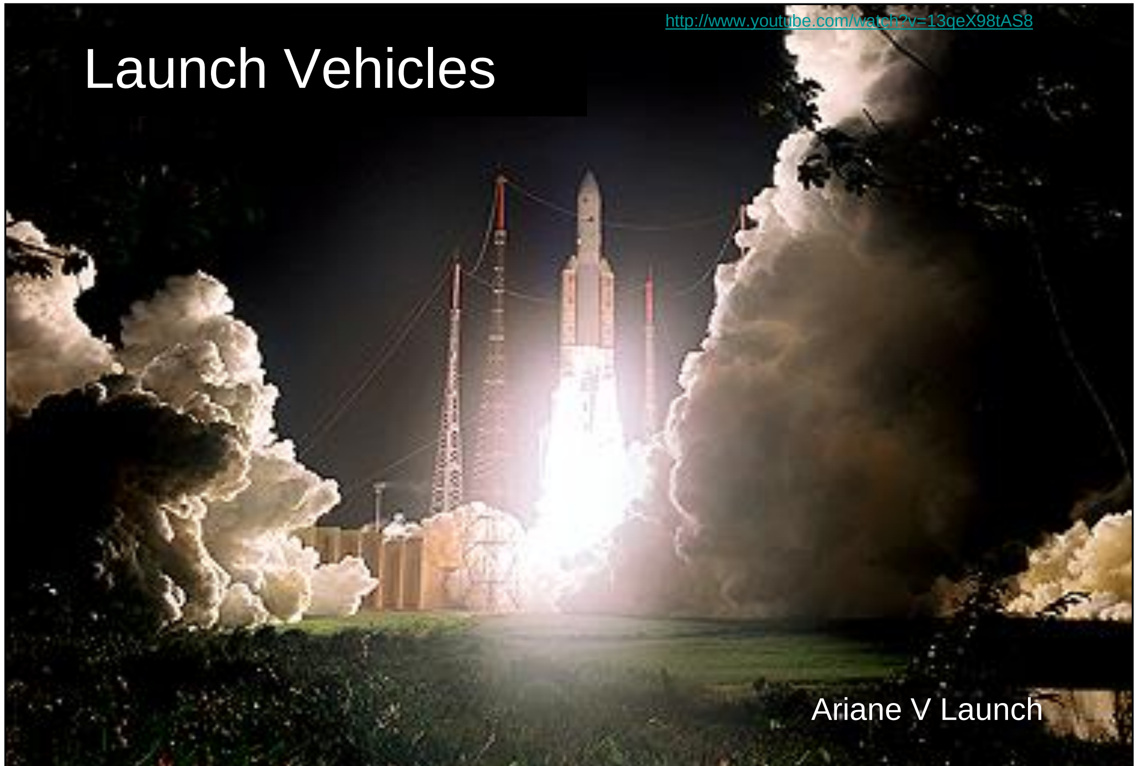
http://www.youtube.com/watch?v=13qeX98tAS8

Launch Vehicles



Ariane V Launch

Learning Objectives

1. Know the types of propellants used and be familiar with the Isp for various propellants
2. Know what needs to be done to get to orbit - losses to overcome
3. Understand the necessity for staging and be able to calculate simple staging velocities
4. Know the difference between series and parallel staging strategies, apply the Rocket Equation to both.
5. Understand what a mass fraction and propellant fraction is, and be able to use it.

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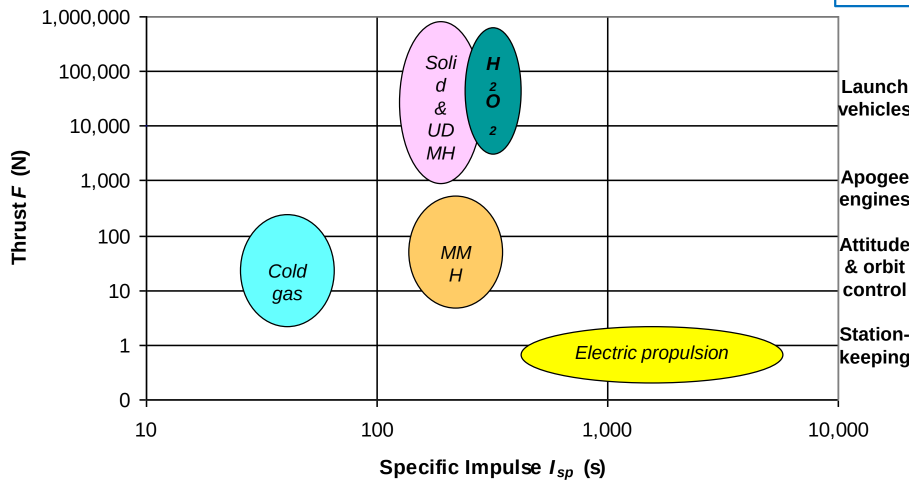
Thrust versus specific impulse for various propellants

$$I_{sp} = \frac{F}{\dot{m}}$$



Specific impulse (usually abbreviated Isp) is a way to describe the efficiency of rocket and jet engines. It represents the force with respect to the amount of propellant used per unit time.
Note that electric propulsion has very low thrust but high specific impulse so it takes AGES to get anywhere.

