ΔΙΙΔΡΥΜΑΤΙΚΟ ΠΡΟΓΡΑΜΜΑ ΜΕΤΑΠΤΥΧΙΑΚΩΝ ΣΠΟΥΔΩΝ

ΔΙΑΣΤΗΜΙΚΕΣ ΤΕΧΝΟΛΟΓΙΕΣ, ΕΦΑΡΜΟΓΕΣ και ΥΠΗΡΕΣΙΕΣ

Space Systems, Missions & Launchers

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ΠΑΝΕΠΙΣΤΗΜΙΟ ΠΑΤΡΩΝ
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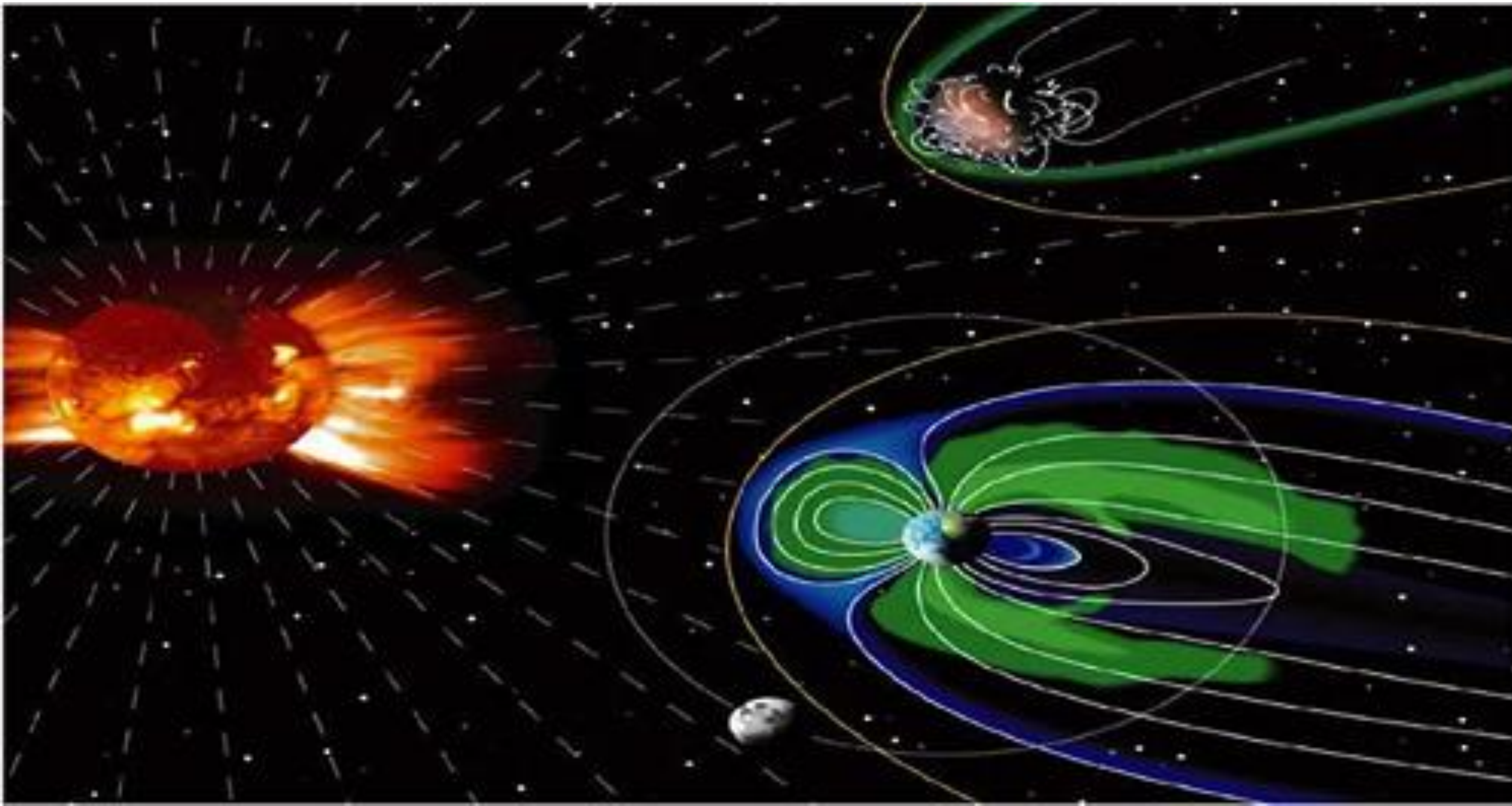
ΕΛΛΗΝΙΚΗ ΔΗΜΟΚΡΑΤΙΑ
Εθνικόν και Καποδιστριακόν Πανεπιστήμιον Αθηνών
ΙΔΡΥΘΕΝ ΤΟ 1837

Course Content

- **Week 1/2:** Introduction to space system design methodology: requirements, trade-off analysis, design specifications, system budgets. Introduction to space system architecture. Launch Vehicles.
- **Week 3:** Orbit Mechanics: celestial mechanics, orbits, trajectory design and spacecraft maneuvers
- **Weeks 4-9:** Spacecraft sub-systems design: Structure & configuration; Power, the power budget and solar array and battery sizing; Communications and the link budget; Attitude determination and control; Orbit determination and control; propulsion Thermal control.
- **Week 10:** Review, tutorial problems/exam mock up
- Recommended Text Books:
  - Understanding Space: An Introduction to Astronautics, 3rd Edition (Space Technology), J. Sellers
  - SMAD-Space Mission Analysis and Design, J. Wertz

