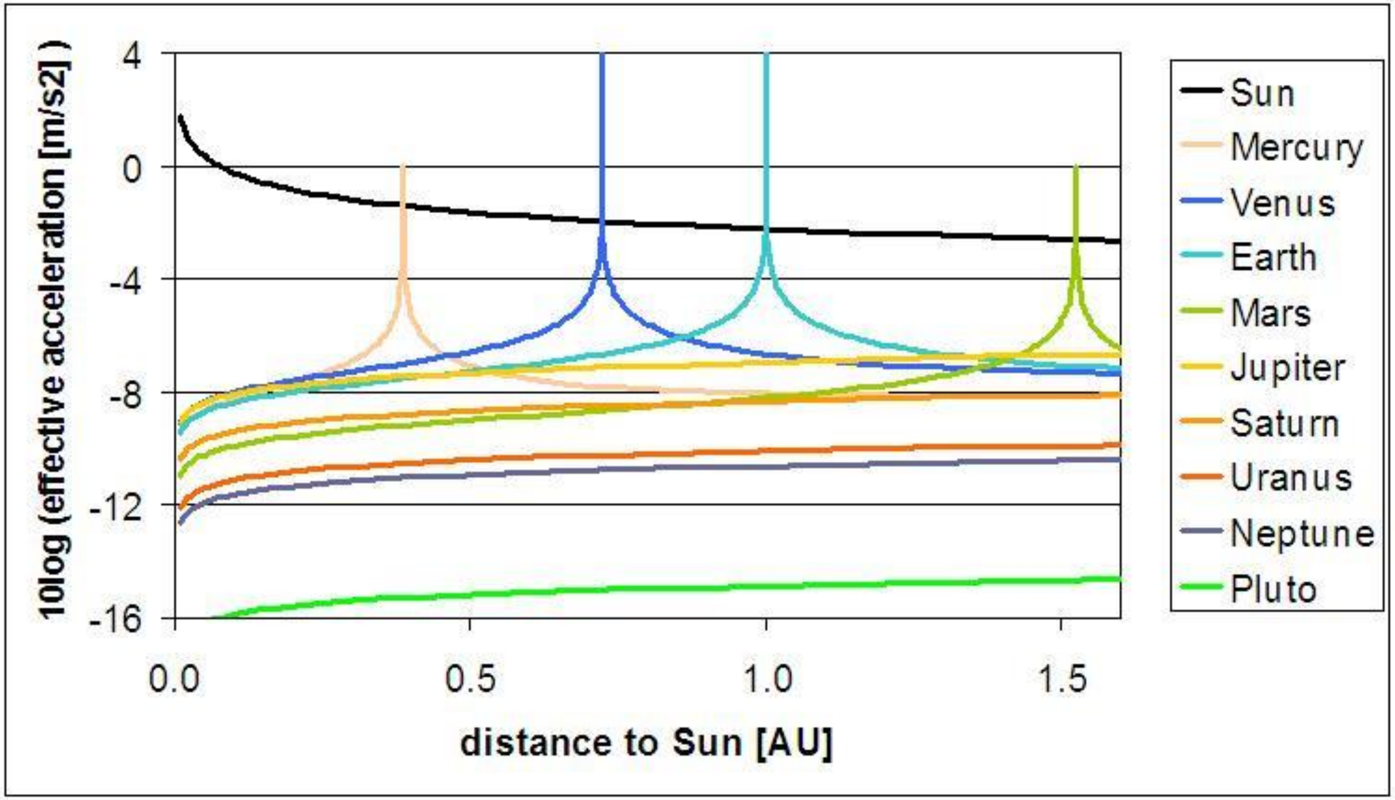


Conclusions:

- influence increases with distance to Sun
- perturbation $O(10^{-7})$ m/s^2

Third-body perturbations (cnt'd)

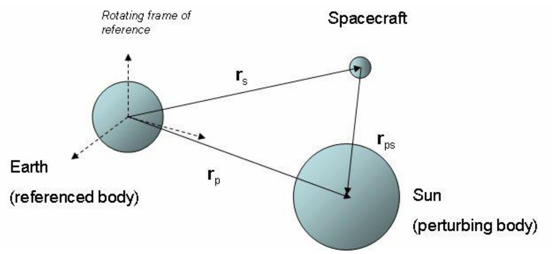


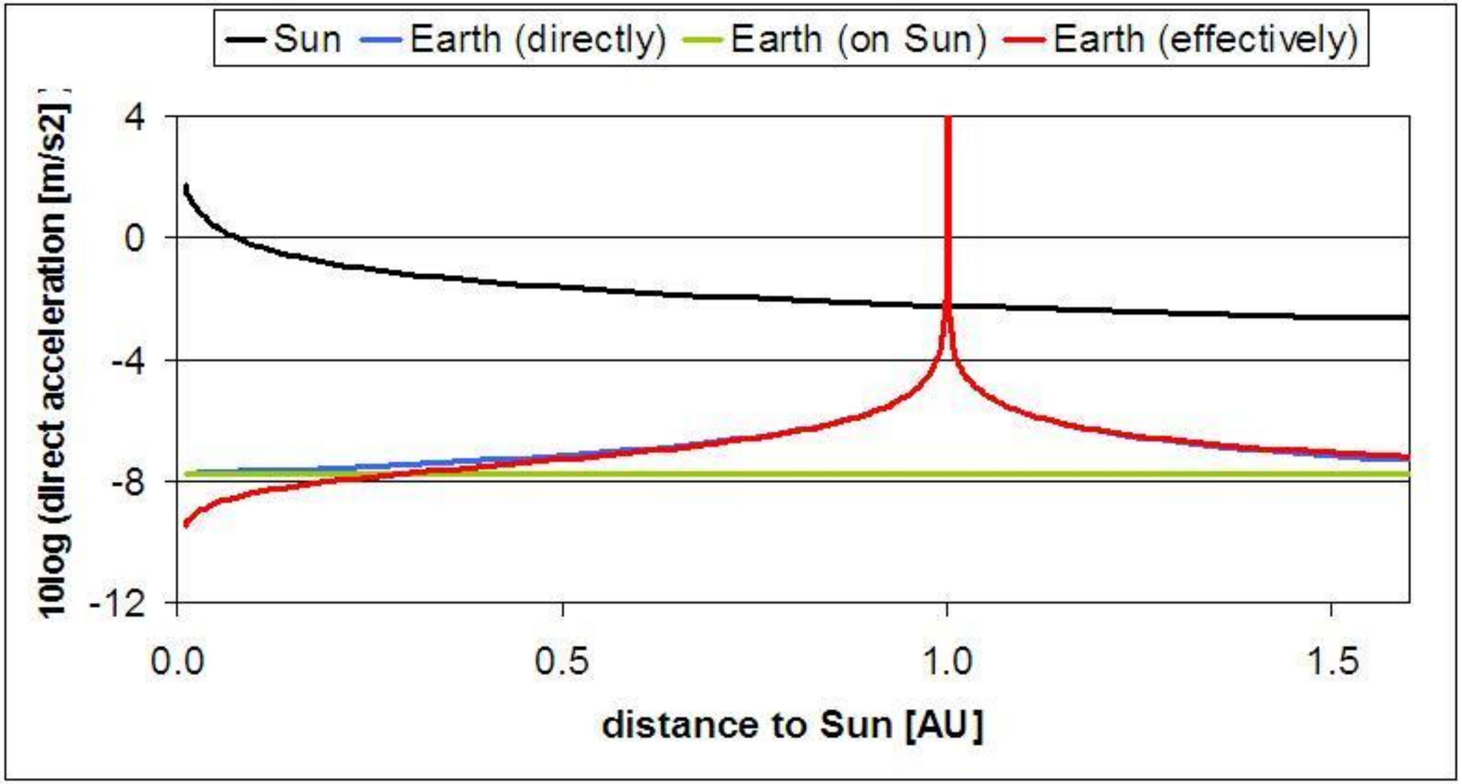


Conclusions:

- influence Sun decreases with distance to Sun
- influence planets increases with distance to Sun
- acceleration from Sun $O(10^{-2})$ m/s^2; dominant (central body!!)
- near planet: 3rd body becomes dominant

Third-body perturbations (cnt'd)

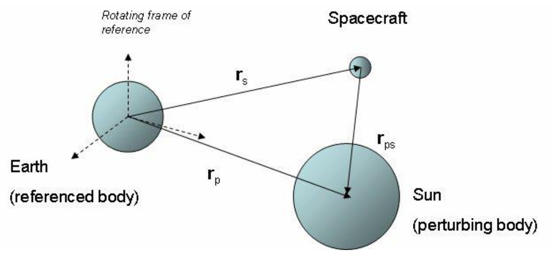




Conclusions:

- in Solar System: Sun dominant
- near Earth: Earth itself dominant
- central body vs. 3rd body perturbation ??

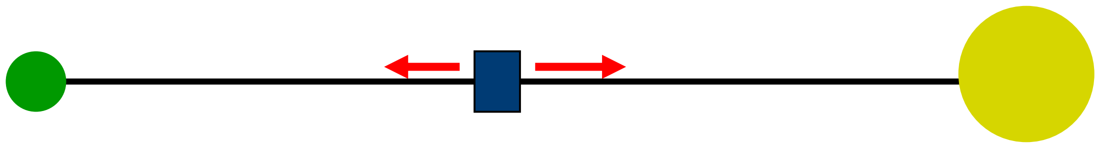
Third-body perturbations (cnt'd)



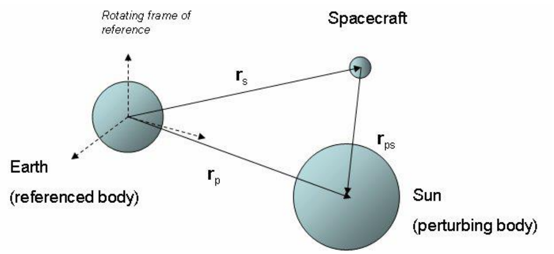
Sphere of Influence:

- area around planet where gravity from planet is dominant (compared with gravity of other celestial bodies)
- 3-dimensional shape
- boundary
- 1st-order approximation: sphere with constant radius
- definition for determination location:

relative acceleration w.r.t. system 1 = relative acceleration w.r.t. system 2



Third-body perturbations (cnt'd)



$$\frac{acc_{Sun,3rd}}{acc_{Earth,main}} = \frac{acc_{Earth,3rd}}{acc_{Sun,main}}$$

without derivation :

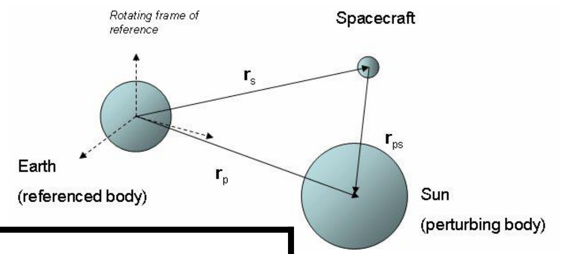
$$r_{SoI} = r_{3rd}\left(\frac{M_{main}}{M_{3rd}}\right)^{0.4}$$

example : Earth main body, Sun 3rd body :

$$r_{SoI,Earth} = dist_{Earth-Sun}\left(\frac{M_{Earth}}{M_{Sun}}\right)^{0.4} \approx 930{,}000\ km$$

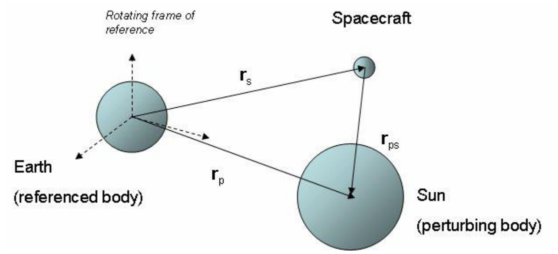
Moon is at 384,000 km....

