Conventional methods for satellite launch

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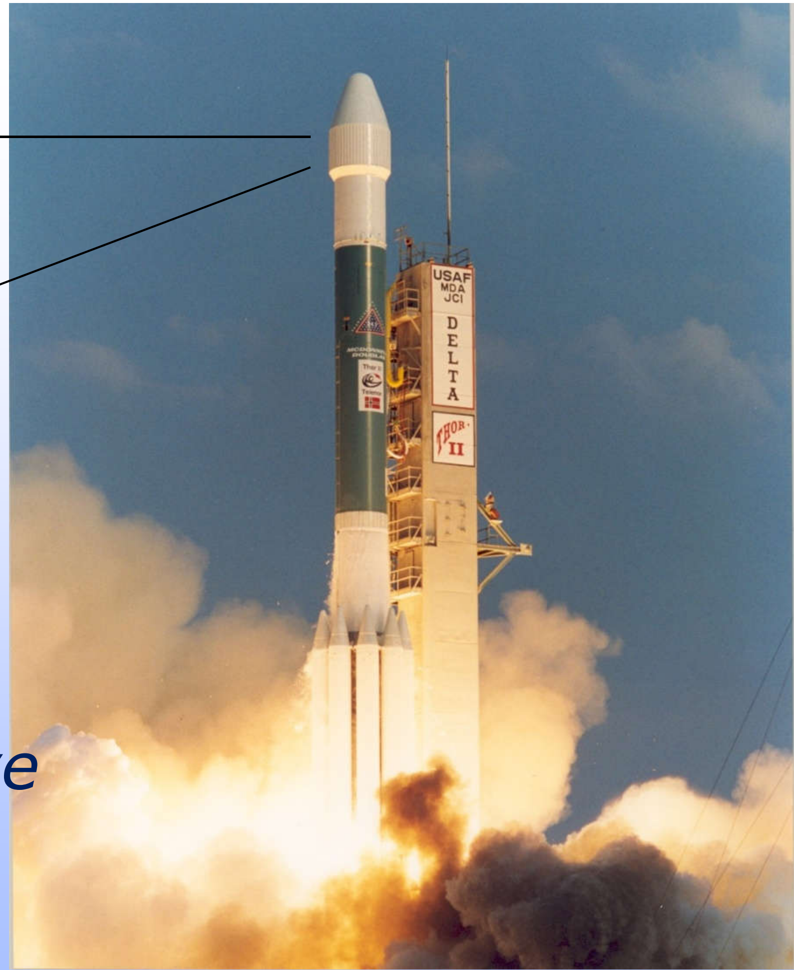
approx. 2000 kg

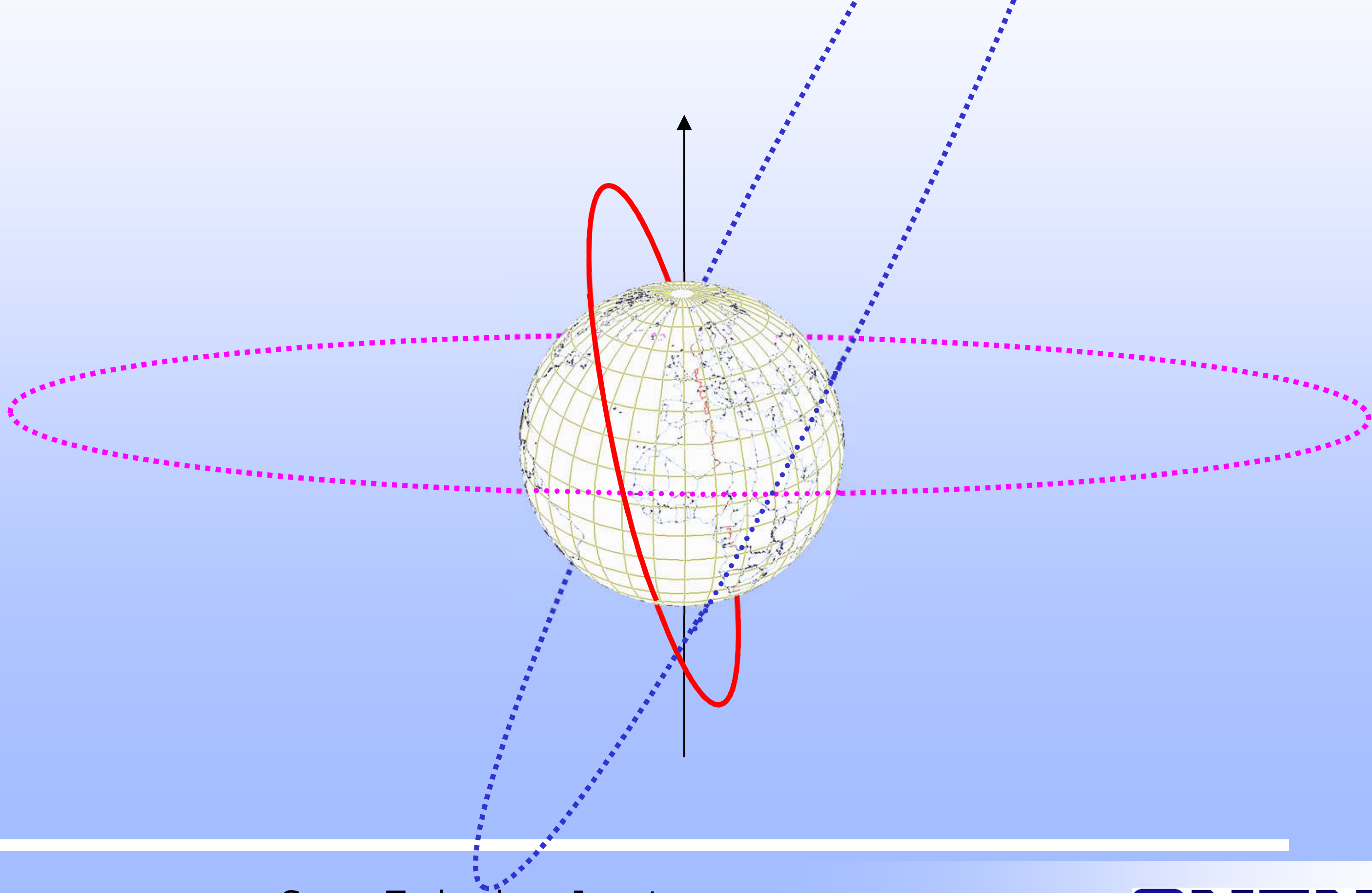


approx. 200 000 kg

«Many more millionaires have gone bankrupt trying to develop rockets than satellites. They have overlooked the fact that the operative word in "controlled explosion" is controlled»
(P. Berlin: Bluffers guide to rocket science)

GEO/MEO/LEO/HEO

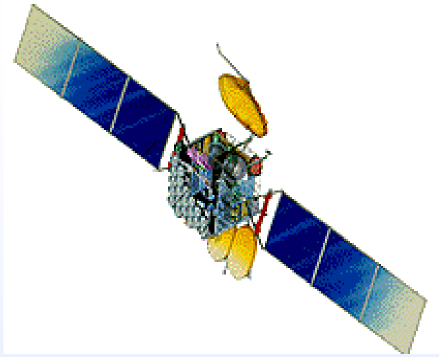
GEO - Geostationary Orbit (36 000 km) - Communications, Earth obs.
MEO - Medium Earth Orbits (20 000 km km) - Navigation (GPS)
LEO - Low Earth Orbits (800 km) - Mobile Com.(Iridium), Earth obs.
HEO - Highly Inclined Orbits (800 x 42 000 km) – Com., Earth obs.



A satellite project consists of 4 parts: 1) spacecraft 2) launcher 3) ground control 4) insurance

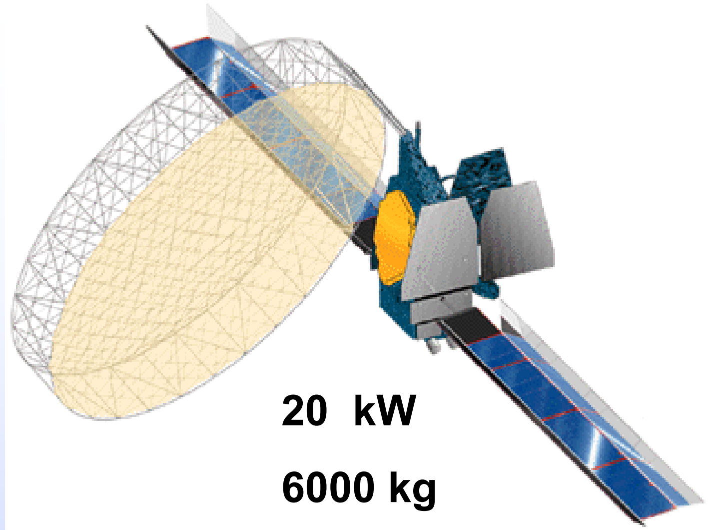
1.7 kW

1500 kg



4 - 7 kW

3800 kg



20 kW

6000 kg

Delta II

120-150 MUSD

Atlas 2 AS

220-260 MUSD

Proton

Ariane 5

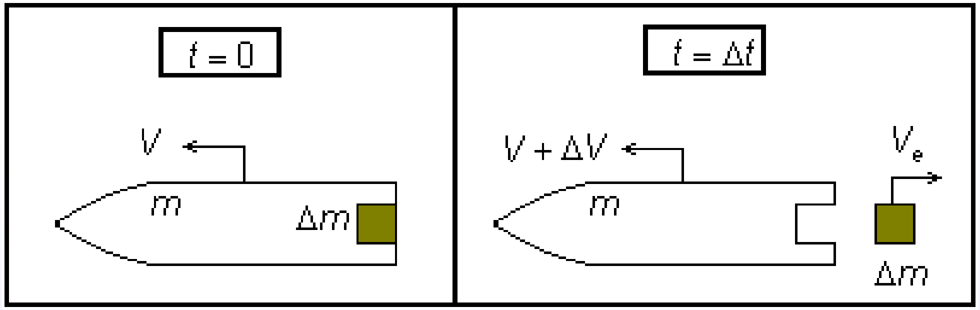
Atlas 5

275 – 350 MUSD

Derivation of the Rocket equation (1)