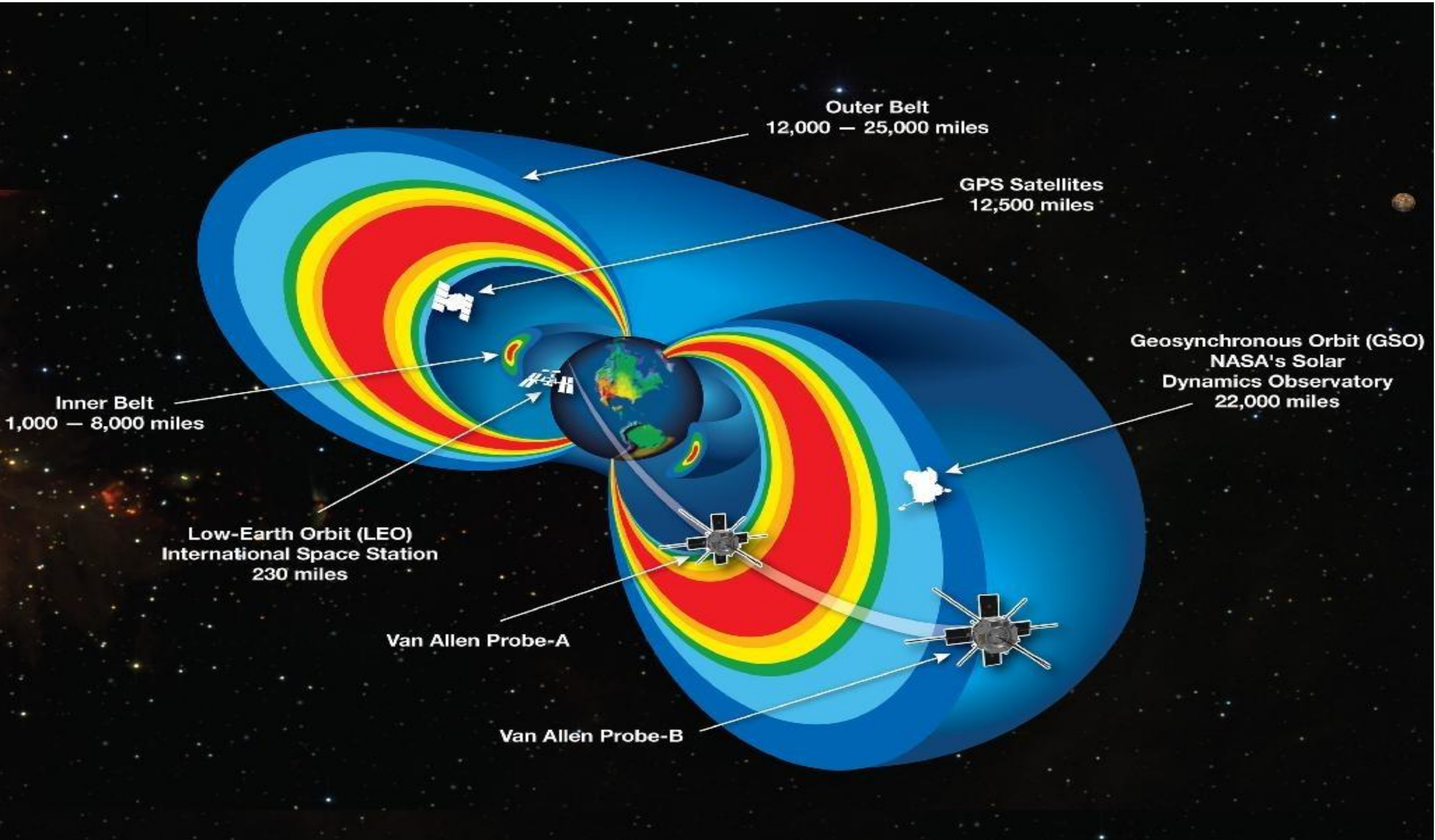
Space and Spacecraft Environment: Radiation, vacuum, debris, spacecraft charging, material behaviour and outgassing



Radiation

- Space is a radiation environment from the Sun (thermal infrared, ultraviolet/X-ray, solar wind particles), the Earth (thermal infrared, reflected radiation, van Allen radiation belts), and the cosmos (galactic cosmic rays whose source is unknown).
- There are two belts – an inner belt centred at 1.5 R_E primarily of protons >100 MeV and an outer belt centred at 3.5 R_E primarily of electrons >1.6 MeV. At lower energies, the particle distribution is more diffuse out to 10 R_E.
- The slot region between the inner and outer Van Allen belts at 2-3 R_E is dominated by transient trapping of solar flare generated particles.
- Most missions generally experience a total accumulated dose of <100 krad. Such particle radiations can cause SEU (single event upsets) in electronics, primarily due to protons in SAA and GCR and solar flares at high latitudes.
  - SAA: South Atlantic Anomaly, GCR: Galactic Cosmic Rays

Van Allen Belts in 3D (NASA)