Propellant types

- Chemical propellants are unusual in that they are propellant and power source in one
  - Low dry mass, high mass fractions
- Old technology, simple, high energy density (next after Nuclear)
- Three main types:
  - Solids
  - Liquids
  - Hybrids

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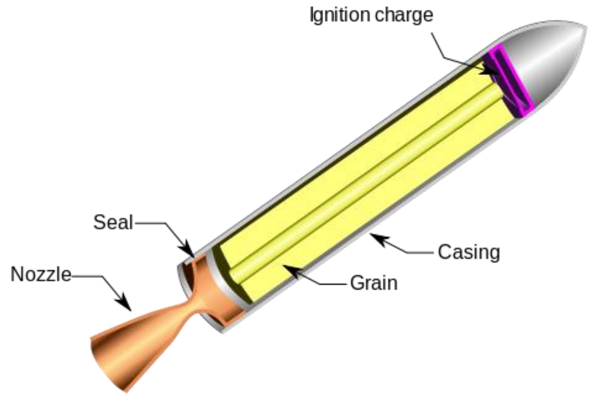
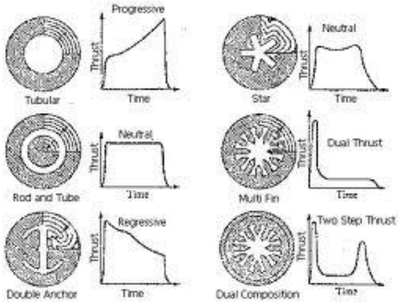
Chemical propellants lead to low launcher dry mass as they require little infrastructure, this leads to high mass fractions ie: lots of propellant for the total mass.

1.Solids





Basically big fireworks, called “motors”. Solids combine fuel and oxidiser in one composite called ‘grain’ . Motor case needs to be less massive than with liquids. Propellant must burn at correct rate to maintain pressure. Typically high mass flows, high thrusts, but low Specific impulse (~2000-2500 m/s). Grain can be shaped to control burn rate and therefore thrust profile (see diagrams).
Uses are: apogee boost motors for geostationary satellites and the shuttle solid rocket motor.

2.Liquids

3 types: monopropellant, bipropellant (and dual mode)

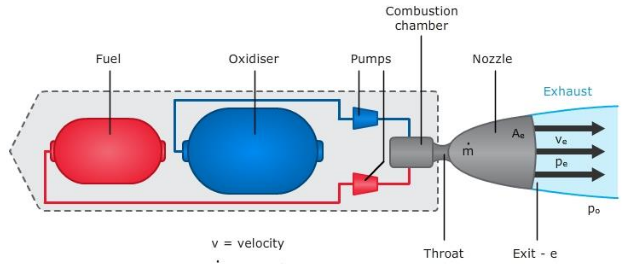
Pros and cons

1. Monopropellant
   - exothermic decomposition in presence of catalyst
   - e.g. Hydrazine N_2H_4; one type is MMH: monomethylhydrazine; another is UDMH: unsymmetrical dimethylhydrazine
   - Main use: satellite control thrusters

Called an "engine" . Liquids are pumped in to chamber at a controlled rate.
Liquids give higher Isp, thrust control, restartable engines and provide greater thrust than solids.
BUT lower reliability, problems with temperature due to the propellant lines freezing up.
Catalyst: iridium
We are not going to cover dual mode here.