Newton's second law of motion relates external forces F_i the change in linear momentum of the whole system (including rocket and exhaust) as follows:

$$\sum F_i = \lim_{\Delta t \to 0} \frac{P_2 - P_1}{\Delta t}$$



P1 is the momentum of the rocket at time t=0
P_2 is the momentum of the rocket and exhausted mass at time t=Δt

$$P_1 = (m + \Delta m)V$$

$$P_2 = m(V + \Delta V) + \Delta m V_e$$

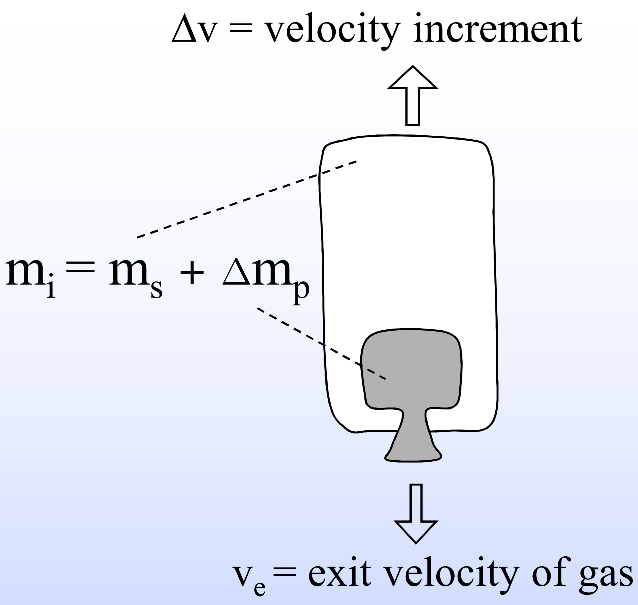
$$V_e = V - v_e$$

The rocket is moving at velocity V with respect to the Earth, the propellant is moving at speed $-v_e$ *relative to the rocket*, so the velocity of the propellant relative to the Earth is $V - v_e$

$$P_2 - P_1 = m\Delta V - v_e \Delta m$$

Derivation of the Rocket equation (2)

$$\Sigma F_i = m\frac{\Delta V}{\Delta t} + v_e\frac{\Delta m}{\Delta t} = 0$$ (no external forces)



Using $\Delta m = -dm$, since ejecting a positive Δm results in a decrease in mass, the equation will be

$$dV = -v_e\frac{dm}{m}$$

Integrating (assuming v_e constant) and inserting the mass before ignition, $m_{initial}$ and the mass after burning, $m_{final}=m_{initial}-\Delta m_{prop}$

The velocity increment Δv:

$$\Delta V = V_{final} - V_{initial} = -v_e\int_{m_i}^{m_i-\Delta m_p}\frac{dm}{m} = v_e\ln\left(\frac{m_i}{m_i-\Delta m_p}\right)$$

Specific impulse (Isp) of the rocket motor

- Unit (in seconds) as a measure of the propellant efficiency
- Specific impulse is defined as thrust per spent kilogram per second

$$I_{sp} = \frac{F \cdot \Delta t}{g \cdot \Delta m} = \frac{F}{\frac{\Delta m}{\Delta t} g} = \frac{v_e}{g} \ [\mathrm{sec}]$$

F = thrust (Newton)= $v_e \dfrac{\Delta m}{\Delta t}$

$\dfrac{\Delta m}{\Delta t}$ = propellant mass flow (kg/s)

g = acceleration of gravity = 9.81 m/s^2

Propellant consumption

The rocket equation may also be written:

$$\Delta m_p = m_i \cdot \left(1 - e^{-\frac{\Delta V}{I_{sp} \cdot g}} \right)$$

Given the motor's Isp in sec, we can calculate the propellant mass Δm in kg needed to achieve a certain velocity increment Δv in m/s.

$$\Delta V = I_{sp} \cdot g \cdot \ln \left(\frac{1}{1 - \frac{\Delta m_p}{m_i}} \right)$$

$$m_i = m_s + \Delta m_p$$

Maximize

Launch Vehicles (general comments)

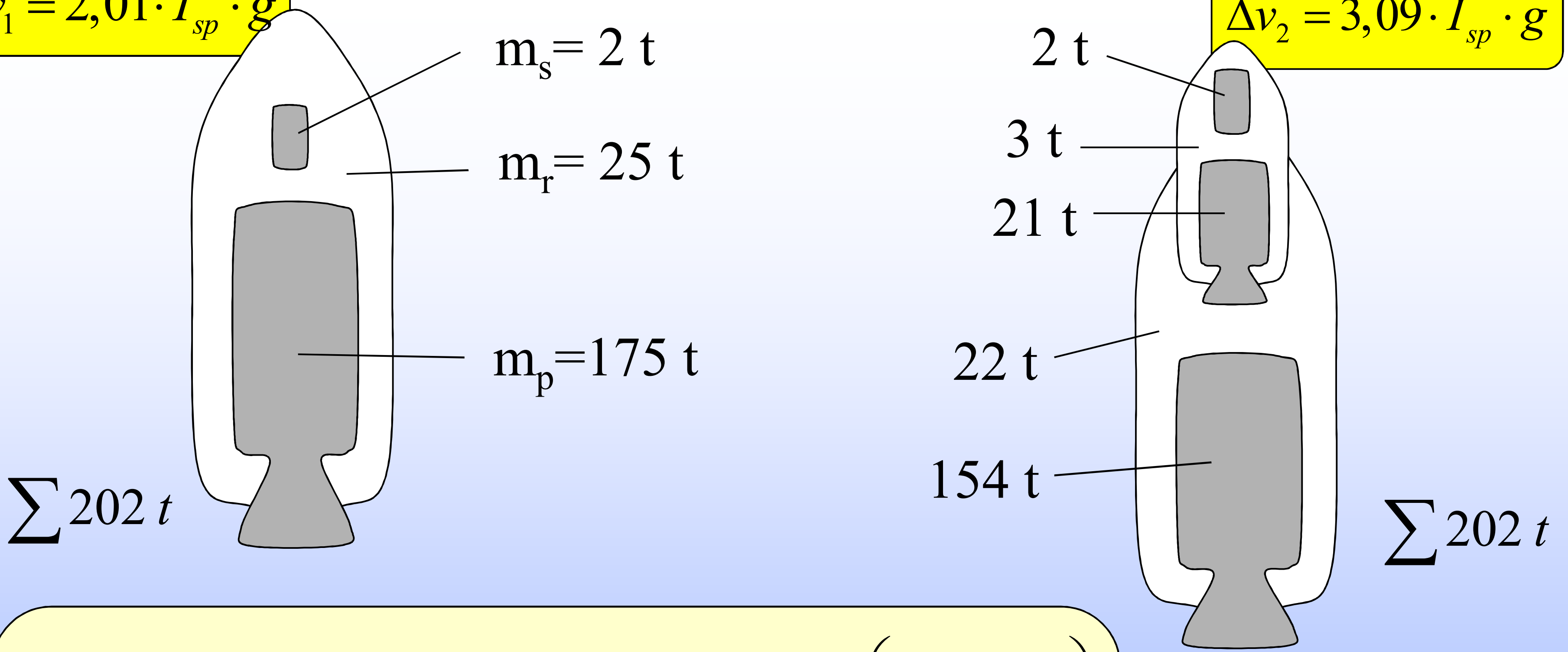
Must use a multistage launch vehicle because:

- Technology for a single stage rocket does not exist
- Effective mass fraction of propellant is increased by dropping dry structural mass during launch
- Lower stages accelerate the last stage, which carries the payload