Space Environment

- During a space mission, a spacecraft will need to survive in a number of different environments:

  i. **launch environment** subjecting the spacecraft to launch accelerations, acoustic noise and vibration

  ii. **space environment** in orbit subjecting the spacecraft to atmosphere (eg. atomic oxygen), electromagnetic radiation, high energy particles, and temperature extremes

  iii. **entry environment** (for lander) subjecting the spacecraft to high thermal loads, high pressure and high decelerations

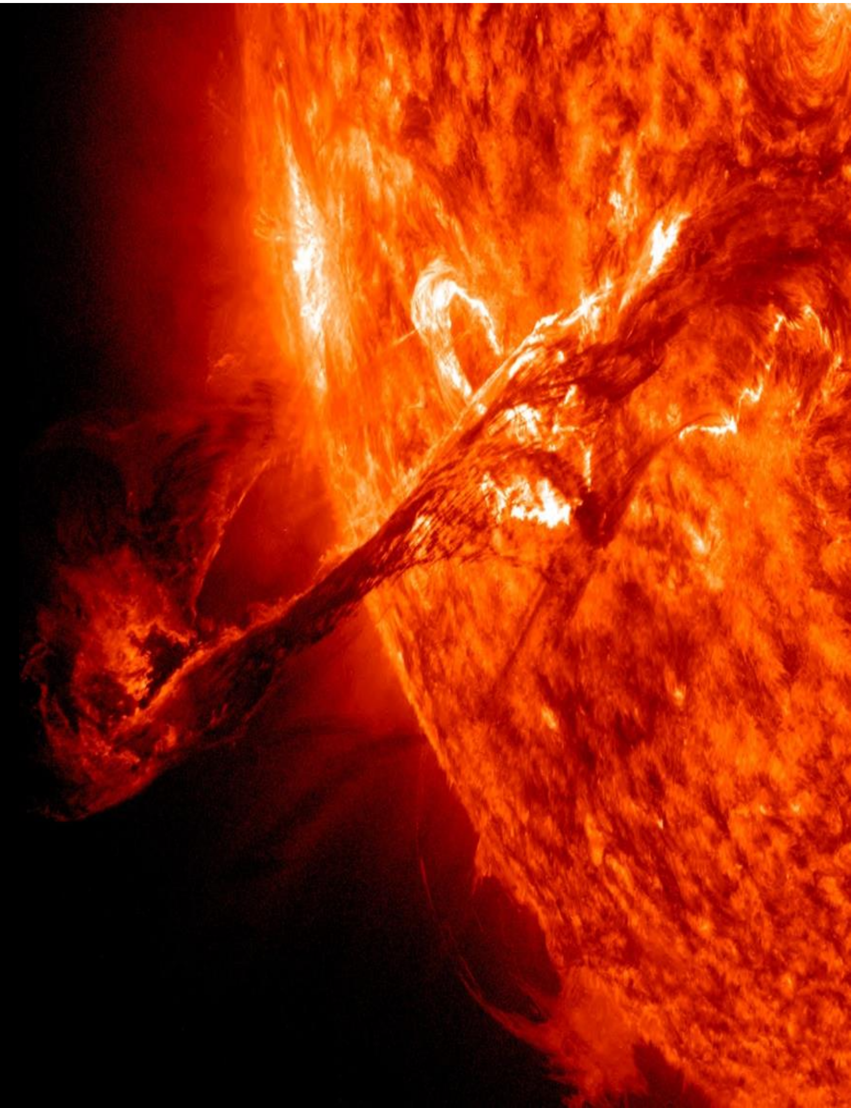
  iv. **planetary environment** (for lander) subjecting the spacecraft to the local conditions on the celestial body, commonly, low temperatures

Introduction: Our Solar System

- Earth is the third innermost planet to our Sun, one of the 8 planets of our solar system.
- All the planets lie close to the Earth's orbital plane (the ecliptic) and travel in the same direction as the Sun rotates.
- The Sun is composed of 78% hydrogen, 20% helium and 2% heavier elements by mass.
- The Sun of mass 2 x 10^{30} kg has a radius defined by its photosphere of 696,000 km with a surface temperature of 5,700 K.
- The Earth of mass 6 x 10^{24} kg has a radius of 6,378 km on average. The Sun-Earth average distance is defined as 1 AU=150 million km. Neptune, the furthest planet, lies at 30 AU. Pluto, the largest (known) dwarf-planet in our solar system, lies at 30-49 AU.
- The heliosphere defines a boundary (heliopause) at around 100 AU – Pioneer 10 and Voyager 1 were at around 60 AU in 1996 on their way out of the solar system.