Third-body perturbations (cnt'd)



planet	Sphere of Influence		
	[km]	[% AU]	[% distance planet-Sun]
Mercury	1.1×10^5	0.08	0.2
Venus	6.2×10^5	0.4	0.6
Earth	9.3×10^5	0.6	0.6
Mars	5.8×10^5	0.4	0.3
Jupiter	4.8×10^7	32.2	6.2
Saturn	5.5×10^7	36.5	3.8
Uranus	5.2×10^7	34.6	1.8
Neptune	8.7×10^7	57.9	1.9
Pluto	3.2×10^6	2.1	0.1

Third-body perturbations (cnt'd)



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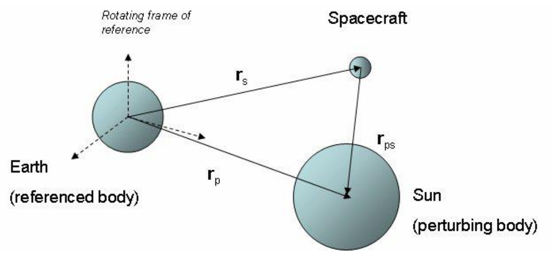
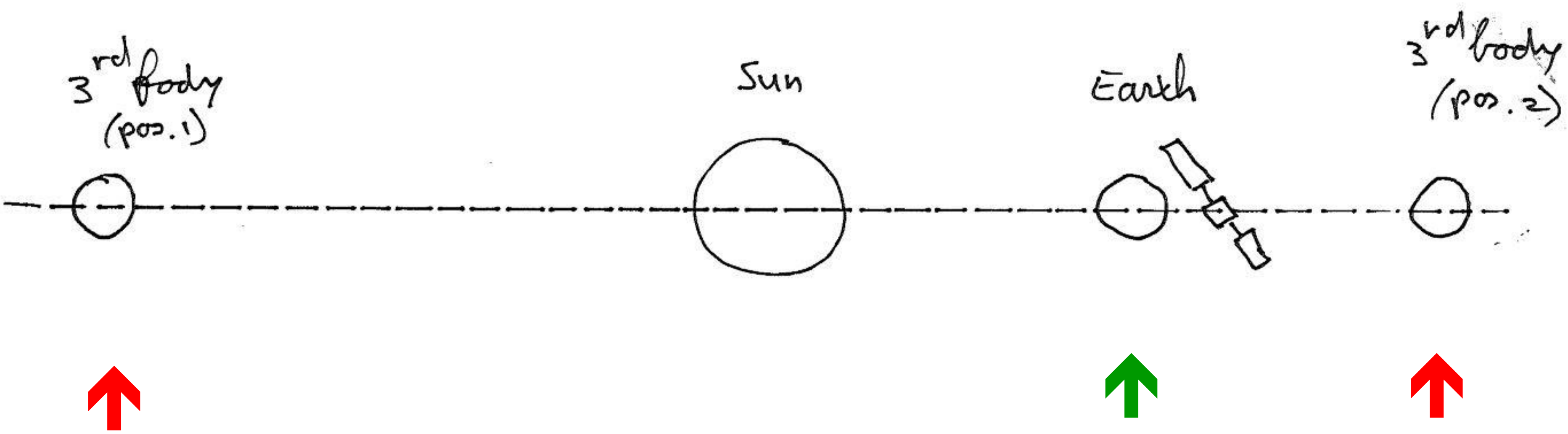
2. planetocentric (*i.e.*, orbits around Earth)

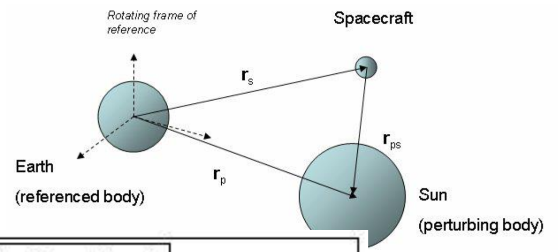
Third-body perturbations (cnt'd)

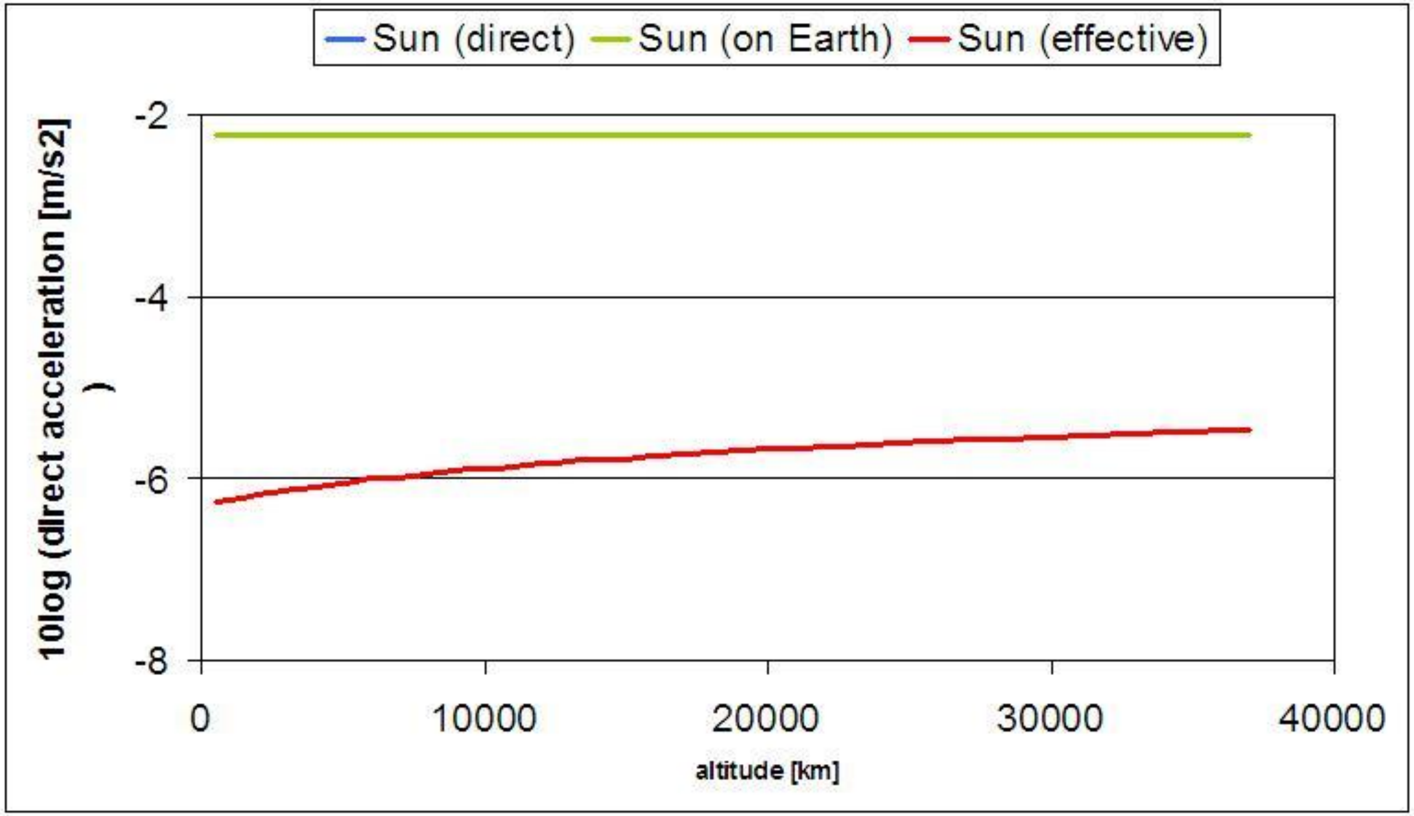




center of
ref. frame

Third-body perturbations (cnt'd)

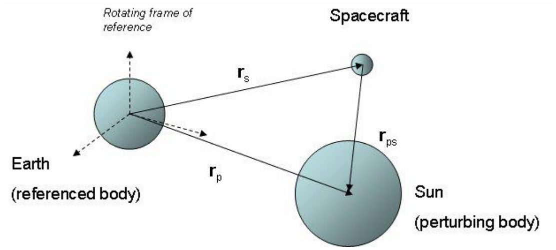




Conclusions:

- influence of Sun as 3rd body increases with distance from Earth
- effective 3rd body acceleration by Sun $O(10^{-6})$ m/s^2

Third-body perturbations (cnt'd)



Question 1:

a) Compute the dimension of the Sphere of Influence of the Earth (when the Sun is considered as the perturbing body). The SoI is given by the following general equation:

$$r_{SoI} = r_{3rd} \left(\frac{M_{main}}{M_{3rd}} \right)^{0.4}$$

b) What is the value of the radial attraction exerted by the Earth at this distance? (if you were unable to make the question a, use a value of 1×10^6 km for this position).

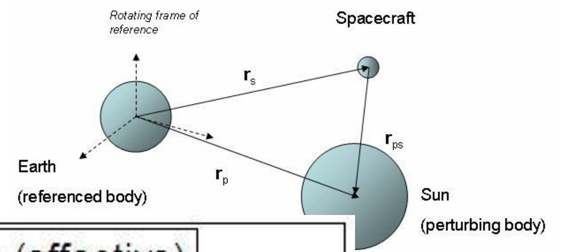
c) What is the effective gravitational acceleration by the Sun at this position? Assume that Earth, Sun and satellite are on a straight line.