Ex.: 1-stage compared to a 2-stage rocket

$$\Delta v_1 = 2{,}01 \cdot I_{sp} \cdot g$$

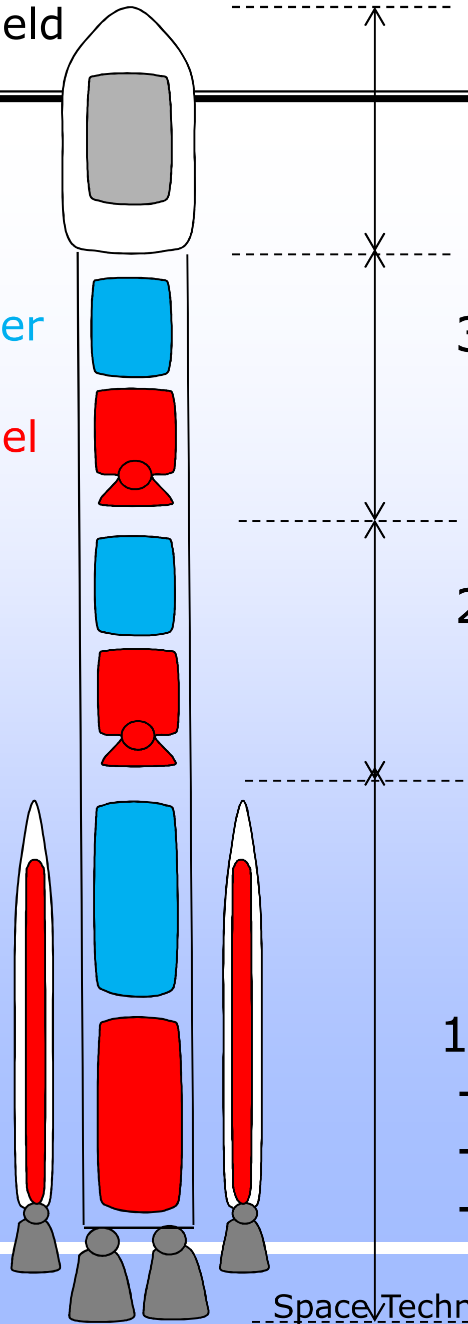
$$\Delta v_2 = 3{,}09 \cdot I_{sp} \cdot g$$



$$\Delta v = I_{sp} \cdot g \cdot \ln\left(\frac{m_{tot}}{m_{tot} - m_p}\right) = I_{sp} \cdot g \cdot \ln\left(\frac{1}{1 - \dfrac{m_p}{m_{tot}}}\right)$$

$$\frac{\Delta v_2}{\Delta v_1} \approx 1{,}5$$

Multistage launch vehicle



Payload section (satellite)

3rd stage
- cryogenic $H_2 + O_2$
- high Isp (>400 s)

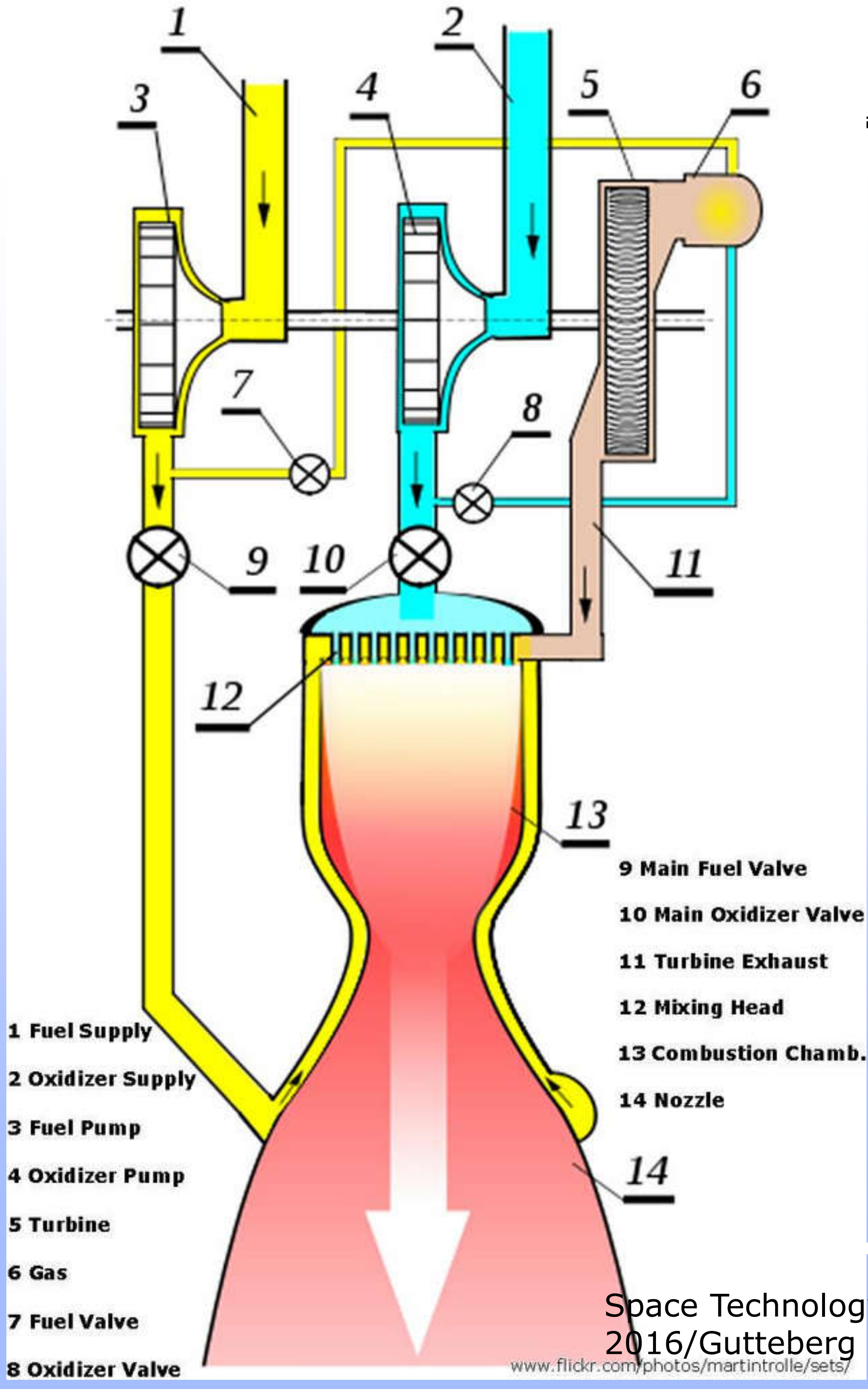
2rd stage
- UDMH+ N_2O_4

UDMH = Unsymmetrical dimthyl hydrazine $(CH_3)_2NNH_2$
N_2O_4 = Nitrogen tetroxide

- 2 – 4 rocket stages
- 2 – 9 strap-on boosters
- Propellent 80% of launch mass

1st stage
- kerosene + O_2
- Isp ~ 250 s
- High thrust (1.5 – 3 times liftoff weight)

Liquid rocket engine



The propulsion is created by two elements:

Oxidizer and fuel, kept in separate tanks. The oxidizer and fuel are fed through pumps down to the combustion chamber under pressure.

The fuel and oxidizer pumps are driven by a turbine, which runs on the same propellant as the rocket.

The oxidizer and fuel ignite spontaneously (hypergolic) creating hot gasses that are squeezed out by a nozzle creating propulsion for the rocket.

Data for some typical rocket fuels

Propellant type	**I_{sp} (s)**	**Comments**
Solid	250-290	Not restartable, easy to store
Kerosene+O_2	300-350	Cryogenic, difficult to store, hazardous
UDMH+N_2O_4	300-350	Difficult to store, hazardous
H_2+O_2	440-460	Cryogenic, very hazardous and diffult to store

Solid = Aluminium powder + Ammonium Perchlorate (NH4ClO4) + synthetic rubber

UDMH = Unsymmetrical dimthyl hydrazine $(CH_3)_2NNH_2$

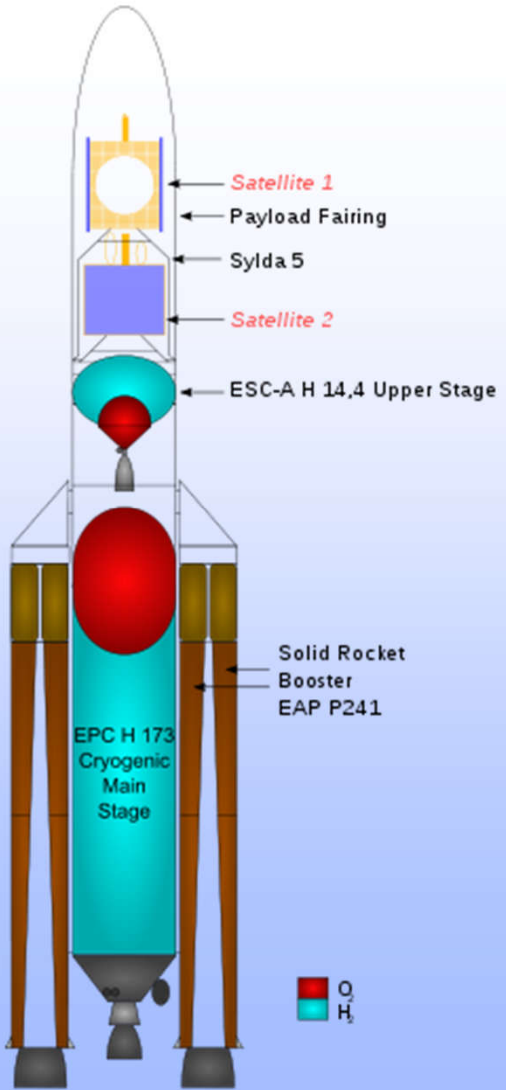
N_2O_4 = Nitrogen tetroxide

Remember the key word: "<u>controlled</u> explosion"

Ariane 5-ECA Launch Vehicle

Height: 53 m
Diameter: 5.4 m
Launch mass :777 000 kg
Mass to LEO: 21 000 kg
Mass to GTO: 10 500 kg

Ariane 5 ECA



Upper Stage

- 14.9 t of LOX + LH_2
- Burn time: ~945 sec (~16 min)
- Thrust: 6.8 t (67 kN)

Main Stage

- 170 t of LOX + LH2
- Burn time: ~540 sec (9 min)
- Fuel consumption:

Liquid hydrogen: ~48 kg/s (720 liters/sec)