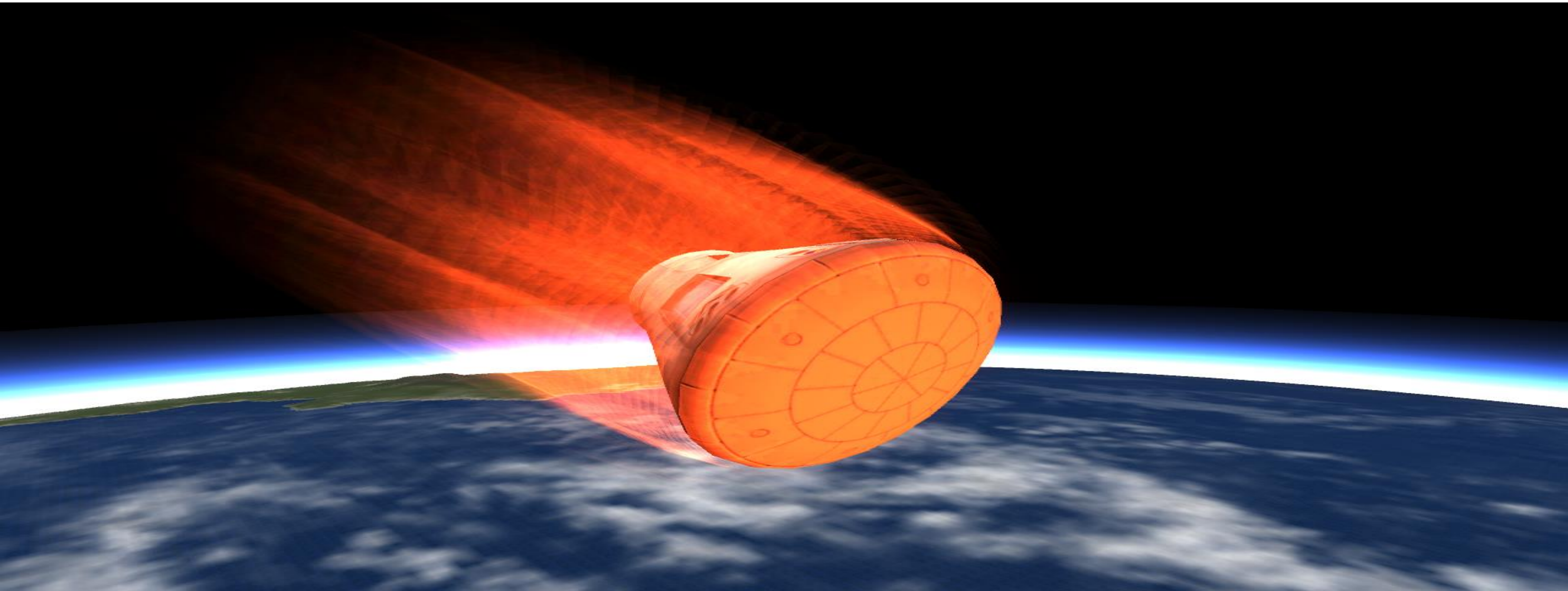
Solar Wind

- In the interplanetary medium, solar wind is the dominant environment but its density drops as $1/r^2$ with distance from the Sun.
- It is generally more benign than the near-Earth environment but still interplanetary probes must pass through the radiation belts.
- Special precautions need to be taken for some planetary missions, e.g. 4 Mrad total dose protection behind 25 mm Al required for Jovian missions (including Europa).



Thermal Environment

- Entry/re-entry environment will impose severely high temperatures to the spacecraft due to atmospheric friction and high deceleration loads. Surface radiative cooling is the dominant mode of cooling during entry/re-entry.
- Typically, special materials are required to absorb the thermal energy (thermal protection systems). Finally, the landing itself will also impose structural loads on the spacecraft.



Space Debris – Διαστημικά Απόβλητα

Space Systems Engineering

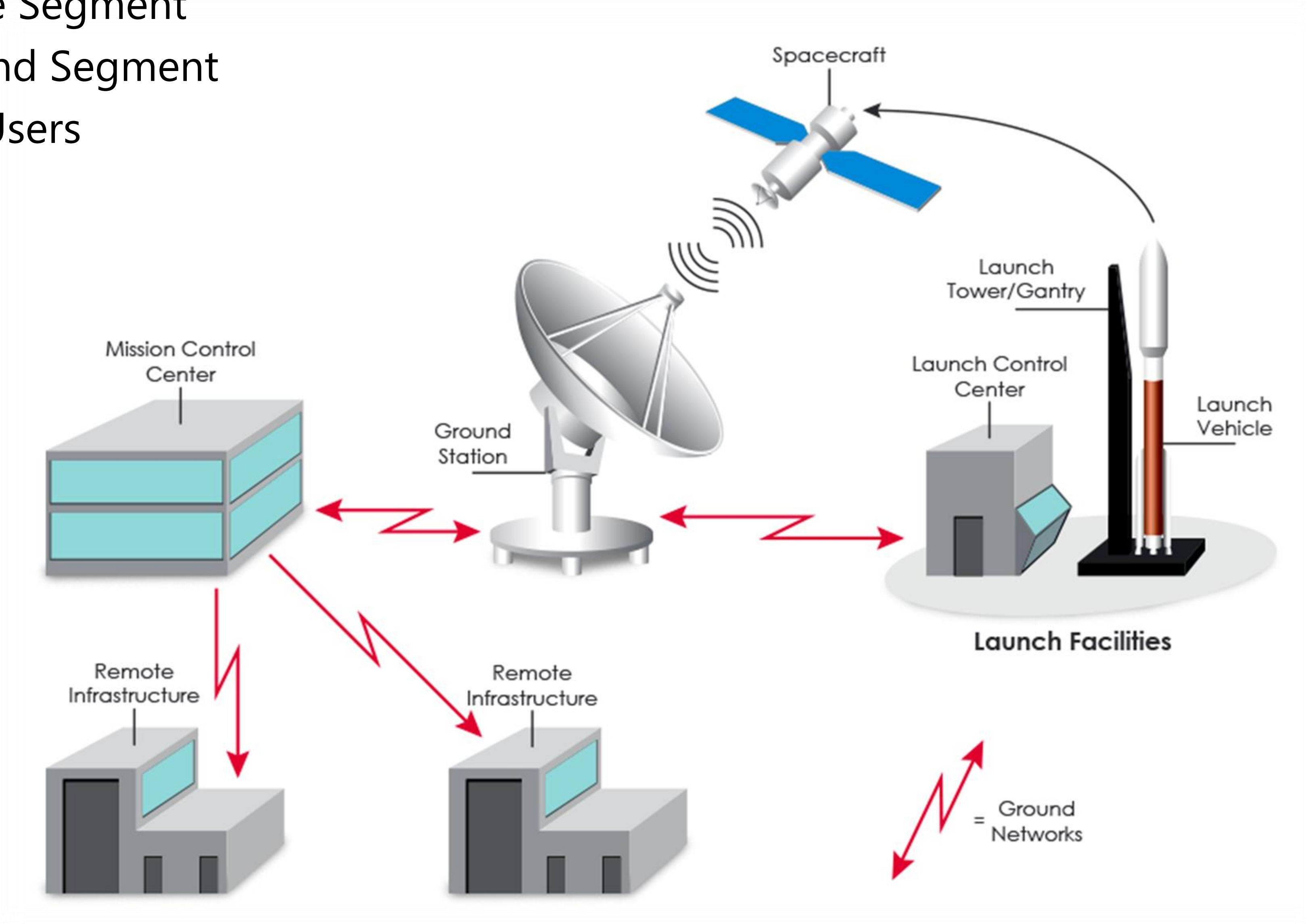
- All space missions are born of a set of requirements – objectives to be fulfilled within certain constraints such as budget and time (the latter sometimes defined by launch windows for exploration missions.
- The requirements of the mission must be well-defined and concise.
- Systems engineering plans and integrates technical solutions within the schedule and within budget. Much of the methodology resembles that from software engineering.
- The mission analysis should describe the mission, its operations, system configuration, subsystem specifications, quality assurance and reliability.
- The specification should flow-down from systems to subsystems to components to parts at increasing levels of detail to ensure consistency.