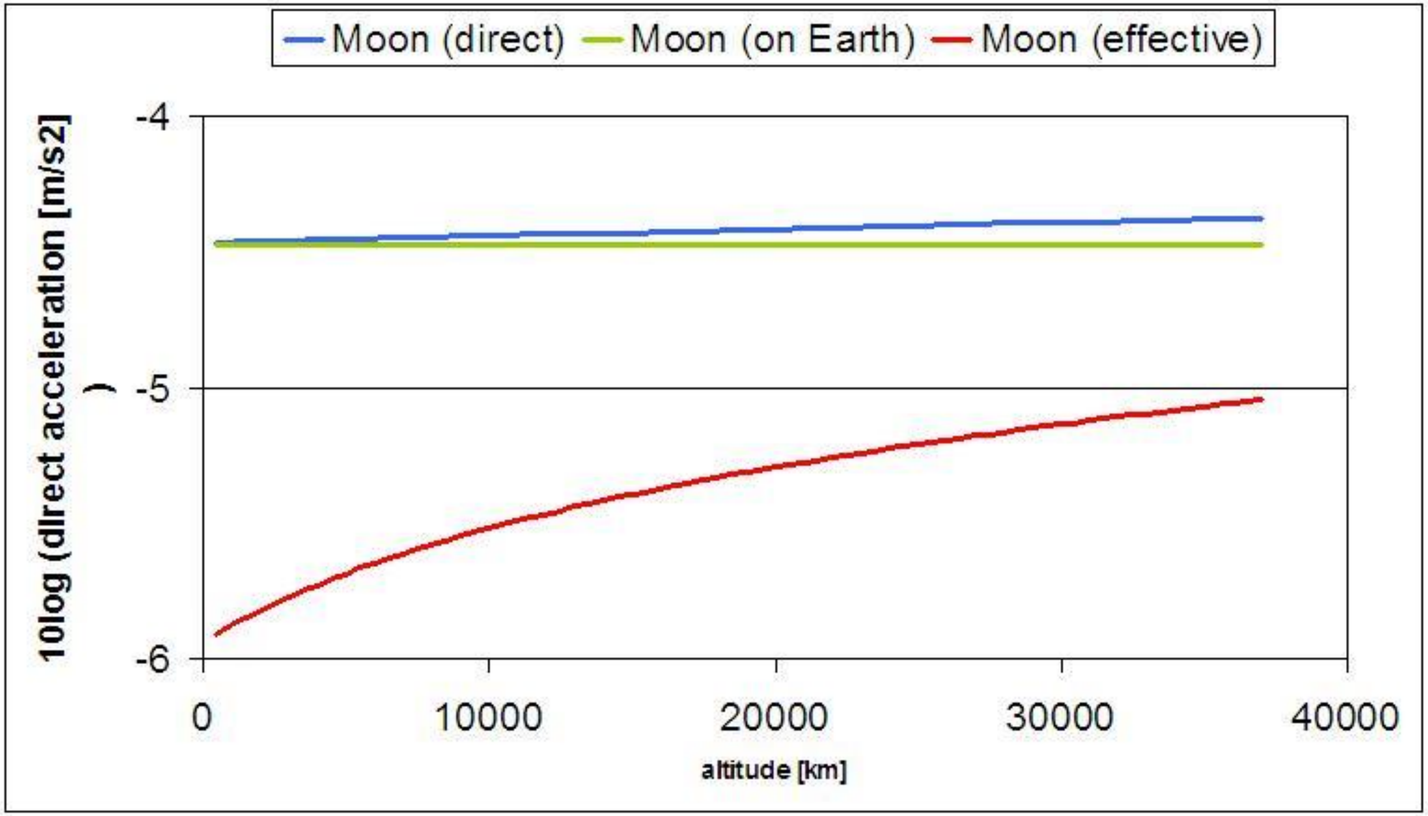
d) What is the relative perturbation of the solar attraction, compared to that of the main attraction of the Earth?

Data: 1 AU = 149.6×10^6 km, $m_{Sun} = 2.0 \times 10^{30}$ kg, $m_{Earth} = 6.0 \times 10^{24}$ kg, $\mu_{Sun} = 1.3271 \times 10^{11}$ km^3/s^2, $\mu_{Earth} = 398600$ km^3/s^2.

Answers: see footnotes below **(BUT TRY YOURSELF FIRST!!)**

Third-body perturbations (cnt'd)

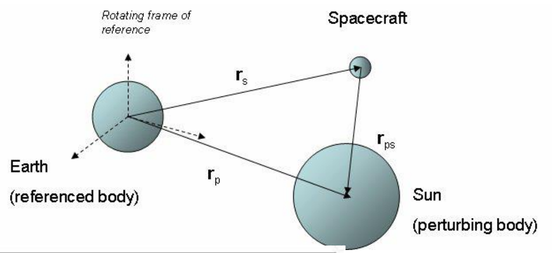


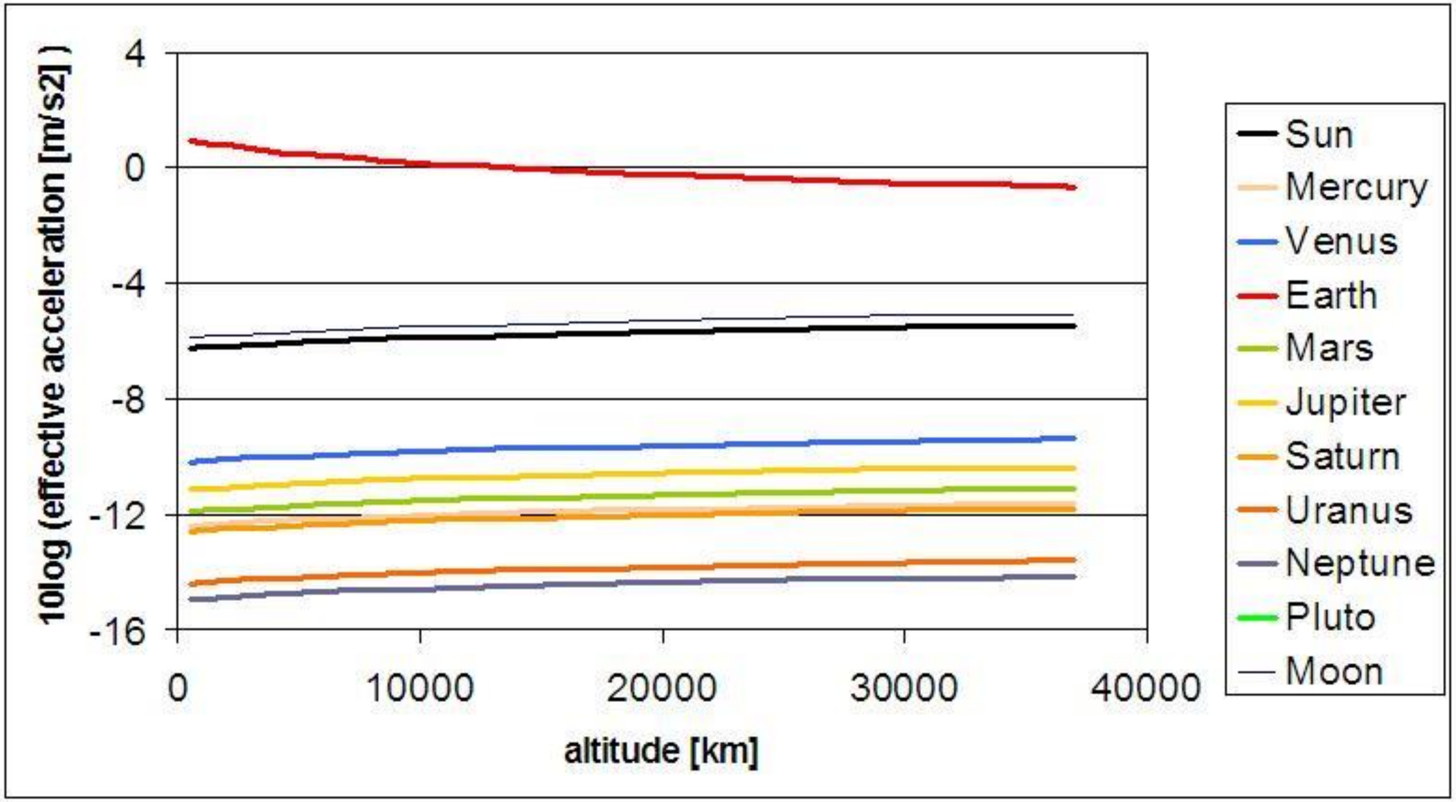


Conclusions:

- influence of Moon as 3rd body increases with distance from Earth
- effective 3rd body acceleration by Moon O(10^{-5}) m/s^2 at GEO

Third-body perturbations (cnt'd)

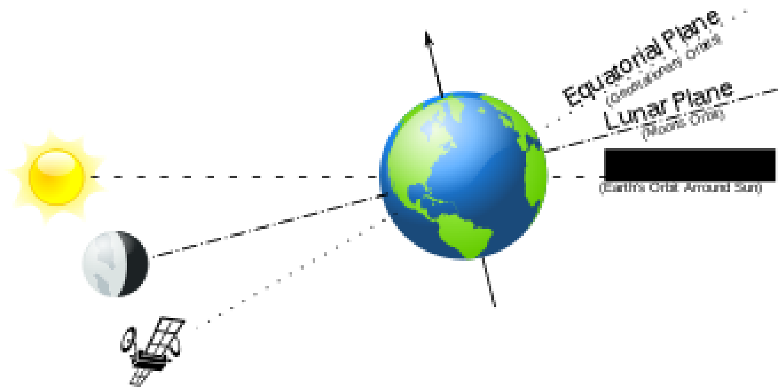


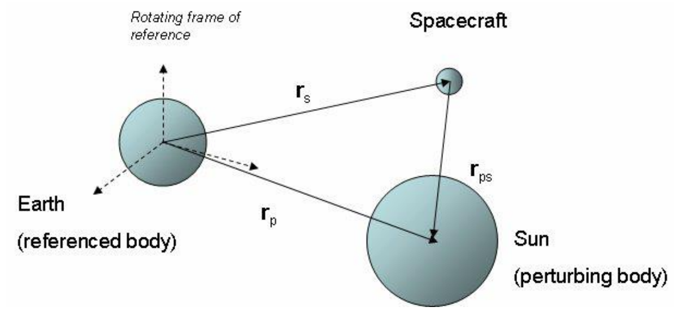


Conclusions:

- Earth dominant (central body; within SoI)
- Moon most important 3rd body; Sun directly after
- effect of planets about 4 orders of magnitude smaller

Third-body perturbations (cnt'd)

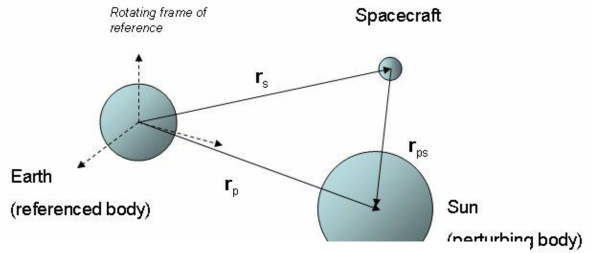




[Wikimedia, 2010]

celestial body	*ΔV*-budget for GEO sat [m/s/yr]
Moon	36.93
Sun	14.45

Third-body perturbations (cnt'd)



Question 2:

Consider a hypothetical planet X with mass 5×10^{25} kg, orbiting the Sun in a circular orbit with radius 3 AU. The orbital plane coincides with the ecliptic (*i.e.*, the orbital plane of the Earth).

a) Make a sketch of the situation when the gravitational attraction of this planet X on satellites around the Earth is largest.