2. Common Liquids

2. Bi propellants – two chemicals going bang, more common.



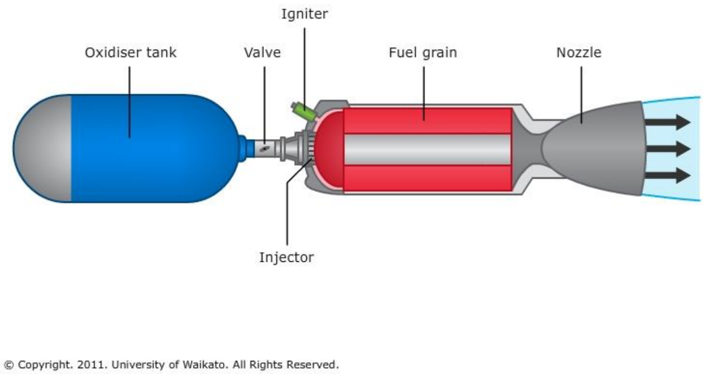
Examples:

- Liquid Oxygen and Kerosene - eg: Atlas 1st stage
- Nitrogen Tetroxide and Hydrazine - eg: Ariane V 2nd stage
- Liquid Oxygen and Liquid Hydrogen - eg: Space Shuttle main engine

Fuel and oxidiser stored separately and mixed in the combustion chamber.
Liquid at room temp or need to be stored cryogenically eg: H2 or O2.
Liquids pumped in to chamber at controlled rates.
Hypergolic propellants are fuels and oxidizers that ignite spontaneously on contact with each other and require no ignition source eg: hydrazine, MMH and UDMH.

3. Hybrids

- Normally solid fuel, liquid oxidiser



- Eg: Plastic fuel (polyethylene/PVC)+ oxidiser (N_2O/H_2O_2)

- Example: VirginGalactic SpaceshipTwo uses HTPB/N_2O

Normally solid fuel, liquid oxidiser. The rocket can be turned on and off, can be throttled and is relatively safe. Its performance similar to solids ie: lower specific thrust but higher reliability than liquids. Some of the fuels used are toxic. What might be the other disadvantages of hybrids, over eg: solids?
N2O = nitrous oxide. H2O2 = hydrogen peroxide. HTPB= Hydroxyl-terminated Polybutadiene.

Example Isp values

Class	Propellant	Isp (m/s)
Solid	Rubber	2200-2800
Liquid monopropellant	Hydrazine	1600-1900
Liquid bipropellant	Liquid Oxygen/Liquid Hydrogen	3800-4500
	Fluorine/Hydrogen	3800-4500
	Liquid oxygen/Hydrazine	3200-3800
	Liquid oxygen/Kerosene (RP-1)	2900-3400
Hybrid	Nitrous oxide/rubber (HTPB)	2400-2800

Many launch vehicles use a combination of different rocket motors. The Ariane 5 and the space shuttle both get most of their thrust at low altitude from very high thrust (but low specific impulse) solid fuel boosters and then use high specific impulse but lower thrust) liquid hydrogen/liquid oxygen motors at higher altitudes and in space. The largest rocket ever flown beyond the trialling stage was the Saturn V that launched the Apollo missions to the Moon. This used liquid fuel engines at all stages, but used relatively 'energy dense' kerosene/liquid oxygen at low altitude and liquid hydrogen/liquid oxygen at high altitude and in space.

The purpose of the rocket

- These motors/engines are used for a variety of things
  – Orbital manoeuvring systems
  – Orbit raising
  – Station keeping
  – Launch Vehicles
- Launching a satellite is tricky!

Launch losses

- **Potential energy** is energy required to raise object from one altitude to another
- Large horizontal Impulse takes finite time, so extra propellant needed to counteract **gravity losses**
- Atmosphere means we need to go up before we go horizontally to minimise **drag losses**
- **Steering losses** due to axis of rocket not aligned with **V.**