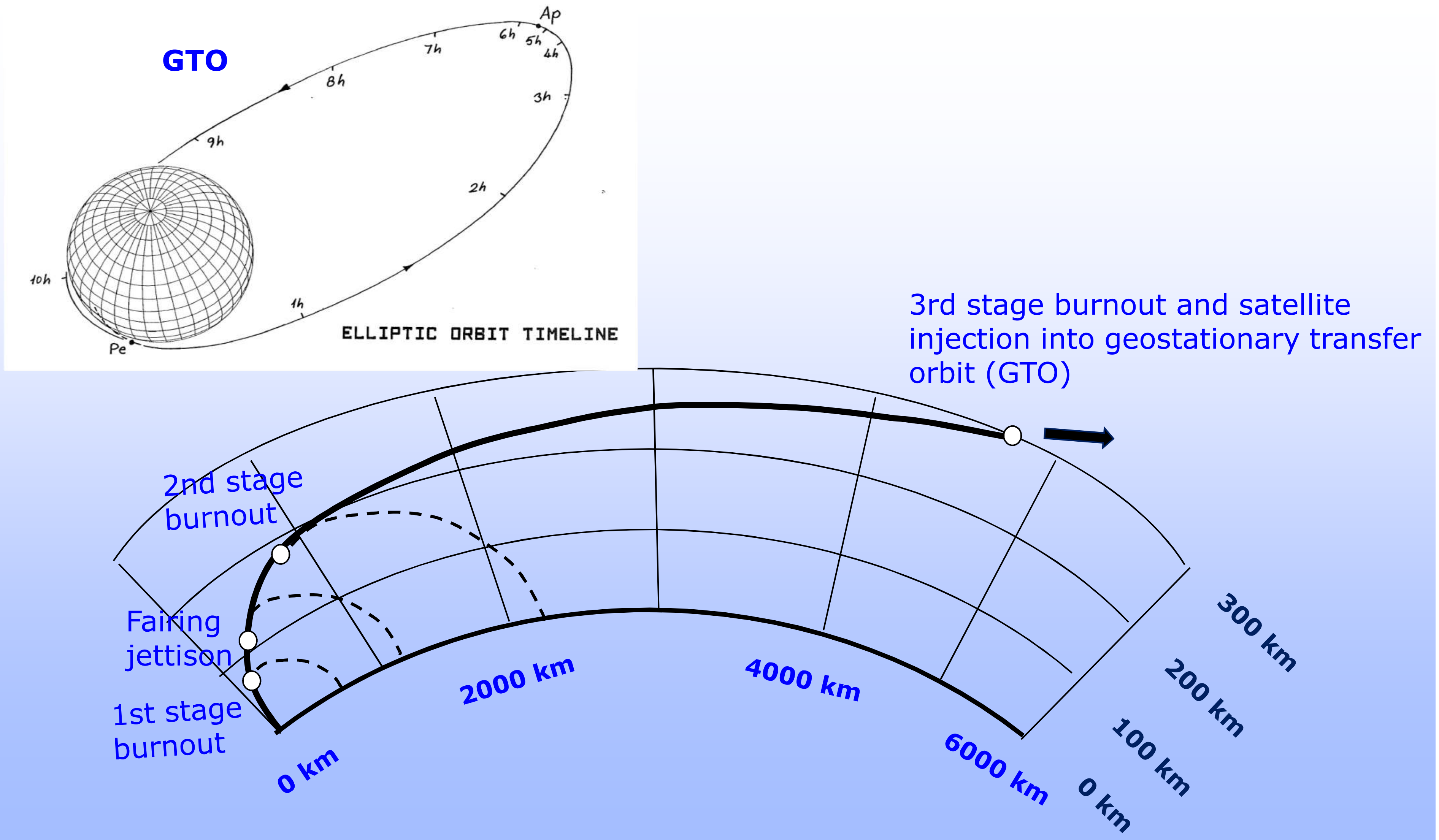
Liquid oxygen:~228 kg/s (220 liters/sec)

- Thrust: 142 t (1390 kN)

Two Solid Fuel Boosters

- 240 t solid fuel per booster
- Burn time: 130 sec. (2 min. 10 sec.)
- Fuel consumption: 1800 kg/sek. per booster
- Thrust: 714 t pr booster (7000 kN)

Launch vehicle trajectory



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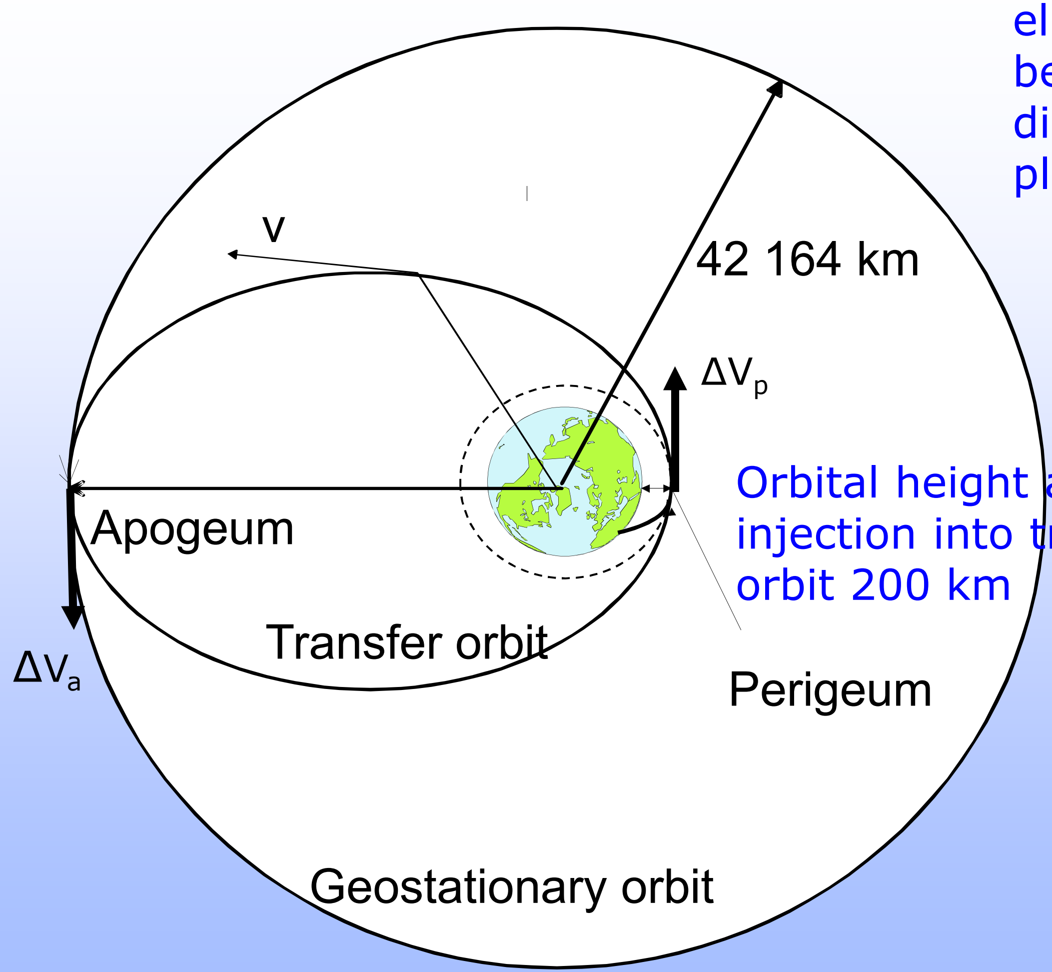
Transfer to geostationary orbit (Hohmann-transfer)



Orbital height at injection into transfer orbit 200 km

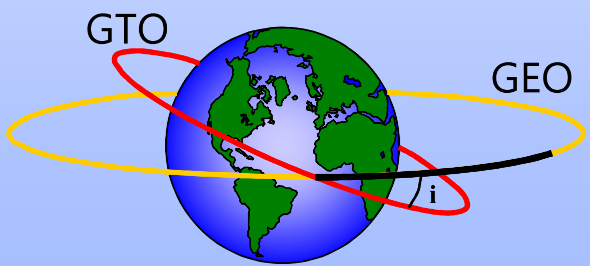
The Hohmann transfer orbit is an elliptical orbit used to transfer between two circular orbits of different altitudes, in the same plane.