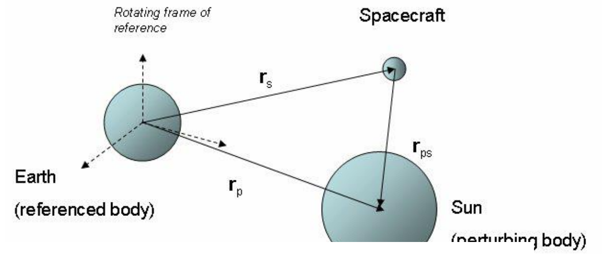
b) Idem for the case when this would be smallest.

c) Compute the maximum and minimum perturbing acceleration due to this planet X, on a geostationary satellite (radius orbit is 42200 km).

Data: G = 6.673×10^{-11} m^3/kg/s^2; μ_{Earth} = 398600 km^3/s^2; 1 AU = 149.6×10^6 km

Answers: see footnotes below **(BUT TRY YOURSELF FIRST!!)**

Third-body perturbations (cnt'd)



Question 3:

The treatment of the motion of a satellite is driven by the fact whether the vehicle is inside the Sphere of Influence (SoI) or not.

a) Describe the concept of the SoI, and give its mathematical (underlying) definition.

b) The dimension of the SoI can be approximated by the equation given below. Consider the Earth-Moon system: μ_{Earth} = 398600 km^3/s^2, μ_{Moon} = 4903 km^3/s^2, average distance Earth-Moon = 384,000 km. What is the radius of the SoI of the Moon?

$$r_{SoI} = r_{3rd} \left(\frac{M_{main}}{M_{3rd}} \right)^{0.4}$$

c) Suppose that the Earth and Moon have equal masses. What would now be the radius of the SoI? What would it be from a physical point of view? Discuss the results.

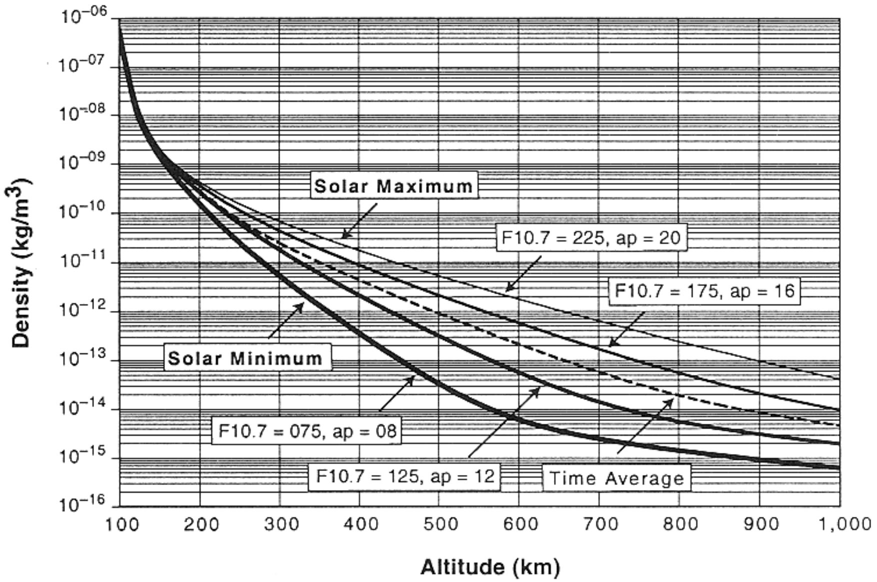
ANSWERS: see footnotes below. **TRY YOURSELF FIRST!**

Perturbations:

- **Irregularities gravity field**
- **Third body**
- **Atmospheric drag**
- **Solar radiation pressure**
- **Thrust**

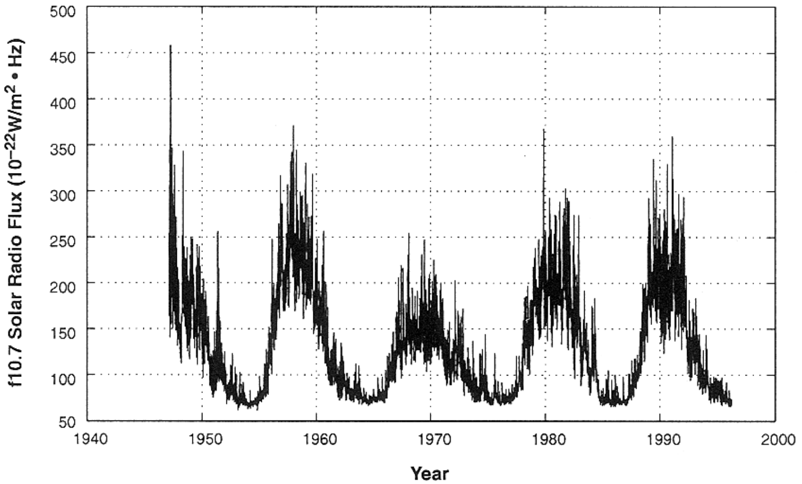
Atmosphere

$$\mathbf{a}_{\mathbf{drag}} = -\frac{C_D\, S}{m}\frac{1}{2}\rho V^2 \frac{\mathbf{V}}{V} \quad ; \qquad \rho = \rho_0 \exp\left(-\Delta h / H\right)$$



[Wertz, 2009]

[Wertz, 2009]



Atmosphere (cnt'd)

after

[Wertz, 2009]:

Altitude [km]	Atmospheric density [kg/m^3]	
	minimum	maximum
200	1.8×10^{-10}	3.5×10^{-10}
300	8.2×10^{-12}	4.0×10^{-11}
400	7.3×10^{-13}	7.6×10^{-12}
500	9.0×10^{-14}	1.8×10^{-12}
600	1.7×10^{-14}	4.9×10^{-13}
700	5.7×10^{-15}	1.5×10^{-13}
800	3.0×10^{-15}	4.4×10^{-14}
900	1.8×10^{-15}	1.9×10^{-14}
1000	1.2×10^{-15}	8.8×10^{-15}
1250	4.7×10^{-16}	2.6×10^{-15}
1500	2.3×10^{-16}	1.2×10^{-15}

Atmosphere (cnt'd)

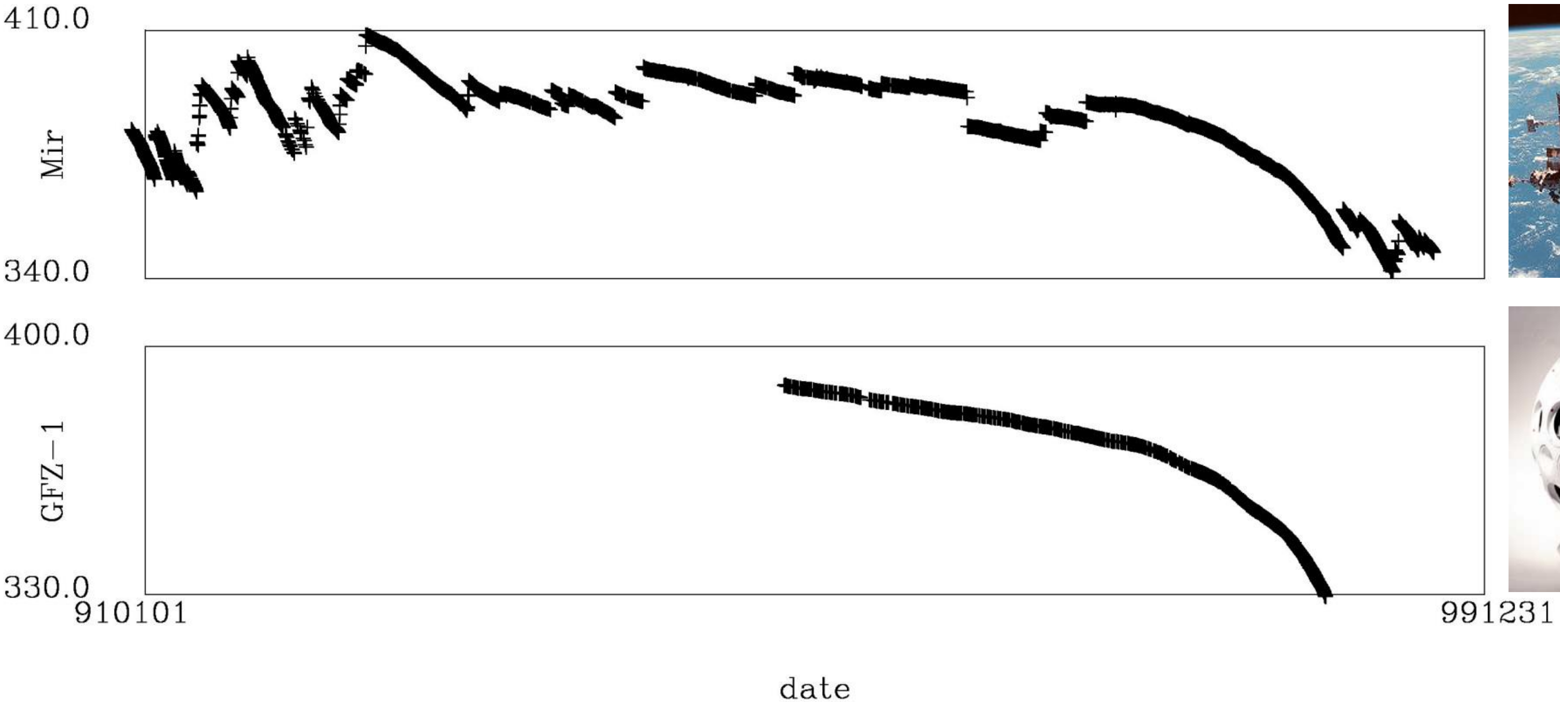
[Wertz, 2009]:

Satellite	Mass (kg)	Shape	Max. XA (m^2)	Min. XA (m^2)	Max. Drag Coef.	Min. Drag Coef.	Max. Ballistic Coef. (kg/m^2)	Min. Ballistic Coef. (kg/m^2)	Type of Mission
Oscar-1	5	box	0.075	0.0584	4	2	42.8	16.7	Comm.
Intercos.-16	550	cylind.	2.7	3.16	2.67	2.1	82.9	76.3	Scientific
Viking	277	octag.	2.25	0.833	4	2.6	128	30.8	Scientific
Explorer-11	37	octag.	0.18	0.07	2.83	2.6	203	72.6	Astronomy
Explorer-17	188.2	sphere	0.621	0.621	2	2	152	152	Scientific
Sp. Teles.	11,000	cylind.*	112	14.3	3.33	4	192	29.5	Astronomy
OSO-7	634	9-sided	1.05	0.5	3.67	2.9	437	165	Solar Physics
OSO-8	1,063	cylind.*	5.99	1.81	3.76	4	147	47.2	Solar Physics
Pegasus-3	10,500	cylind.*	264	14.5	3.3	4	181	12.1	Scientific
Landsat-1	891	cylind.*	10.4	1.81	3.4	4	123	25.2	Rem. Sens.
ERS-1	2,160	box*	45.1	4	4	4	135	12.0	Rem. Sens.
LDEF-1	9,695	12-face	39	14.3	2.67	4	169	93.1	Environment
HEAO-2	3,150	hexag.	13.9	4.52	2.83	4	174	80.1	Astronomy
Vanguard-2	9.39	sphere	0.2	0.2	2	2	23.5	23.5	Scientific
SkyLab	76,136	cylind.*	462	46.4	3.5	4	410	47.1	Scientific
Echo-1	75.3	sphere	731	731	2	2	0.515	0.515	Comm.
Extrema							437	0.515	

*With solar arrays

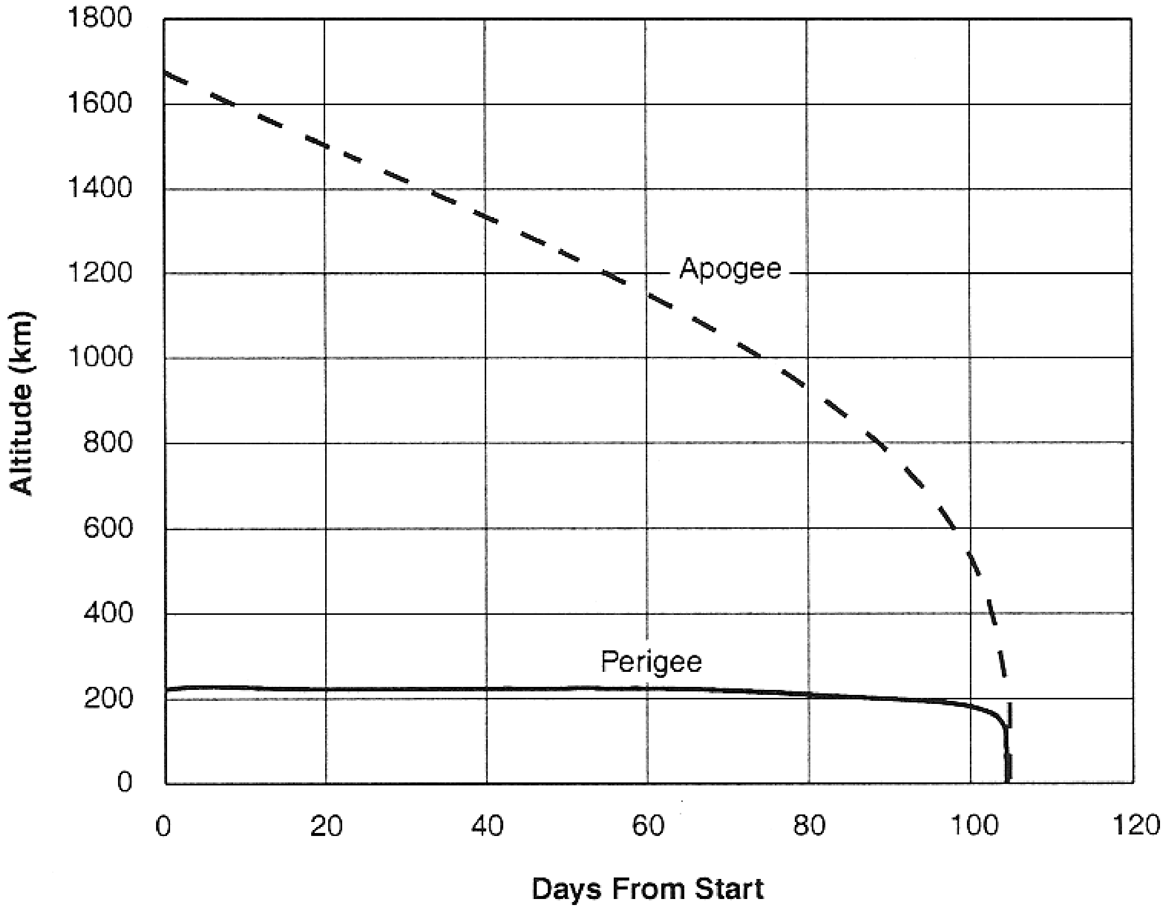
Atmosphere (cnt'd)

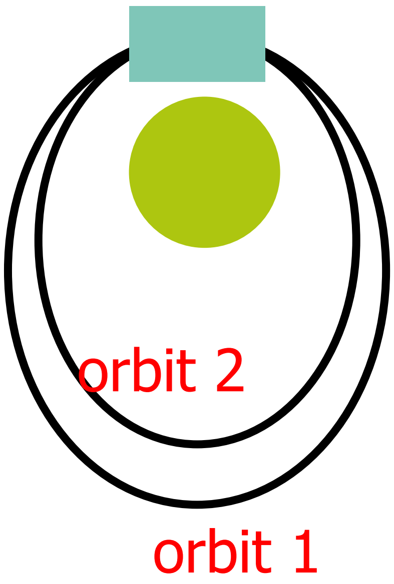
altitudes of Mir and GFZ-1:



Atmosphere (cnt'd)

[Wertz, 2009]:





Atmosphere (cnt'd)

circular orbits:

$$\Delta a_{2\pi} = -2\pi \, (C_D A/m) \, \rho \, a^2 \qquad \text{[m]}$$

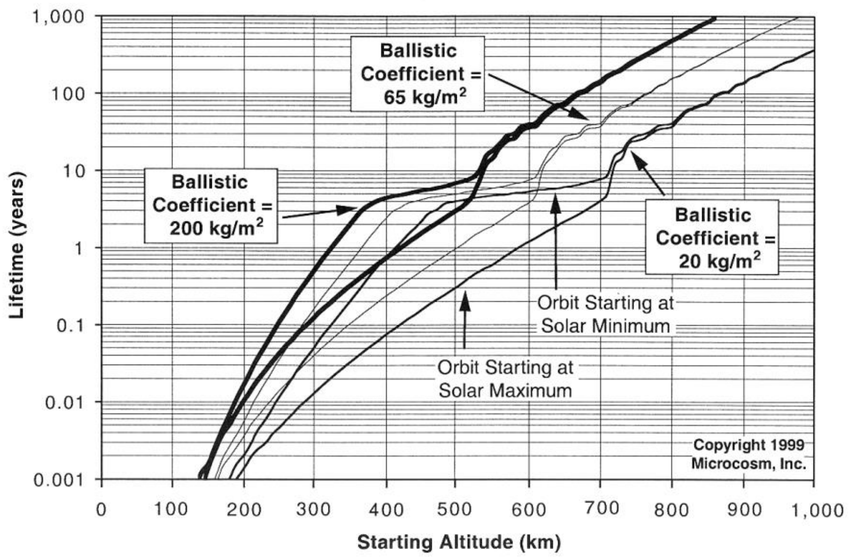
$$\Delta T_{2\pi} = -6\pi^2 \, (C_D A/m) \, \rho \, a^2 / V \qquad \text{[s]}$$

$$\Delta V_{2\pi} = \pi \, (C_D A/m) \, \rho \, a \, V \qquad \text{[m/s]}$$

$$\Delta e_{2\pi} = 0 \qquad \text{[-]}$$

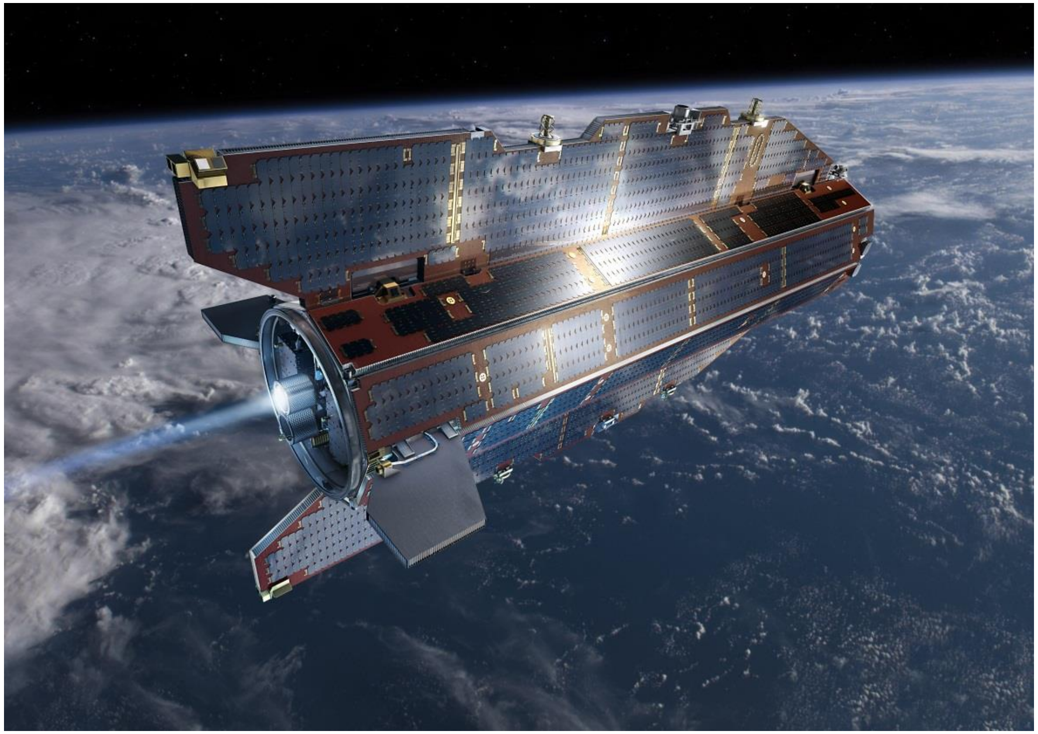
$$L = -H / \Delta a_{2\pi} \qquad \text{[rev]}$$

[Wertz, 2009]:



Atmosphere (cnt'd)

GOCE:



[ESA, 2010]

- **launch date: March 17, 2009**
- **altitude: 250 km**
- **phase solar cycle?**
- **…..**

Perturbations:

- **Irregularities gravity field**
- **Third body**
- **Atmospheric drag**
- **Solar radiation pressure**
- **Thrust**