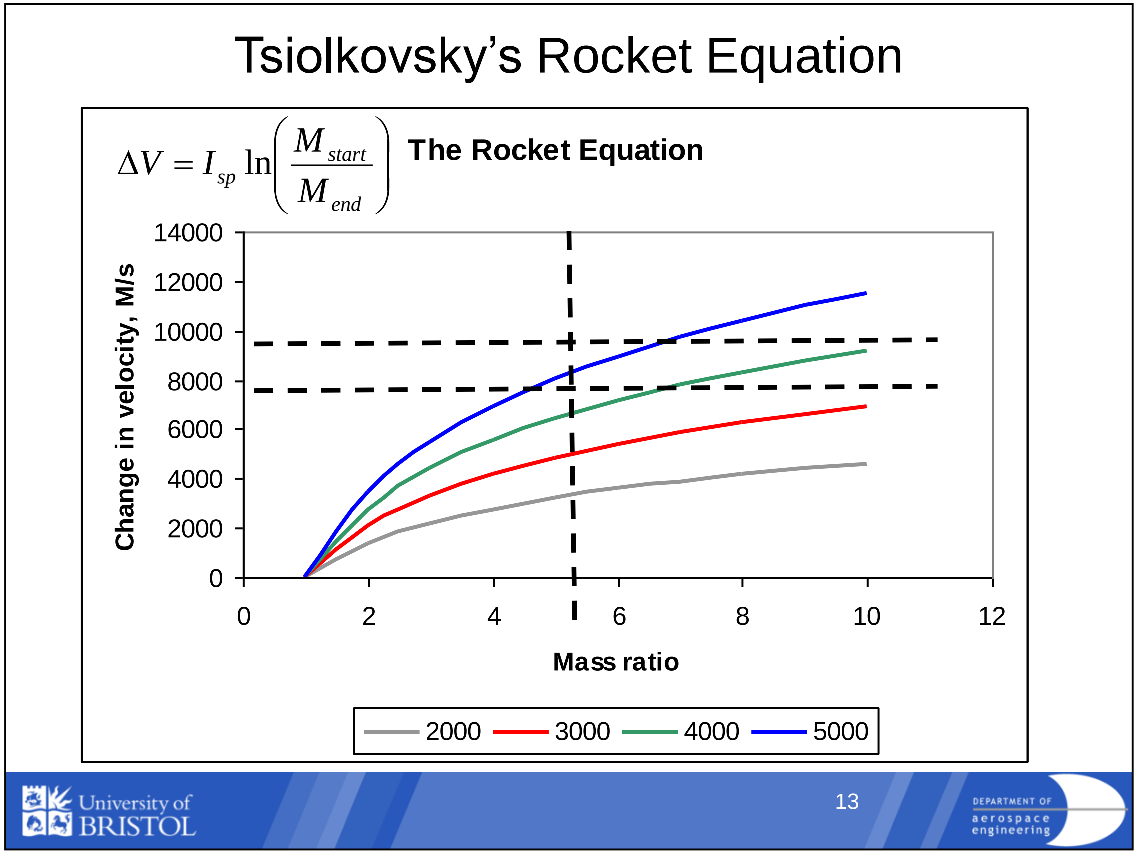
As we cannot accelerate in the horizontal direction straight away, we need to go up to a certain altitude to get through the atmosphere, the amount of energy we expend to raise our altitude is the launch losses due to **potential energy**. **Gravity losses** are due to the finite time that it takes to do the engine burn (the rocket equation assumes that the burn is instantaneous and thus is the ideal situation). Potential energy can account for 15-20% of total energy. Gravity losses account for 10-15% of fuel, drag losses only 0.5%, steering even less. If you play Kerbal you will have noticed that low velocity wastes delta V on gravity (gravity losses), high velocity wastes fuel on air resistance ie: drag losses.

Summary

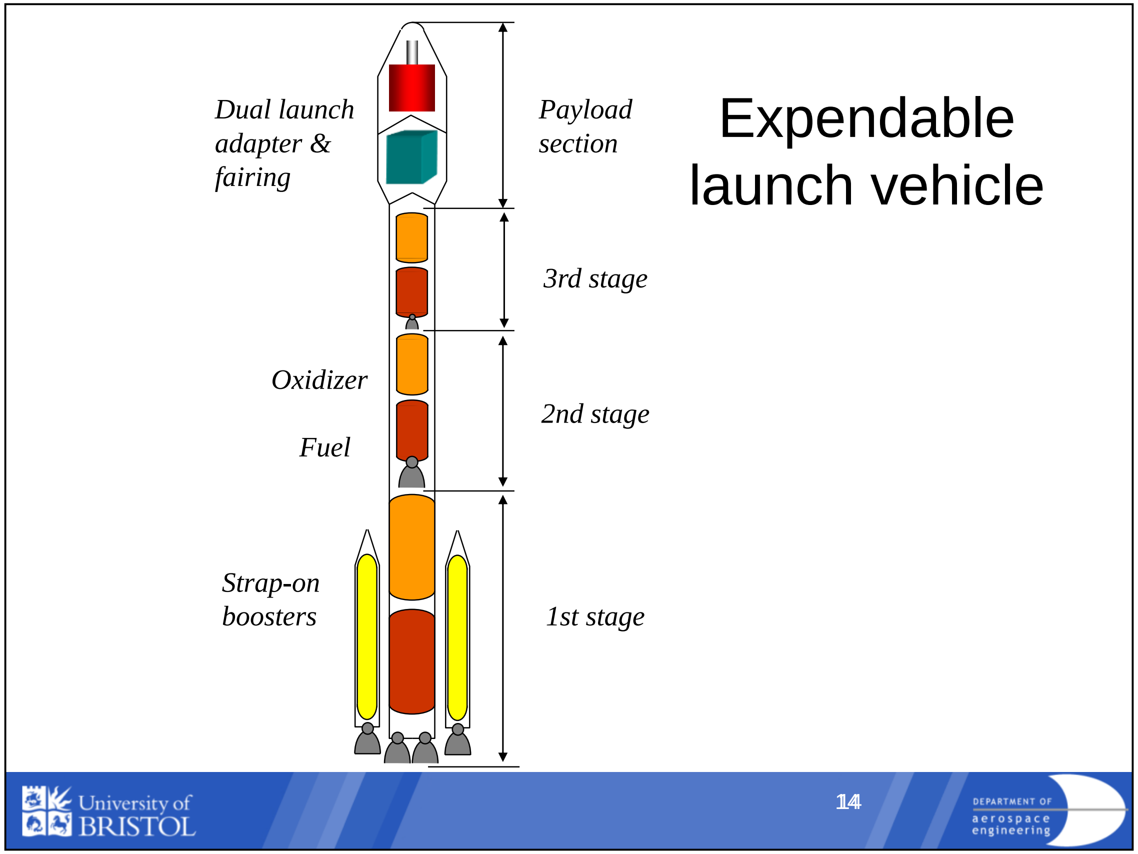
- To reach (LEO) orbit you need:

Orbital Velocity	7.7 Km/s
Get to Altitude (P.E.)	1.3 Km/s
Gravity Losses (finite time)	0.7 Km/s
Atmosphere losses (drag)	0.1 Km/s
Earth's Rotation (varies)	-0.5 Km/s
Total	9.3 Km/s

The lowest altitude where a stable orbit can be maintained, is at an altitude of 185 km. This requires an Orbital velocity approximately 7777 m/sec.
A Hydrogen-Oxygen system with an effective average exhaust velocity (from sealevel to vacuum) of 4000 m/sec would require Mi/ Mf = 9.7.



If we take an Isp of 3000m/s for typical solid rocket boosters (for comparison Space Shuttle main engines in a vacuum give 4500m/s, that's about the best we can get) and we need 9300m/s to get to LEO, then we get a mass ratio of 22, (for comparison SSME's give 7.9), so mass of structure and engine (dry mass) is 1/22=4.5% of the whole rocket mass (dry mass + propellant=wet mass). In other words, 95.5% propellant. This is not possible! Ideally we would like a rocket to be 100% propellant and zero dry mass! This is just not possible. For comparison, a car and a ship are about 5-10% propellant mass. We need to reduce the dry mass as much as possible. So, how do we do this?



Multiple stages! Staged rockets allow us to improve the efficiency of the rocket, as we are getting rid of structural mass as we go. Also the upper stages can use motors which are optimised for vacuum rather than atmospheric pressure.

Answer: Multiple stages

A certain fraction of the vehicle mass is dumped after use allowing the non-payload mass carried to orbit to be minimised.

- All vehicles currently multi-stage
- Can be series (as described) or parallel ie: light at same time eg: Space Shuttle main engine and solid rocket boosters.
- **So ΔV changes are cumulative**

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Use a first stage to get you some velocity, Then dump the stage, Then light 2nd, dump, then light 3rd stage.

Important Equations

Rocket Eqn: $$\Delta V = I_{sp} \ln\left(\frac{M_{start}}{M_{end}}\right) \quad \text{(9-1)}$$

Mass propellant

Propellant mass fraction f_p: $$f_p = \frac{\Delta m}{\Delta m + m_d} \quad \text{(9-2)}$$

Dry mass

Also, by rearranging: $$m_d = \Delta m\left(\frac{1}{f_p} - 1\right) \quad \text{(9-3)}$$

Multistages: $M_{start} = m_1 + m_2 + m_3 + \textit{payload mass}$ (9-4)

Where m_1, m_2 are wet masses: ie: dry mass + propellant

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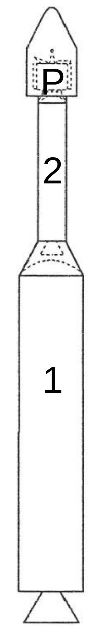
Mstart is defined as the dry mass plus propellant mass, Mend is just the dry mass.