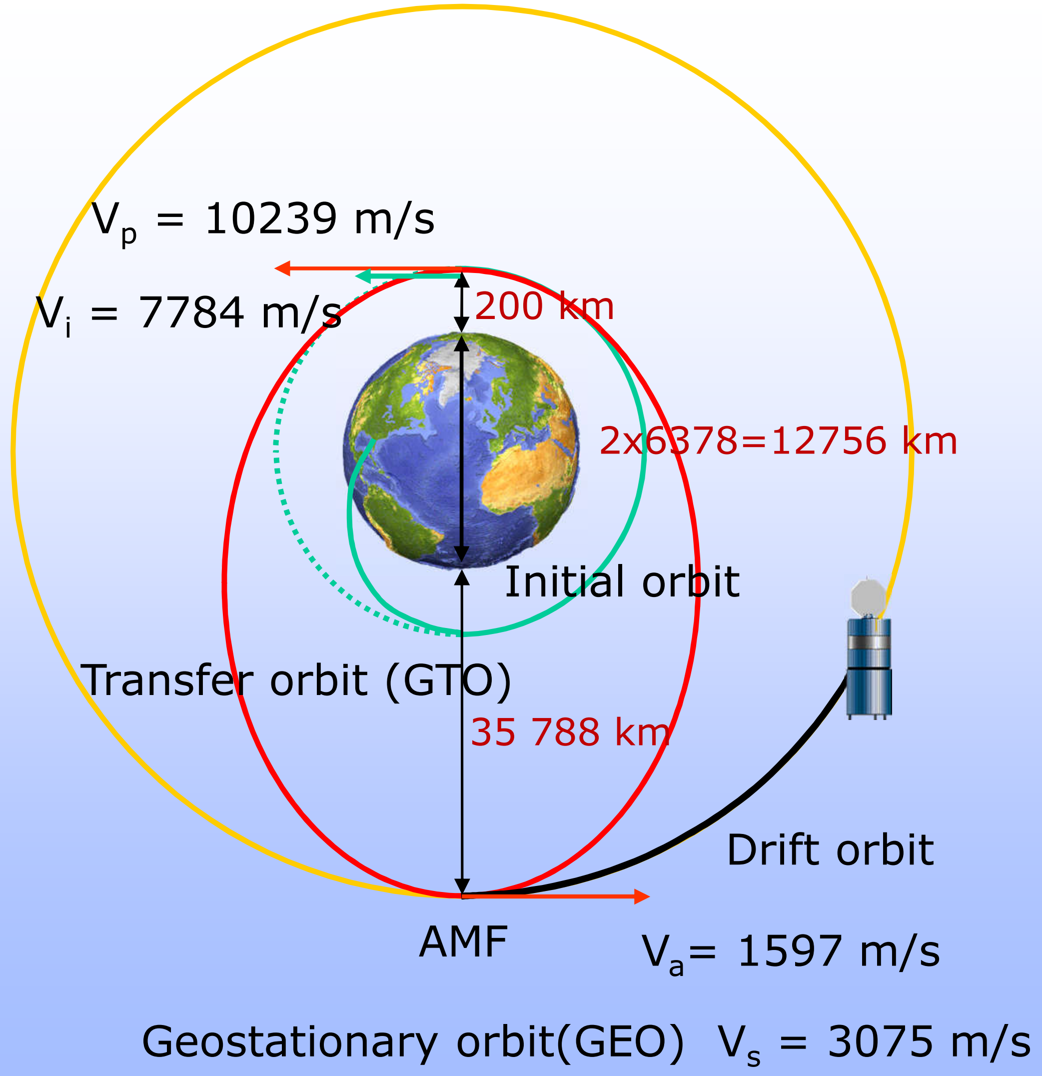
Sequence for launch and injection into transfer and geostationary orbit (Hohmann transfer)

$$v = \sqrt{\mu\left(\frac{2}{r} - \frac{1}{a}\right)}$$

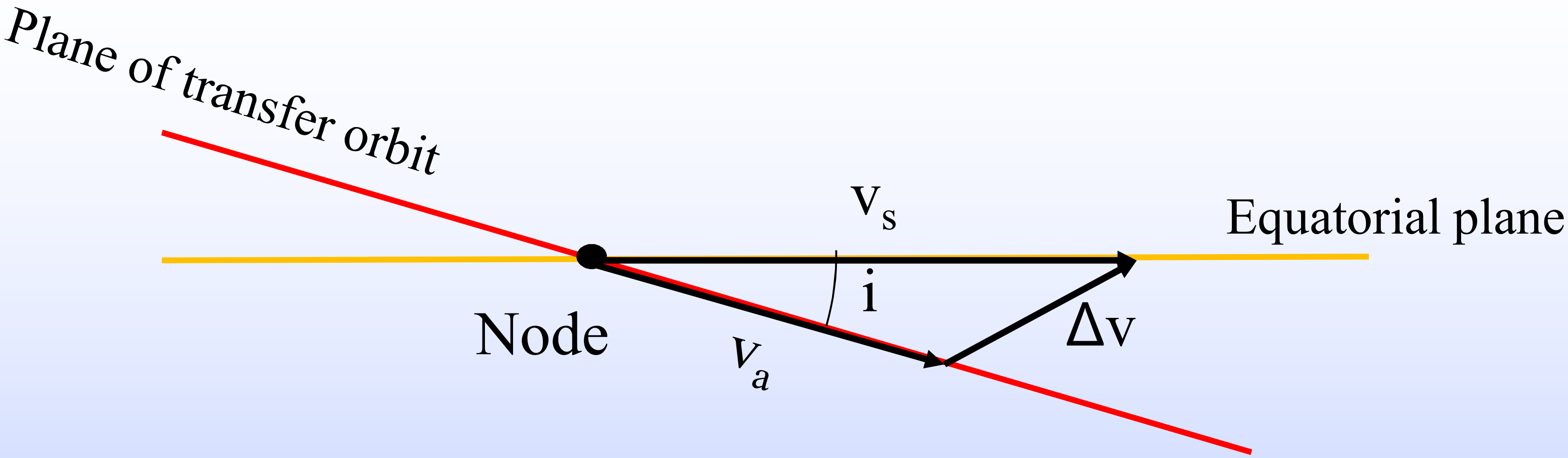
$$\Delta v_p = v_p - v_i$$

$$\Delta v_a = v_s - v_a$$





Correction of inclination and transfer to GEO orbit



$$v_a = \sqrt{\mu\left(\frac{2}{r_a} - \frac{1}{a}\right)}$$

$$\Delta v^2 = v_a^2 + v_s^2 - 2v_s v_a \cos i$$

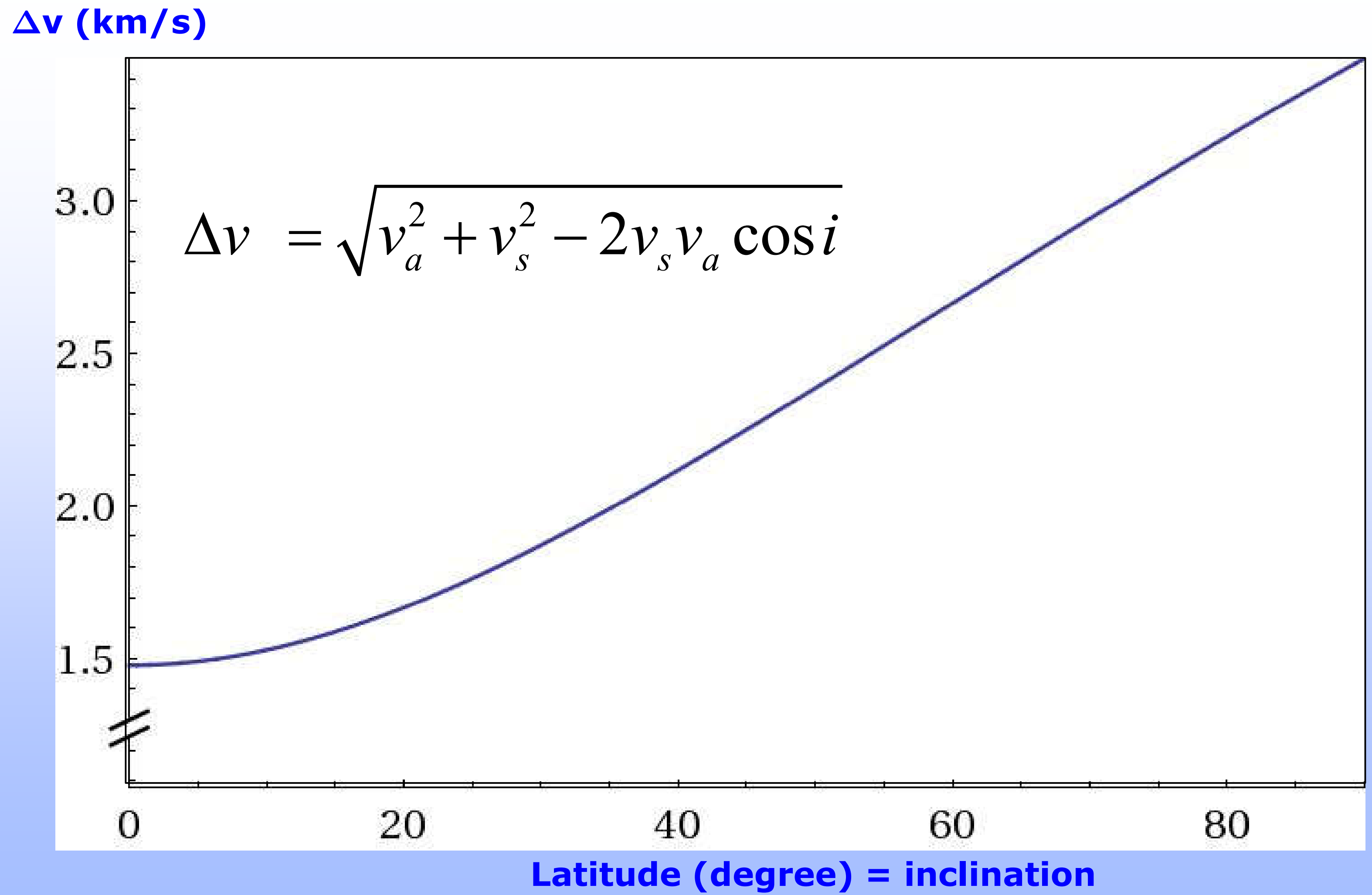
Choice of launch site

$m_i = 3000\ kg \qquad I_{sp} = 310 \sec \qquad v_s = 3{,}075\ km/s$

$$\Delta m_p = m_i \cdot \left(1 - e^{-\frac{\Delta V}{I_{sp} \cdot g}} \right)$$