Solar radiation pressure

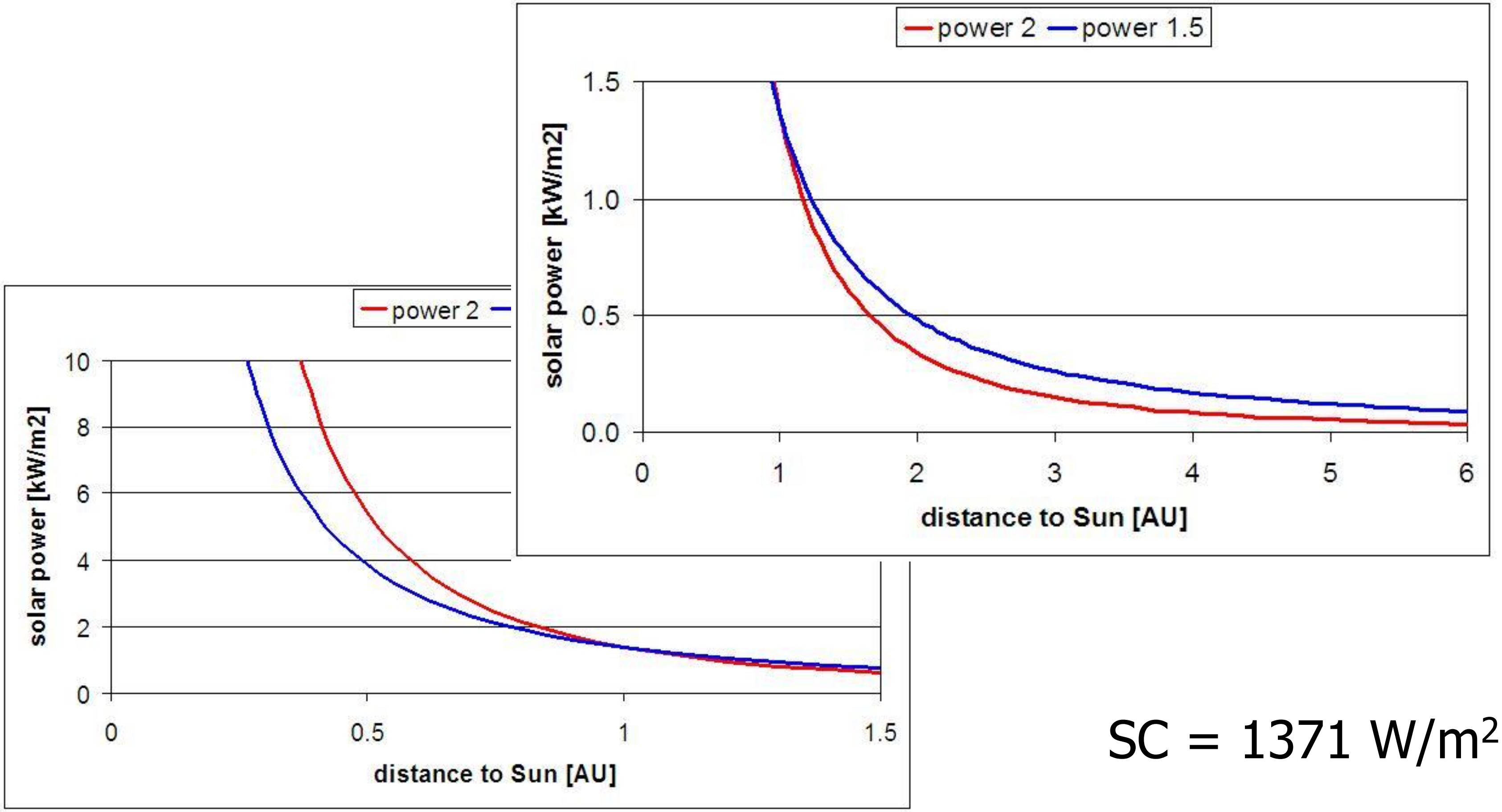
- amount of energy emitted by Sun at 1 AU distance: ≈1371 W/m^2
- value hardly dependent on solar activity ➔ "Solar Constant" (SC)
- Solar radiation pressure [N/m^2]: SC / c
- energy reduces with distance w.r.t. Sun: energy(r) = SC/r^2

$$\mathbf{a}_{\text{rad}} = (1+\rho)\frac{1}{\left(\left\|\mathbf{r}_{\text{Sun-sat/AU}}\right\|\right)^2}\frac{SC}{c}\frac{A}{m}\frac{\mathbf{r}_{\text{Sun-sat}}}{\left\|\mathbf{r}_{\text{Sun-sat}}\right\|} \approx 4.5\times10^{-6}\frac{A}{m}$$

reflection coefficient

unit vector Sun → sat

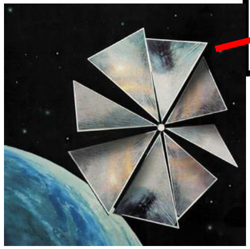
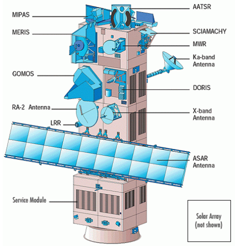
Solar radiation pressure (cnt'd)



SC = 1371 W/m^2

Solar radiation pressure (cnt'd)



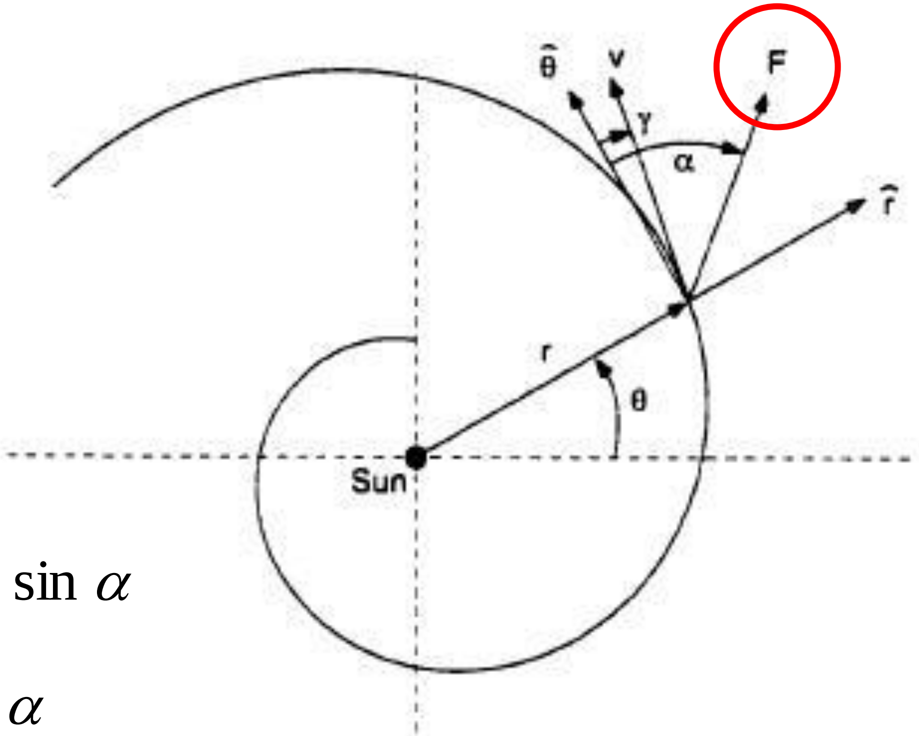




satellite	**a_{rad} [m/s²]**
ISS (100-1000 m²; 500 ton)	$9 - 90 \times 10^{-10}$
ENVISAT	$8.3 - 83 \times 10^{-9}$
LAGEOS	3×10^{-9}
Echo-1	4.4×10^{-5}
solar sail (100x100 m; 300 kg)	1.5×10^{-4}

Perturbations:

- **Irregularities gravity field**
- **Third body**
- **Atmospheric drag**
- **Solar radiation pressure**
- **Thrust**

Thrust



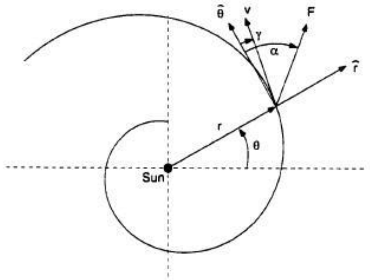
[Petropoulos 2001]

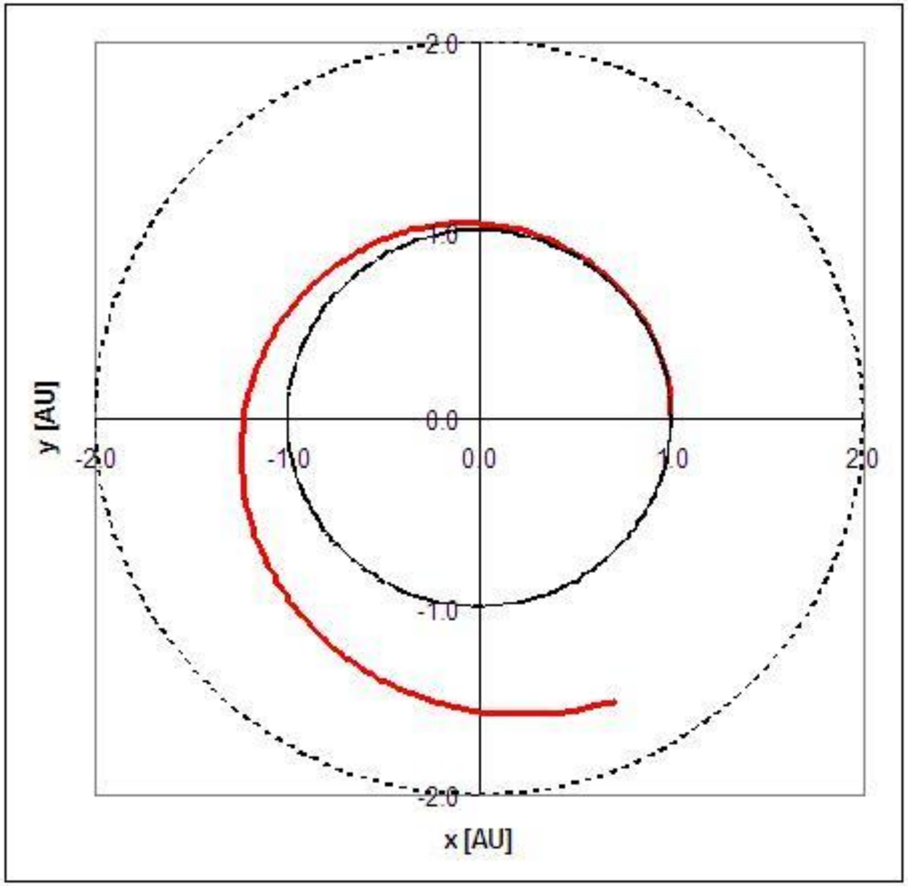
$$\ddot{r} - r\dot{\theta}^2 + \mu / r^2 = a_T \sin\alpha$$

$$\frac{1}{r}\frac{d}{dt}\left(r^2\,\dot{\theta}\right) = a_T \cos\alpha$$

$$(or:\quad \ddot{\theta}\, r + 2\dot{\theta}\,\dot{r} = a_T \cos\alpha\ )$$

Thrust (cnt'd)