Solid Multistage

- Assume Isp=2800 m/s
- Dry mass= 0.12 * propellant
- SSTO not possible, but either example to the right will get us to > 9.3km/s
- 4 stage would weigh 63 tonnes – not much benefit 3->4 stages

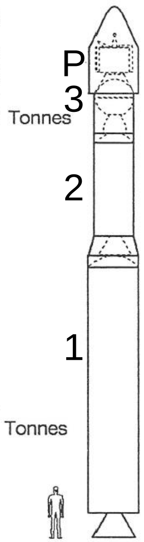


TWO STAGE

Payload	1.00	
Upper stage dry	1.21	
Upper stage propellant	10.10	
Lower stage dry	14.74	
Lower stage propellant	122.80	
Launch Mass	149.85	Tonnes

THREE STAGE

Payload	1.00	
Upper stage dry	0.35	
Upper stage propellant	2.90	
Mid stage dry	1.45	
Mid stage propellant	12.10	
Lower stage dry	6.13	
Lower stage propellant	51.10	
Launch Mass	75.03	Tonnes



SSTO is single stage to orbit – the holy grail of rocketry!. Let us assume that the dry mass is 12% of the propellant mass. The results of the calculations using the rocket equation are on the right for 2 stage and for 3 stages. Check yourself that 4 stages gives a mass of 63 Tonnes!

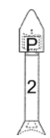
2 stage example

$$\Delta \boldsymbol{V} = \boldsymbol{I_{sp}} \ln \left[\frac{\boldsymbol{M_{start}}}{\boldsymbol{M_{end}}} \right]$$

$$\Delta V_1 = I_{sp1} \ln \left[\frac{m_1 + m_2 + m_{payload}}{m_1 + m_2 + m_{payload} - \Delta m_1} \right]$$

$$\Delta V_1 = 2800 \ln \left[\frac{137.54+11.31+1.0}{137.54+11.31+1.0-122.8} \right] = 4793 \text{m/s}$$

Then…



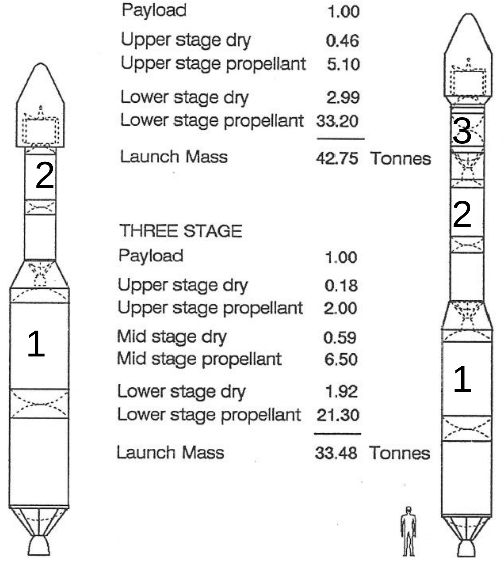
$$\Delta V_2 = I_{sp2} \ln \left[\frac{m_2 + m_{payload}}{m_2 + m_{payload} - \Delta m_2} \right]$$

$$\Delta V_2 = 2800 \ln \left[\frac{11.31+1.0}{11.31+1.0-10.10} \right] = 4809 \text{m/s}$$

$$\Delta V = \Delta V_1 + \Delta V_2 = 4793 + 4809 = 9601 \text{m/s}$$

We apply the rocket equation at the top to the first stage to get the delta V provided by this stage: deltaV1.
m1 is the lower stage dry +lower stage propellant,
m2 is upper stage dry +upper stage propellant
Then we dump the first stage (m1) and apply the rocket equation to what is left without the first stage to find the deltaV2 provided by the 2nd stage.

N_2O_4/UDMH multistage



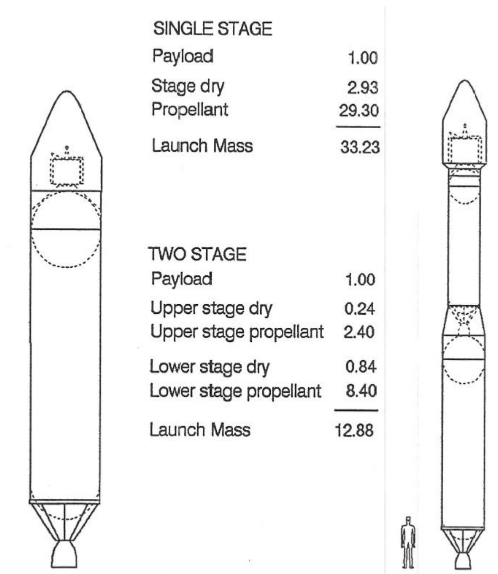
TWO STAGE		
Payload	1.00	
Upper stage dry	0.46	
Upper stage propellant	5.10	
Lower stage dry	2.99	
Lower stage propellant	33.20	
Launch Mass	42.75	Tonnes