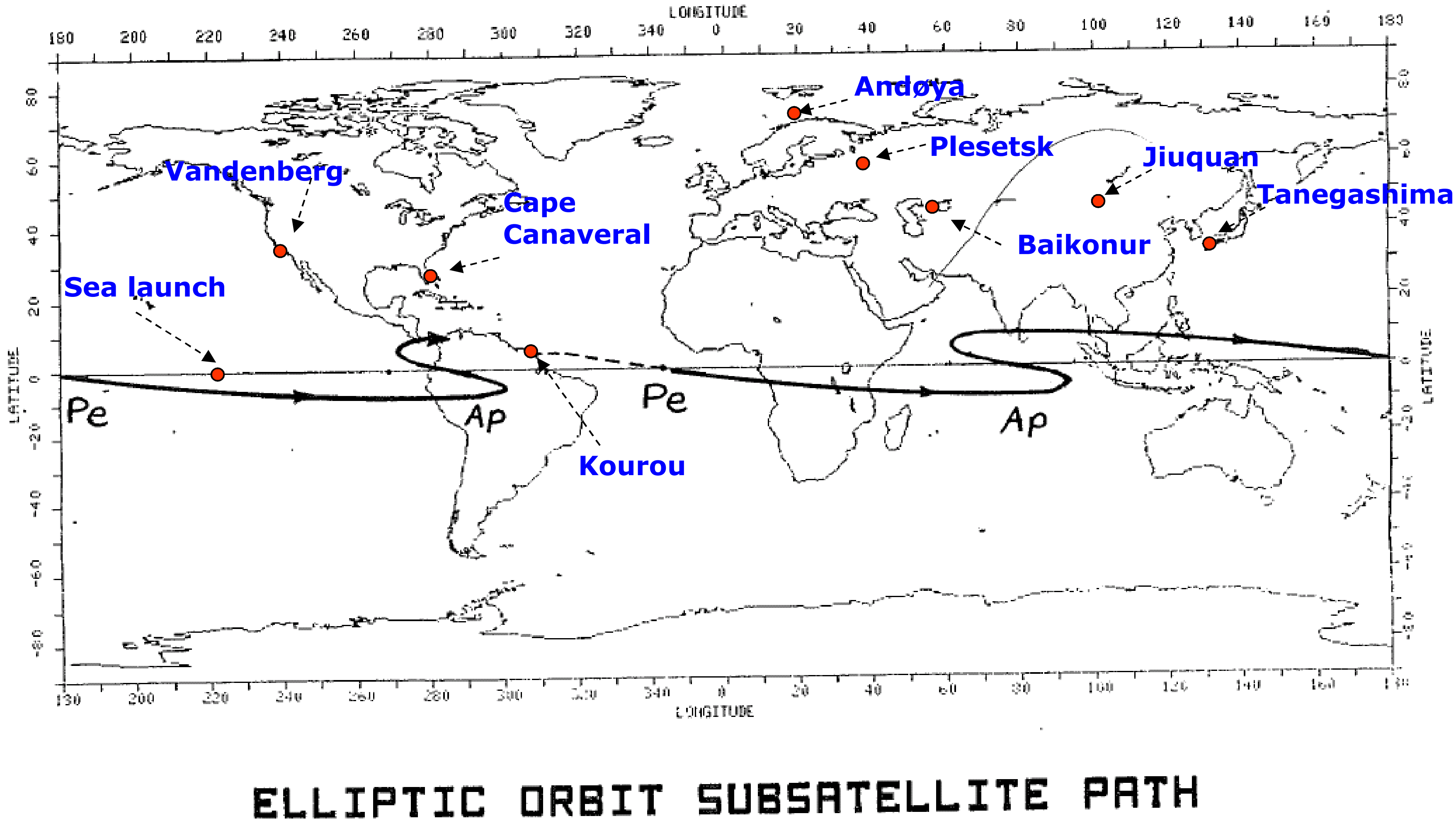
Launch site	Δv (km/s)	Propellant (kg)	Satellite mass (kg)
Kourou (5.23^{0}N)	1.491	1163	1837
Cape Canaveral (28,5^{0}N)	1.837	1360	1640
Baikonur (46^{0}N)	2.277	1581	1419
Equator (0^0), "Sea Launch"	1.478	1154	1846

Required Δv for circularization and correction of inclination



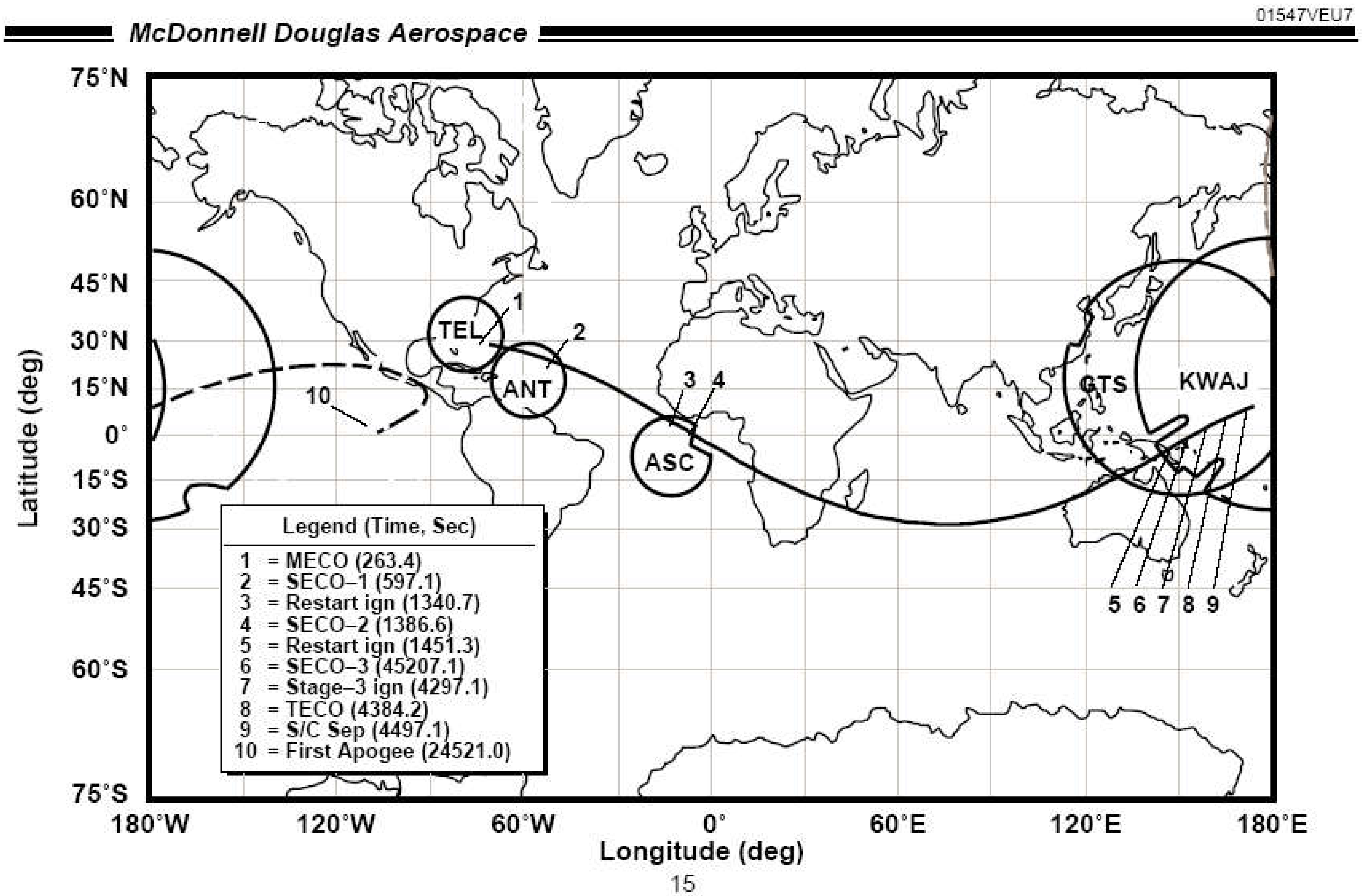
$$\Delta v = \sqrt{v_a^2 + v_s^2 - 2 v_s v_a \cos i}$$