Space Mission Design

- In designing a space mission:
  - mission objectives must be clearly defined (requirements definition)
  - mission users must be defined (e.g. scientific community)
  - resource availability must be defined (typically budget, political considerations and technological maturity)
  - design constraints must be recognized (schedule, mass and power typically)
- Mission architecture must be defined on the basis of system level trade-off studies to achieve a given performance.

Space Systems

- Space Segment
- Ground Segment
- End Users



Space Mission Architecture

- From the mission requirements and constrains, an iterative design based on trade-off analysis is developed – often altering the mission requirements if necessary.
- All space mission comprise several major systems defining its architecture:
  - spacecraft payload which performs the function of the mission
  - spacecraft bus which supports the payload (housekeeping)
  - launcher to place the spacecraft into its required orbit
  - orbital trajectory which defines the ground coverage
  - ground system which controls the mission operations through a communications infrastructure

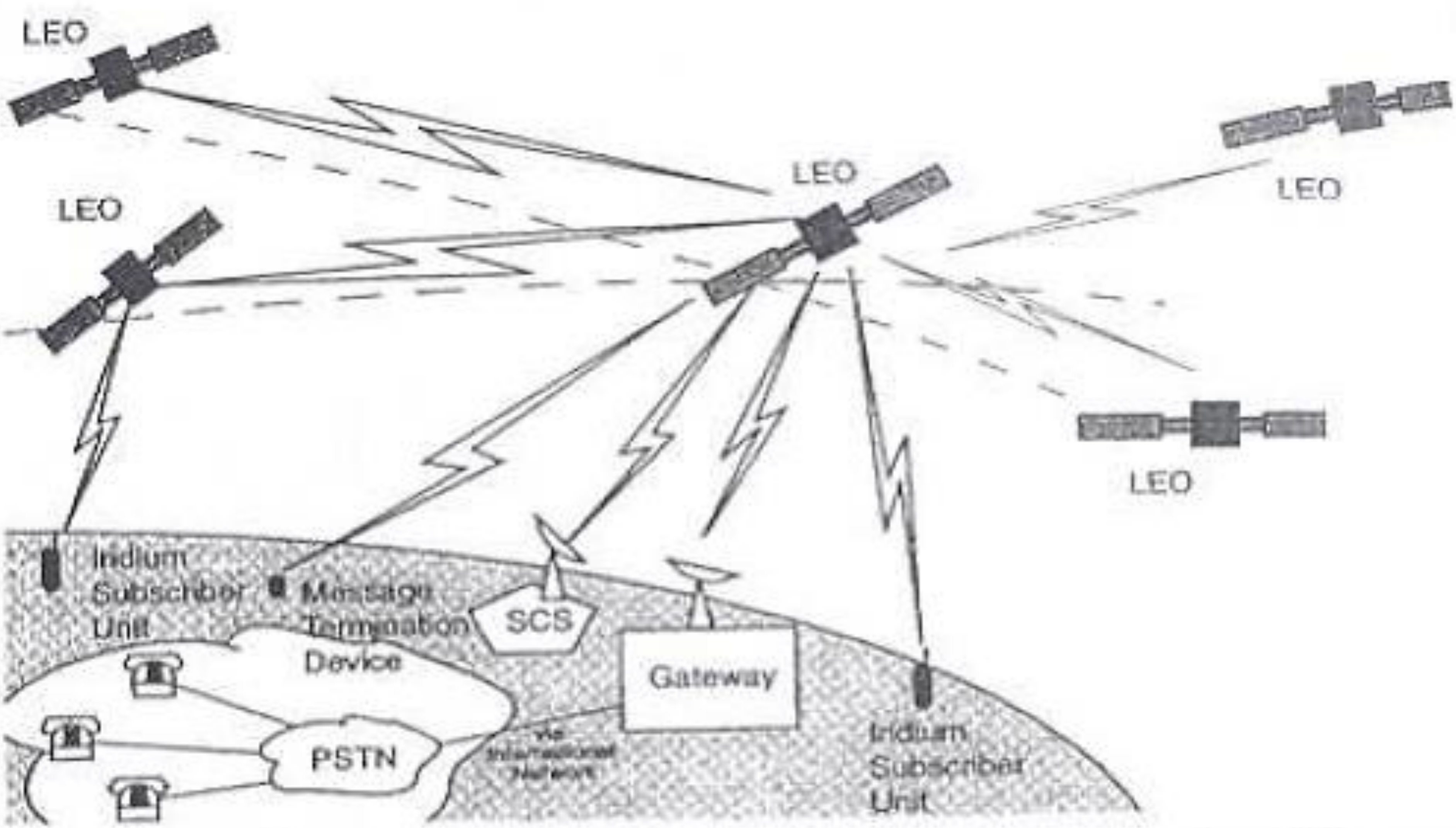
Space Mission

- Space mission design is an iterative process.
- All space missions have a well-defined programme of development with regular reviews with the client, e.g. ESA.
- Reviews provide independent, critical assessment and provide a forum for communication.
- It also ensures that documentation is clear and concise.
- ESA standards are described in ECSS (European Cooperation for Space Standardisation) standards documents (e.g. good soldering practice).

Space Missions I

- A space mission requires people working together in space and the ground.
- Space Segment is the most ‘exciting’.
- Should be reliable, simple, affordable and safe (especially with astronauts in space).
- Ground segment (mission control) is usually undermined but is very important.
- Space segment becoming more complicated: Space Stations, constellations of spacecraft, space tourists….
  - Becoming time critical.
  - Simultaneous missions, different spacecraft.

Space Missions II



Space Missions III

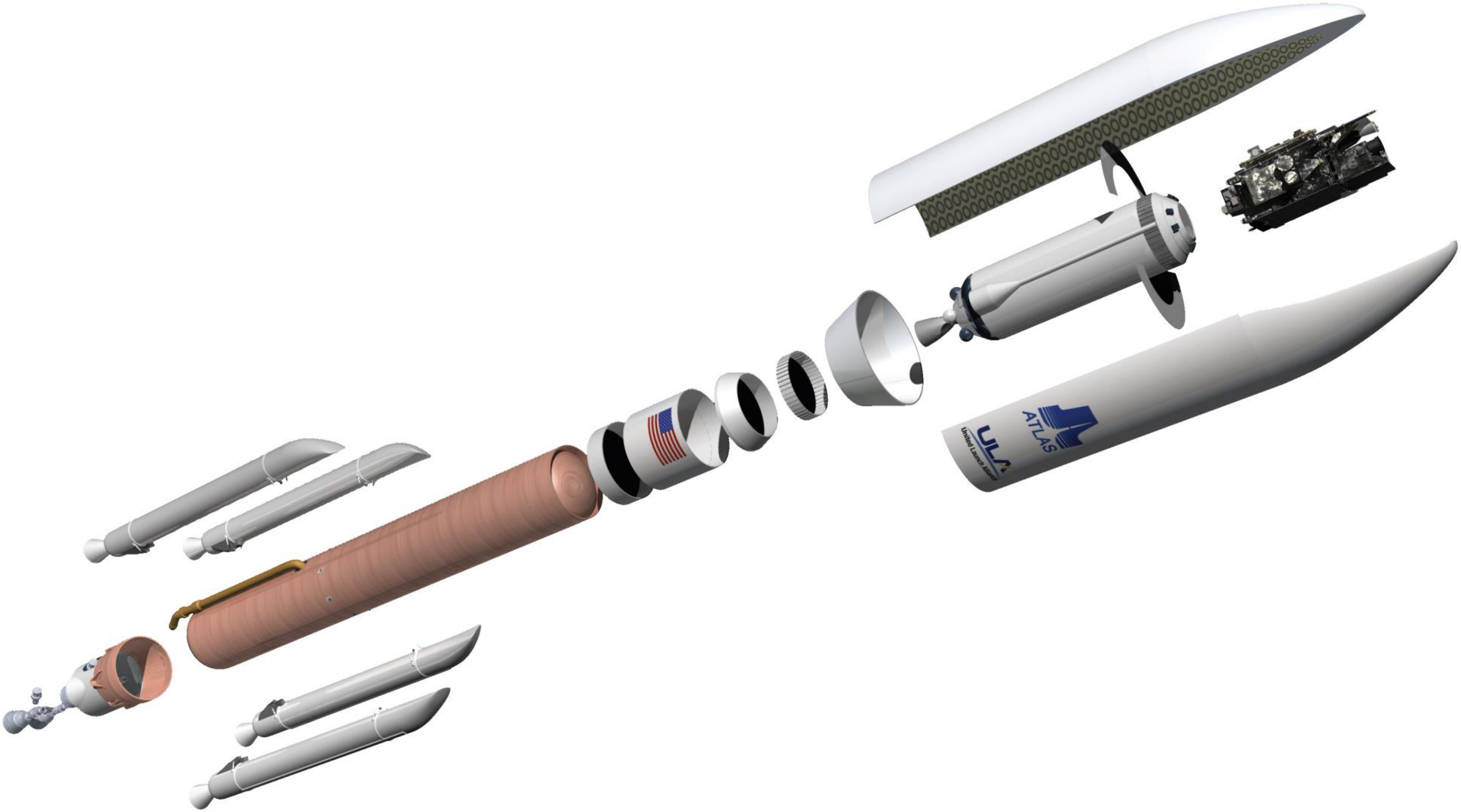
- Launcher Segment
  - Connects the ground and space segment
  - Usually most expensive and with highest risk