THREE STAGE		
Payload	1.00	
Upper stage dry	0.18	
Upper stage propellant	2.00	
Mid stage dry	0.59	
Mid stage propellant	6.50	
Lower stage dry	1.92	
Lower stage propellant	21.30	
Launch Mass	33.48	Tonnes

- Assume Isp=3200 m/s
- Assume dry mass = 0.09*propellant load
- Still can't do SSTO
- Examples accelerate to the same velocity (9600) as the solid on previous overhead
- 4 stage mass = 31 tonnes

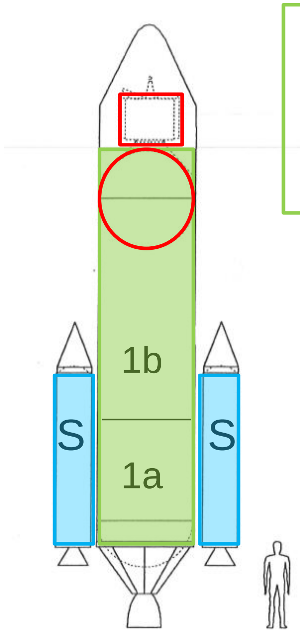
Let's change the dry mass fraction as well as the propellant. Remember that SSTO=Single Stage to Orbit.

LOX/LH Examples



- Assume Isp = 4500m/s
- Dry mass =0.1 propellant load
- SSTO now possible (just)
- Or use a two stage and reduce launch mass
- ΔV = 9600 as previous,
- 3 stage 11.8 tonnes

Parallel Example



LH/LOX Main core
12t propellant
1.2t dry mass
Burn time=540s
Thrust=99900N

2 Solid boosters
5t propellant
0.6t dry mass
Burn time=180s
Thrust=76450N

Payload = 1 t

1. First we need to work out "Effective I_{sp}":

Core

Mass flow = 12000/540=22.2 kg/s

I_{sp} = 99900/22.2=4500 Ns/kg

Booster

Mass flow = 5000/180 = 27.8 kg/s

I_{sp} = 76450/27.8 = 2750 Ns/kg

Effective I_{sp} during parallel burn: $I_{sp} = \frac{F}{\dot{m}}$

(99900 +2x76450)/(22.2+2x27.8) = 3250Ns/kg

t= tonne. There are 2 solid rocket boosters and one main LH/LOX core. BUT at the beginning for 180s we have two systems burning at once: 2 SRBs and part of the main core. This is called the parallel burn. Then afterwards for 540-180s just the main core is burning. We need to work out the effective Isp for the parallel burn as there are 2 Isps involved.

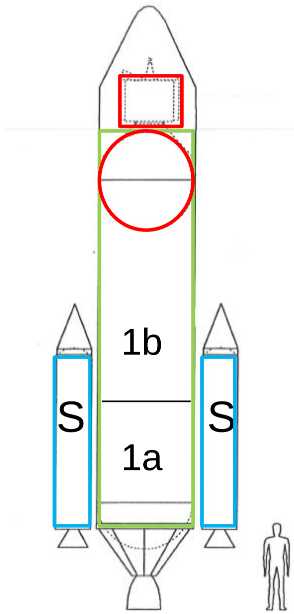
SI=specific impulse or Isp. Start mass at the beginning of final burn is 11.4-1.2 (dry mass)=10.2 tonnes. Final mass= 12-8 tonnes.

If Isp is not same, we must take an average called effective Isp (can also just average it).

LH/LOX Main core
12t propellant
1.2t dry mass
Burn time=540s
Thrust=99900N

Parallel example cont.

2 Solid boosters
5t propellant
0.6t dry mass
Burn time=180s
Thrust=76450N



2. Calculate m_{start} and m_{end}

Total m_{start} = 12+1.2+2*(5+0.6)+1 = 25.4 tonnes

Parallel burn:

$m_{LH/LOX}$ = (180/540)*12 = 4t and $m_{boosters}$ = 10t

m_{end}= m_{start}- ($m_{boosters}$ + $m_{LH/LOX}$) = 25.4-10-4=11.4t

Final burn:

m_{start} = 12+1.2+1-4=10.2t and m_{end}= 10.2-8=2.2t

12-4 prop

3. Use rocket equation to calculate deltaV
Parallel burn (solid and LH/LOX)

$$\Delta V = I_{sp}.ln\left[\frac{m_{start}}{m_{end}}\right] = 3250.ln\left[\frac{25.4}{11.4}\right] = 2604m/s$$

Final burn (LH/LOX only)

$$\Delta V = I_{sp}.ln\left[\frac{m_{start}}{m_{end}}\right] = 4500.ln\left[\frac{10.2}{2.2}\right] = 6902m/s$$