Some major launch sites

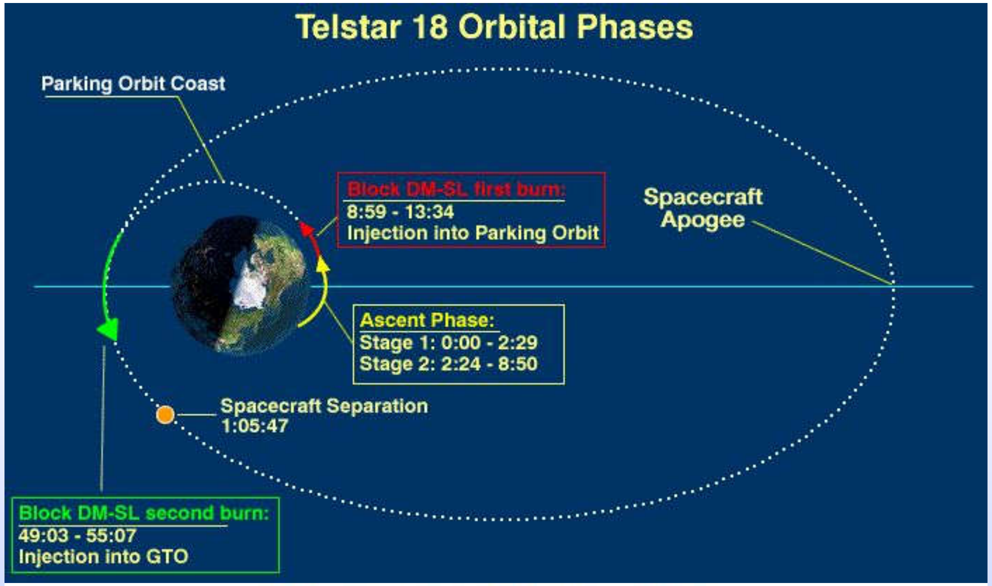


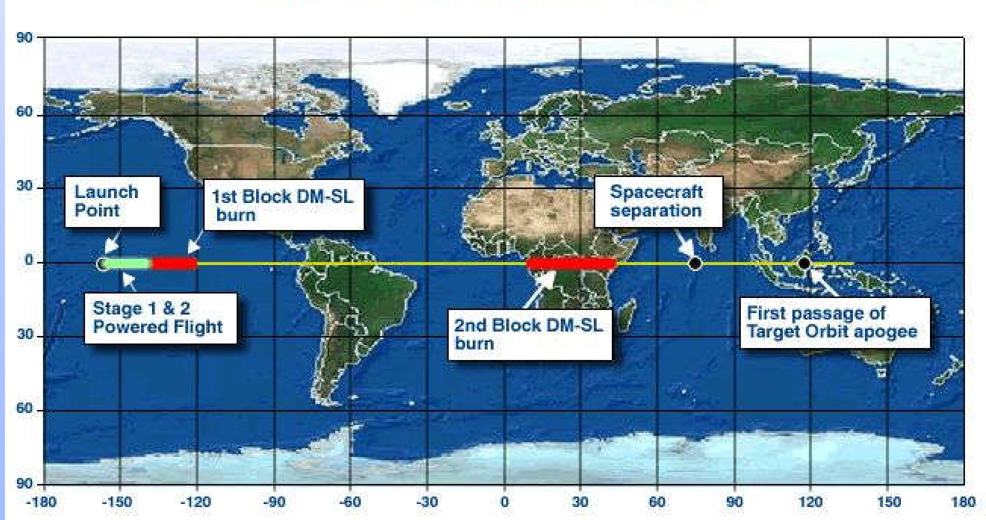
Ariane transfer orbit

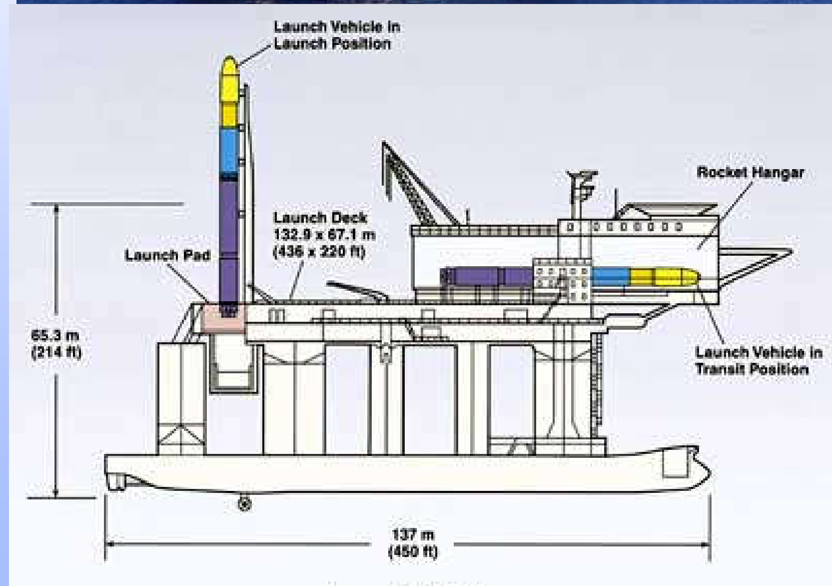
THOR II ORBITAL TRACE (E* ≥ 2 DEG)



Sea Launch



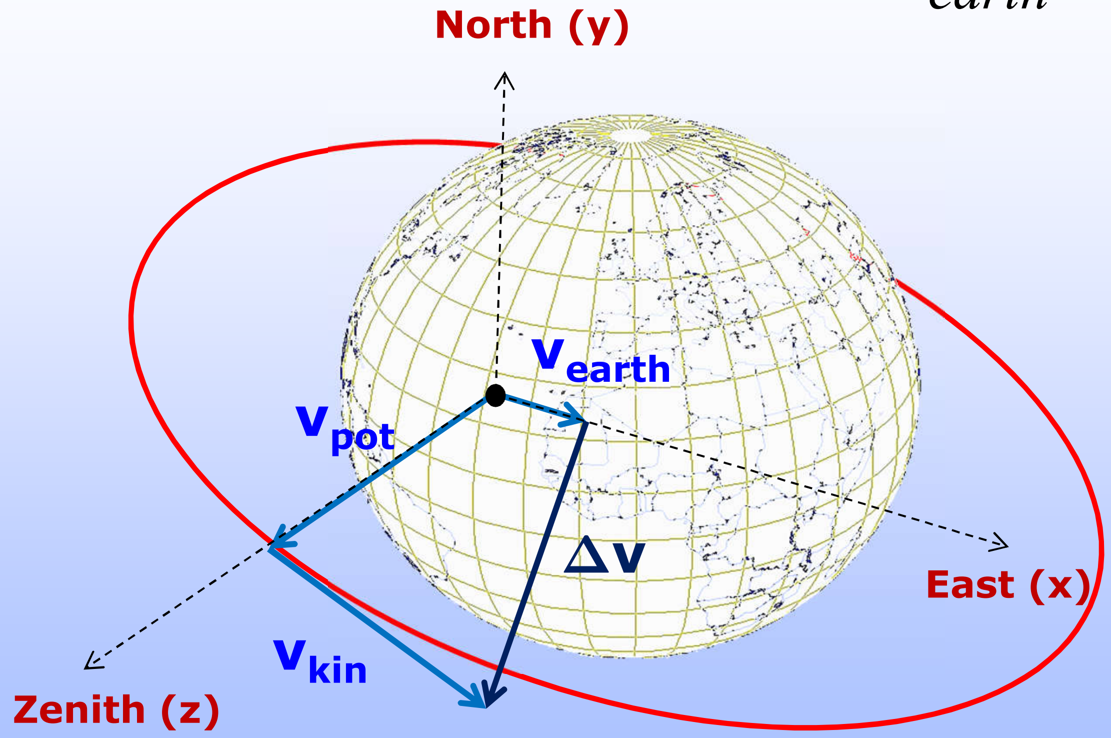




Launch Platform

Launch velocity components

$$\vec{v}_{earth} + \Delta\vec{v} = \vec{v}_{pot} + \vec{v}_{kin}$$



Total velocity the rocket has to produce from lift-off until burnout of the last stage

The rocket has to overcome the earth gravity and to give the satellite the orbital velocity

From classical mechanics the potential energy $$E_{pot} = \frac{m\mu}{r}$$

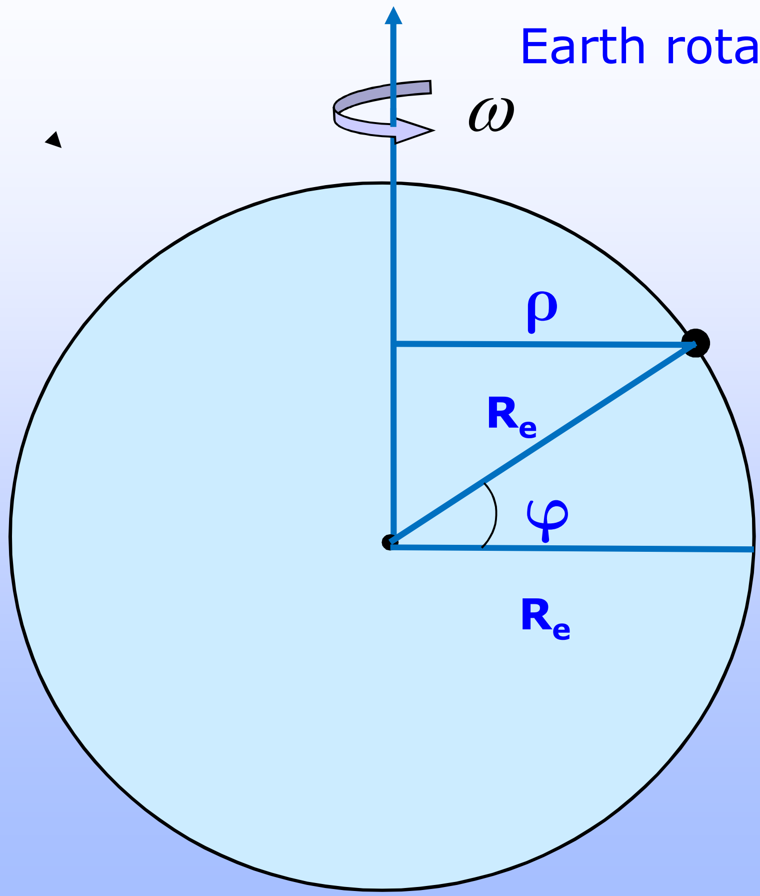
The rocket has to build up kinetic energy equal to the difference between the potential energy on the ground and the potential energy at the orbital height

$$m\frac{v_{pot}^{2}}{2} = m\frac{\mu}{R_{earth}} - m\frac{\mu}{R_{orbit}} \qquad \Rightarrow \qquad v_{pot} = \sqrt{\frac{2\mu h}{R_{earth} \cdot \left(R_{earth} + h\right)}}$$