Phase	Input	Output	Main Actors
O/A	Mission Statement	Mission Architecture	User + Implementation Manager
B	Mission Architecture	Detailed Design	Implementation Manager + (Industrial) Builder
C/D	Detailed Design	Ground and Space Hardware and Software + Launch	Industrial Builder + Implementation Manager
E	Space and Ground Segment	Data/Service	User

Systems Architecture

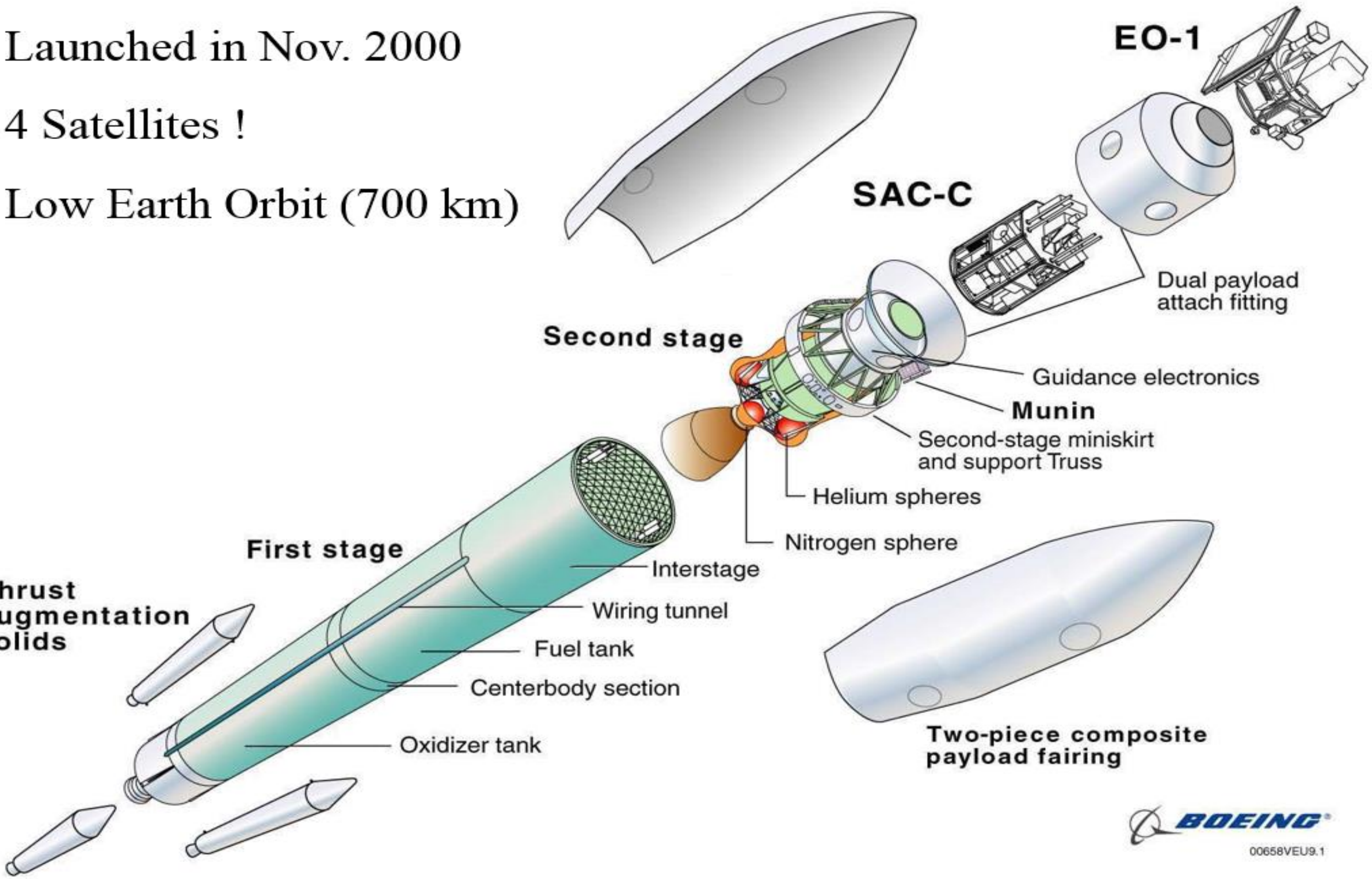
- The systems architecture is the end product of the mission design process consisting of an overall system design, and covering all elements of the system with the necessary specifications to meet the stated mission objectives in an optimum way.
- The systems architecture does not enter into the design of the individual elements any further than is required to establish its functional, cost and schedule feasibility in accordance with the corresponding assumptions included in the systems plan.
- The systems architecture establishes clearly the mutual dependence of the various systems elements, thus providing a complete and structured framework of interdependency formulas for the requirements and characteristics of the various elements.
- Examples: Apollo Program, Mars Exploration, Space Station