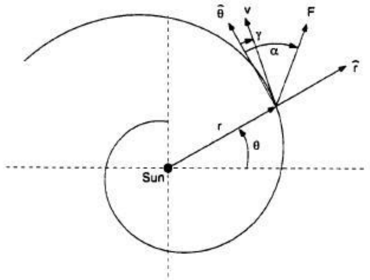


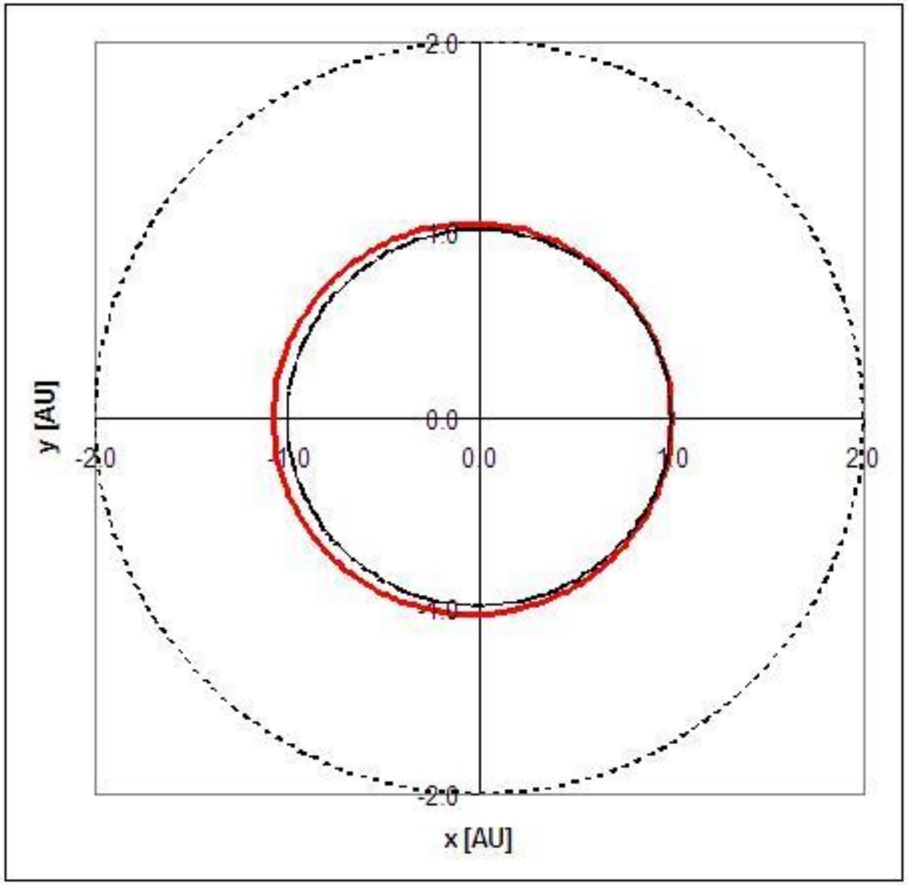


Example 1:

acceleration of 0.190 mm/s² in along-track direction (*i.e.*, α=0°)

Thrust (cnt'd)

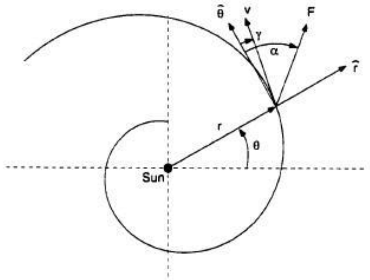


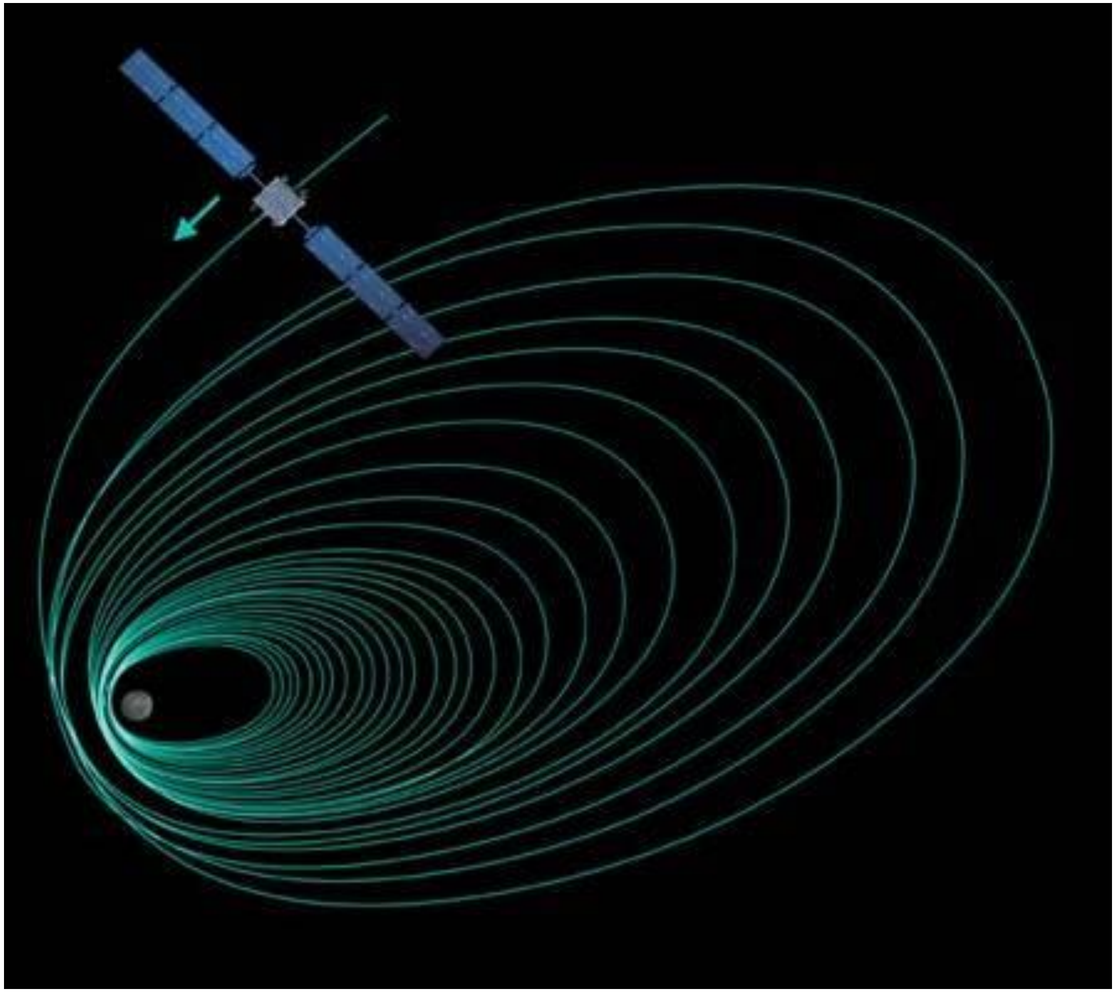


Example 2:

acceleration of 0.190 mm/s^2 in radial direction (*i.e.*, α=90°)

Thrust (cnt'd)





Example 3:

orbit of SMART-1
[ESA, 2010]

Thrust (cnt'd)

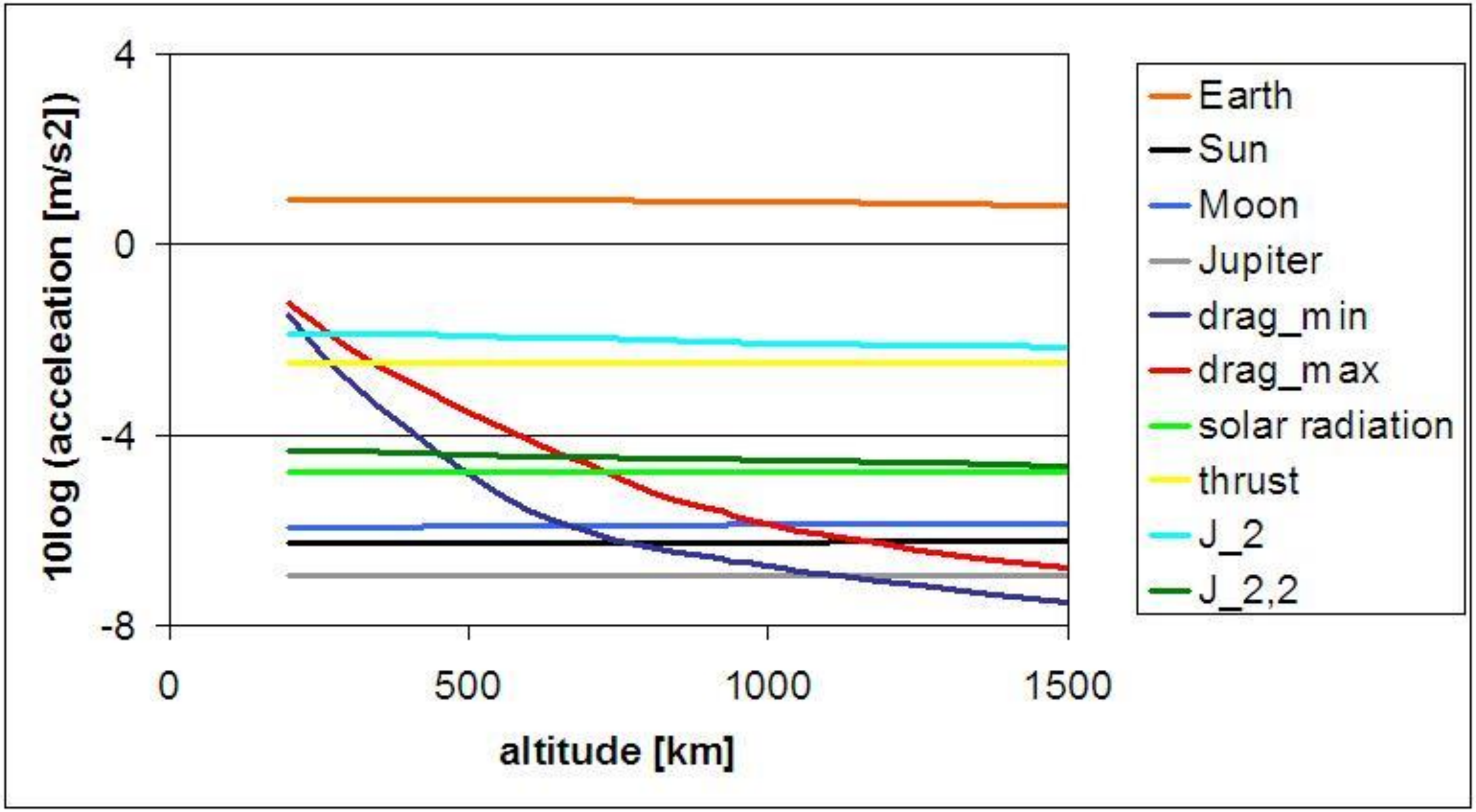
High thrust:

- can compete against central gravity (*i.e.*, launch)
- instantaneous velocity changes (orbits around Earth + interplanetary orbits)

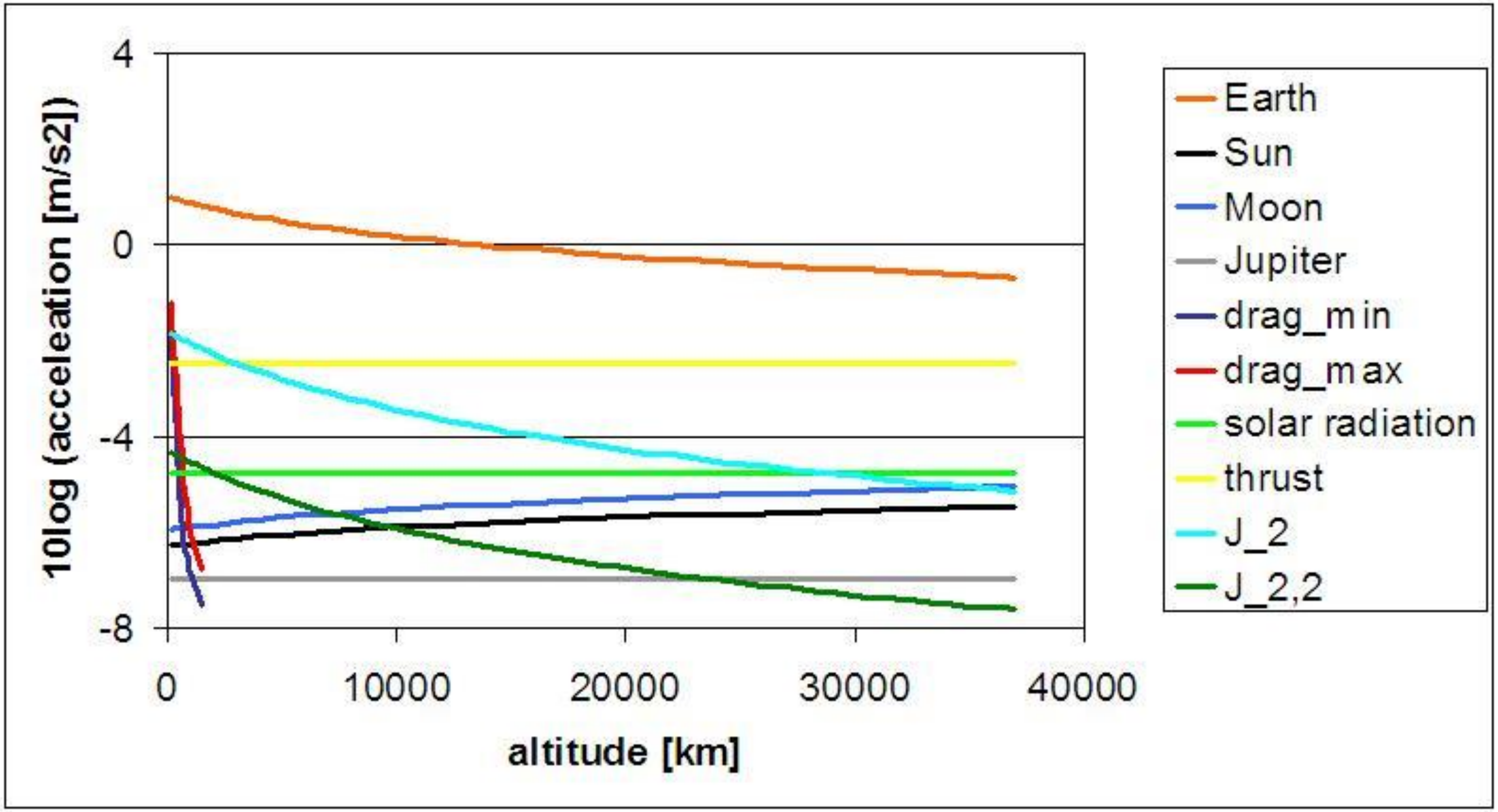
Low thrust:

- attractive since high I_{sp}
- primary propulsion for interplanetary missions
- station-keeping
- since 2012: transfer LEO → GEO

Summary: Low Earth Orbit (1)



Summary: Low Earth Orbit (2)

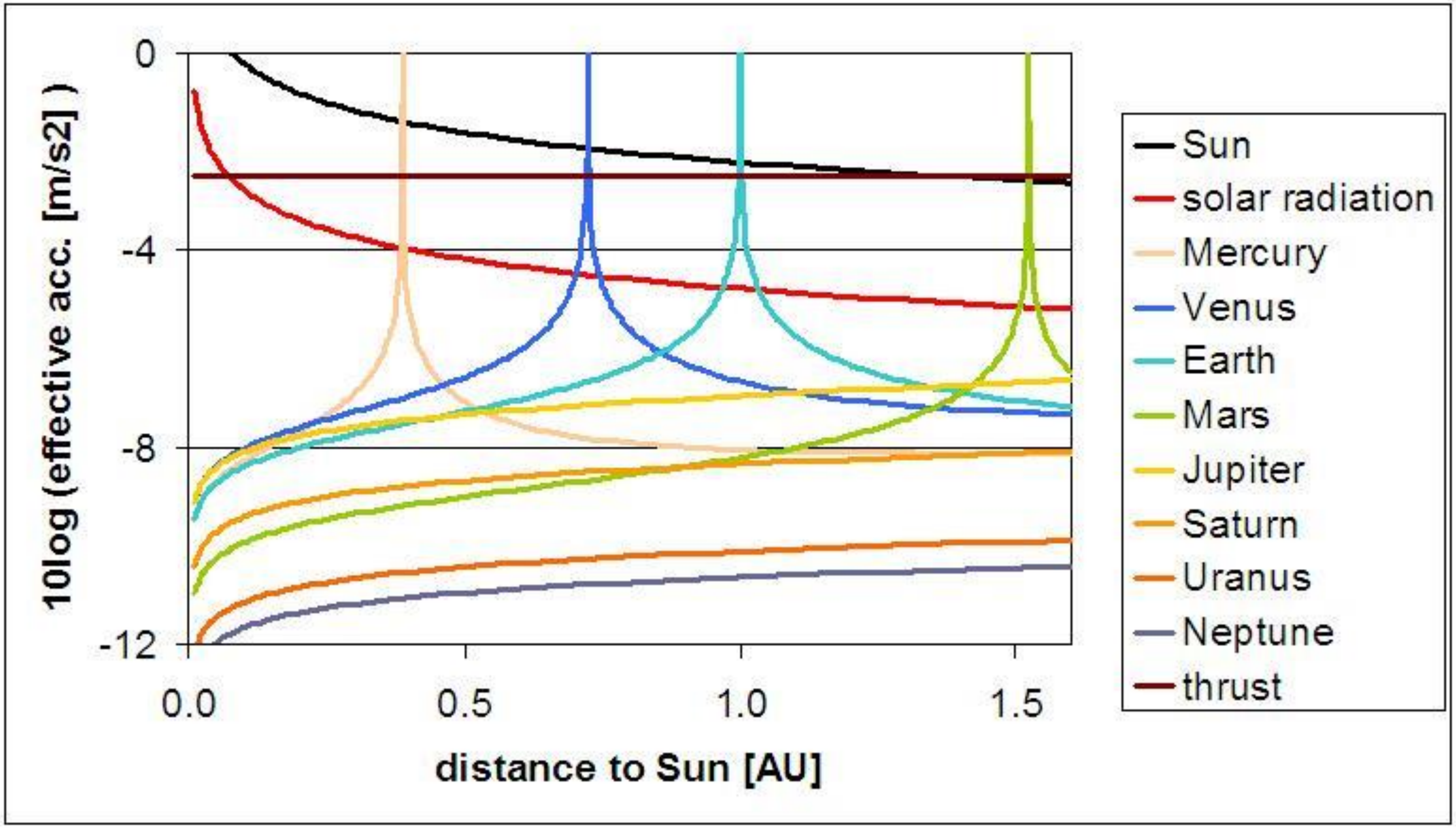


Summary: Low Earth Orbit (3)

Conclusions near-Earth orbits:

- J_2 acceleration is dominant perturbation for all LEO
- low thrust already important
- atmospheric drag dominant perturbation at very low altitudes
- Solar, lunar and $J_{2,2}$ accelerations very small, but build up for GEO
- Kepler orbit very good 1^{st}-order approximation (2-4 orders of magnitude Δ)

Summary: interplanetary orbit (1)



Summary: interplanetary orbit (2)

Conclusions interplanetary orbits:

- low thrust important
- Solar radiation can be important
- Kepler orbit very good 1st-order approximation (4-6 orders of magnitude Δ)

Summary perturbing forces

Question 4:

a) Mention the 5 different categories of perturbing forces that may act on an arbitrary vehicle. Describe each category briefly (about 5 lines per item).

b) Give a brief description of the (relative) importance of these categories.

Summary perturbing forces

Question 5:

a) Mention the 5 different categories of perturbing forces that may act on an arbitrary vehicle.

b) Compute the magnitude of the main gravitational force exerted by the Earth's gravity field.

c) Compute the magnitude of the various perturbing sources.

Data: μ_{Earth} = 398600 km^3/s^2; μ_{Moon} = 4903 km^3/s^2; μ_{Sun} = 1.327 × 10^{11} km^3/s^2; R_e = 6378 km; h_{GOCE} = 250 km; $\rho_{atm,avg}$ = 6.2 × 10^{-11} kg/m^3; AU = 149.6 × 10^6 km; SC = 1371 W/m^2; c = 3 × 10^5 km/s; m_{GOCE} = 1050 kg; S_{GOCE} = 1.0 m^2; $C_{D,GOCE}$ = 2.2; $C_{R,GOCE}$ = 1.2; T_{GOCE} = 10 mN; J_2 = 1082 × 10^{-6}; $J_{2,2}$ = 1.816 × 10^{-6} ; $\lambda_{2,2}$ = -14.9°

Answers: see footnotes below **(BUT TRY YOURSELF FIRST!!)**