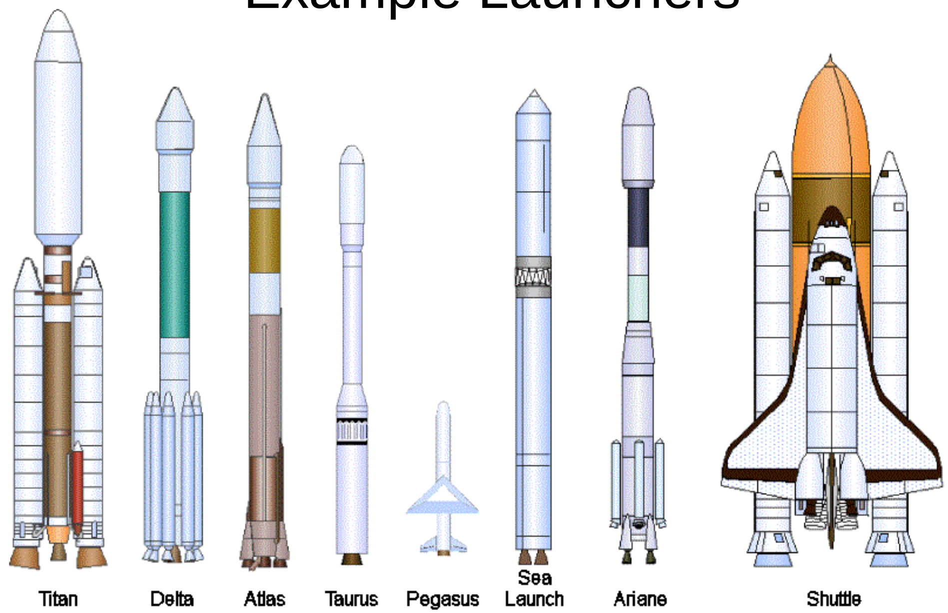
For the final burn, start mass = 12+1.2-4+1=10.2

Example launchers

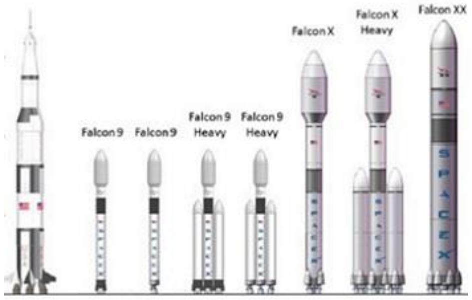
Rocket	Stage	Propellant
Atlas/Centaur (1962)	0 1 2	LOX/RP-1 LOX/RP-1 LOX/LH2
Saturn V (1967)	1 2 3	LOX/RP-1 LOX/LH LOX/LH
Space Shuttle (1981)	0 1	PBAN Solid LOX/LH2
Delta II (1989)	0 1 2	HTPB Solid LOX/RP-1 N2O4/Aerozine 50
Ariane V (1996)	0 1 2	Solid N2O4/MMH LO2/LH2
Long March 3A (1994)	1 2 3	N2O4/UDMH N2O4/UDMH LO2/LH2

If there is a 0 that means there is a parallel stage.
RP-1: Kerosene

Example Launchers



New launchers



Space Launch System (NASA’s shuttle derived system) (top left), Falcon 9 (privately built) (top right)
Chinese new launchers (based on Long March) (bottom left), India’s launch fleet (bottom right).
You do not need to know these for the exam.

Summary

1. There are 3 types of propellants: Solids, Hybrids, Liquids
2. To reach orbit, you need 9.3km/s, due to PE, gravity, drag and steering losses.
3. You can make the rocket more efficient by getting rid of the mass of the structure as you go along. So you will need more than one stage.
4. You can use the rocket equation to work out the propellant masses for the stages:

$$\Delta V = I_{sp} \ln\left(\frac{M_{start}}{M_{end}}\right) \qquad M_{start} = m_1 + m_2 + m_3 + payload$$

5. Some rockets have stages which burn at the same time instead of sequentially, these are called parallel. Use effective I_{sp} to calculate these.
6. Propellant mass fraction is defined as: $f_p = \dfrac{\Delta m}{\Delta m + m_d}$

m_d is mass of the dry structure.

Test Yourself! (Feedback)

1. Which is the propellant with the highest specific impulse?
2. Why is a hybrid so called?
3. How much delta V does a rocket need to reach LEO?
4. How much delta V does the N_2O_4/UDMH multistage rocket in slide 24 produce for each of its 2 stages?