To maintain an orbit it is also necessary to produce enough kinetic energy

$$E_{kin} = \frac{1}{2} m v_{kin}^{2} \qquad \Rightarrow \qquad v_{kin} = \sqrt{\frac{\mu}{\left(R_{earth} + h\right)}}$$

Velocity contribution from earth



Eastward velocity contribution:

$$v_{earth} = \omega\rho = \omega R_{earth} \cos\varphi$$

Example

Launch site: Cape Canaveral, lattitude φ=28.5^0
Orbit: Circular, h=185 km, i=28.5^0
Satellite mass: m=2000 kg

Questions:

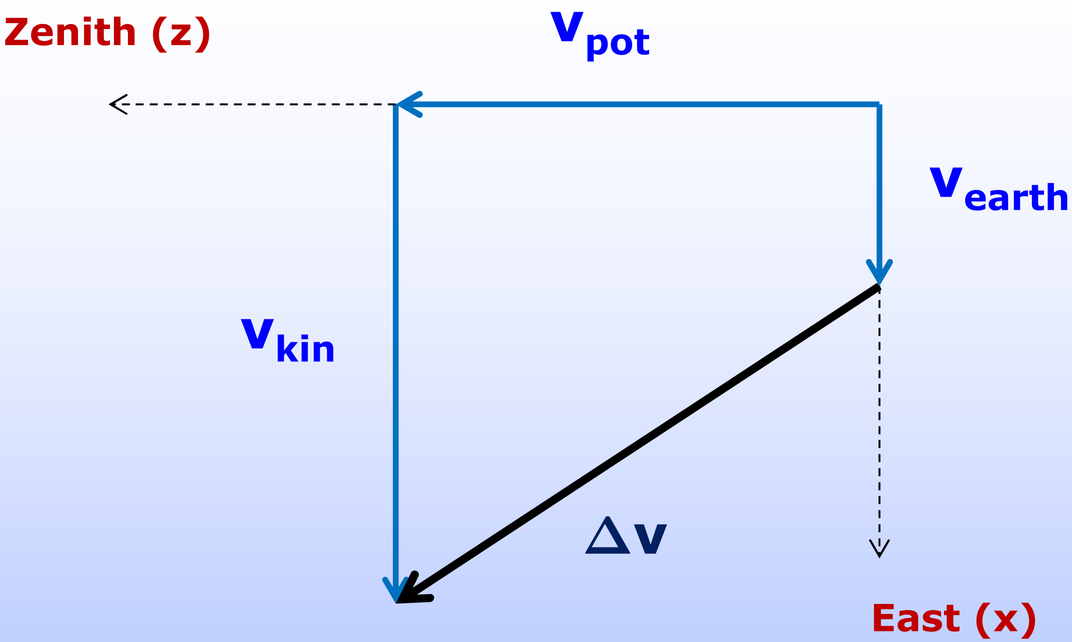
1) What ΔV does a rocket need to generate?
2) Design a launch vehicle for this mission.

Available (commercial) rocket stages:

Stage no	Type of propellant	Stage mass (kg)	Stage propellant mass (kg)	I_{sp} (s)	Burn time (s)
1	Solid	53 000	48 800	270	80
2	Solid	11 000	10 000	290	150
3	Hydrazine	700	340	400	1500

P. Berlin: Satellite platform design, Jan. 2005

Velocity vectors in the orbital plane

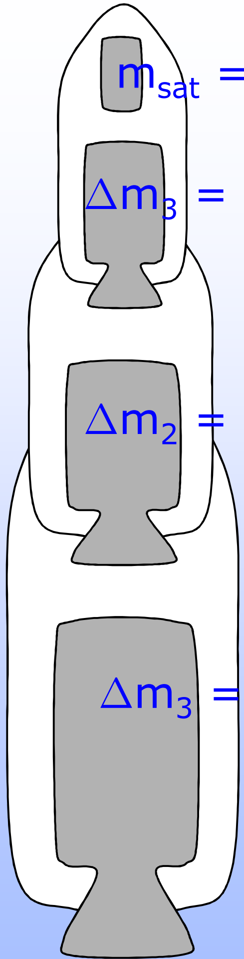


Due to atmosphere, typically ~1.5 km/s

$$\Delta v = \sqrt{\left(v_{kin} - v_{earth}\right)^2 + v_{pot}{}^2} + \Delta v_{loss}$$

$$= 7.624 + 1.500 = 9.124 \quad \left[km / s\right]$$

Assumption: a three-stage rocket



$$\Delta v = gI_p \ln\left(\frac{m_i}{m_i - \Delta m}\right) = gI_p \ln\left(\frac{m_{initial}}{m_{final}}\right)$$

$$\Delta v_{tot} = \Delta v_1 + \Delta v_2 + \Delta v_3 = 9.124 \quad (km/s)$$

Δv for the different stages will be:

$$\Delta v_1 = gI_{sp1} \ln\left(\frac{m_1 + m_2 + m_3 + m_{sat}}{m_1 + m_2 + m_3 + m_{sat} - \Delta m_1}\right) = 3.484 \quad [km/s]$$

$$\Delta v_2 = gI_{sp2} \ln\left(\frac{m_2 + m_3 + m_{sat}}{m_2 + m_3 + m_{sat} - \Delta m_2}\right) = 3.724 \quad [km/s]$$

$$\Delta v_3 = gI_{sp3} \ln\left(\frac{m_3 + m_{sat}}{m_3 + m_{sat} - \Delta m_3}\right) = 0.528 \quad [km/s]$$

Total Δv will be:

$$\Delta v_{tot} = \Delta v_1 + \Delta v_2 + \Delta v_3 = 7.736 \quad (km/s)$$

which is less than the requirement 9.124 km/s.

We can try to add another 1. stage, making it a 4-stage rocket.

The new rocket will look like this

Stage no	Stage mass (kg)	Stage propellant mass (kg)	I_{sp} (s)	Burn time (s)	$\Delta m/\Delta t$ (kg/s)	Δv (m/s)
1	53 000	48 800	270	80	610	1 388
2	53 000	48 800	270	80	610	3 484
3	11 000	10 000	290	150	66.7	3 724
4	700	340	400	1500	0.23	528
Sat.	2000					
Total	119 700	107 940		1810		9 124

meeting the requirement

Thrust calculation

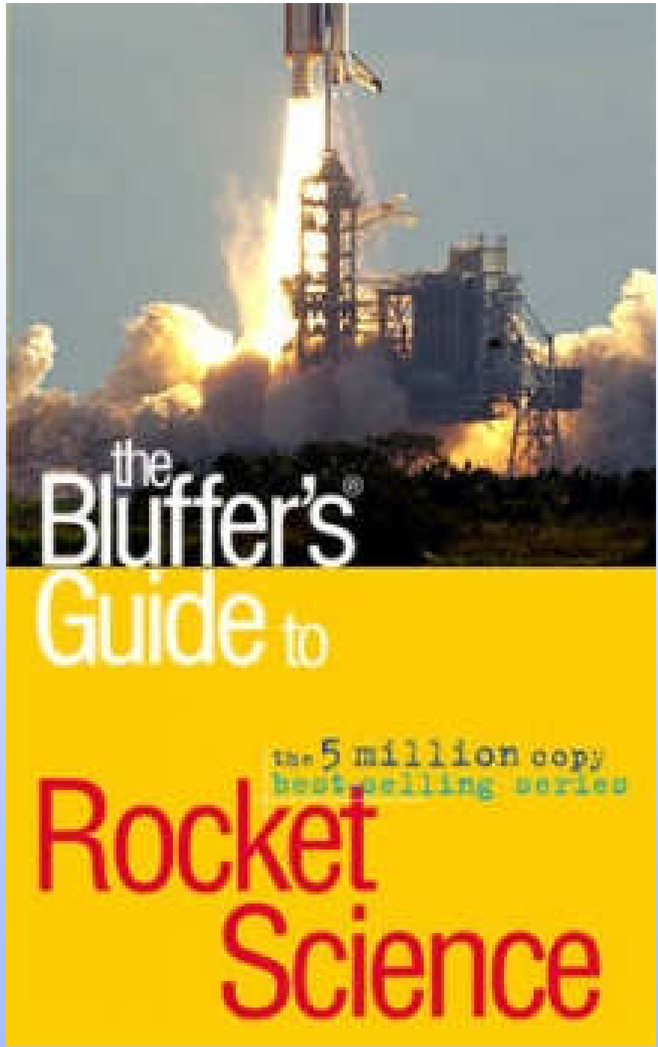
The thrust can be calculated using

$$I_{sp} = \frac{F}{g\frac{\Delta m}{\Delta t}}$$

For the 1st stage:

$$F = \frac{\Delta m}{\Delta t} \cdot I_{sp} \cdot g = \frac{48800}{80} \cdot 270 \cdot 9.81 = 1\,615\,707\ (N)$$

A speedy way to become a «rocket scientist»



But remember:

"Nobody knows what gravity really is, so don't blow your bluffing cover by trying to explain it.

The only thing known for certain is that any two physical bodies will attract each other in proportion to their sizes"

Launch of GPS IIR on Delta II

Launch campaign (Thor II on Delta II)