Launch Vehicles



Example of a Typical Launch Vehicle (Boeing Delta II)

- Delta 2 LV
- Launched in Nov. 2000
- 4 Satellites !
- Low Earth Orbit (700 km)



Rocket Basics I

- Material today is mostly taken from Chapter 14 ‘Understanding Space’ by Jerry Sellers
- A rocket (space transportation, launch vehicle) is s’thing that ‘spits out’ hot gasses
- Some sort of chemical reaction occurs in closed or sealed volume (tank) that causes the escape of a gas with a high speed
- Similar to a balloon

- What makes a balloon go?

Rocket Basics II

- Newton′s Third Law: For every action there is an equal but opposite reaction
- Skin of the balloon is stretched tight, squeezing the air inside so it has a high pressure than than the air in the room
- To equalise this pressure, the air is forced out the stem
- Following Newton′s 3rd law, the air is forced out (action) an equal force pushes the balloon in he opposite direction (the reaction)
- Balloon flies wildly because its unstable
- With some form of ′control′ like adding fins we can have a small rocket!

Rocket Basics III

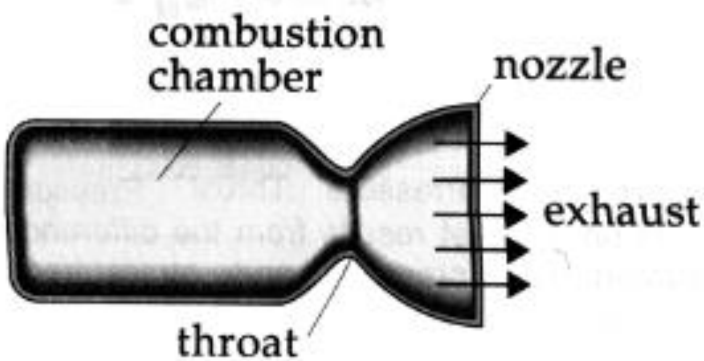
- Consider a student sitting on a wagon armed with a bag of tomatoes aiming to his lecturer
- If he is at rest and begins to throw tomatoes to his lecturer because of Newton's 3rd law an equal but opposite force will move him (and the wagon and tomatoes) in the other direction
- The student had to use some force to throw the tomatoes
- The force moving him on the wagon is identical in magnitude but opposite in direction
- From the concept of conservation of linear momentum:
  - Change of speed of tomato (because its small) will be grater than the change in speed of the wagon.

Rocket Basics IV

- This is how a rocket works: it expands energy to eject mass out the back at high speeds, pushing the rocket in the opposite direction
- Mass is being ejected at a rate we refer to as *mass flow rate*:

$$\Delta m / \Delta t = \dot{m} \ (kg/s)$$

- A rocket engine has three basic parts: combustion chamber, a throat and a nozzle



Rocket Basics V

- The combustion chamber is where the propellant burns, producing hot gasses
- The throat is designed to constrict the flow of hot gasses which will build up the combustion chamber, controlling the chamber pressure and mass flow rate
- The hot gasses reach the speed of sound in the throat
- The nozzle is designed to efficiently direct the flow of gasses in the desired direction.
- The gasses have mass and mve out the back at some exit velocity, V_{exit}
- For typical chemical rockets like those used by the Space Shuttle, V_{exit} can be as high as 3 km/s

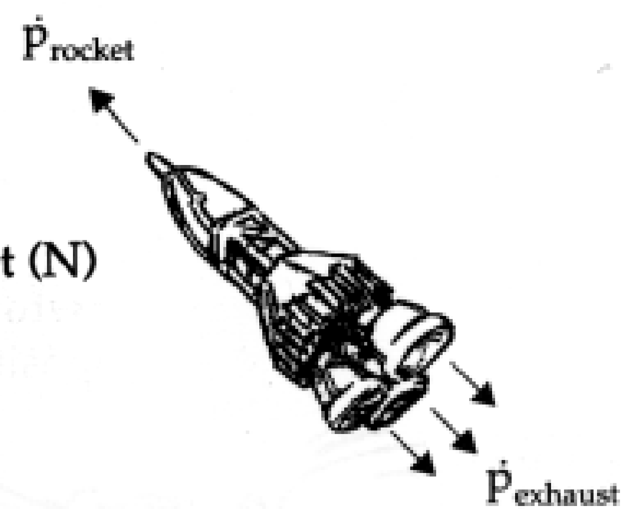
Rocket Basics VI

- Linear Momentum: mass x velocity
- This means that the ejected mass acquires momentum at a rate of $\dot{m}V_{exit}$
- Momentum is always conserved
- As the momentum of the ejected mass increases in one direction, the momentum of the rocket must likewise be increasing in the other direction.

$$\dot{p}_{rocket} = \dot{p}_{exhaust}$$

$$\boxed{\dot{p}_{rocket} = \dot{m}V_{exit}}$$

$\dot{p}_{rocket}$ = time rate of change of momentum of the rocket (N)
$\dot{m}$ = mass flow rate of exhaust products (kg/s)
V_{exit} = exit velocity of exhaust (m/s)



Rocket Basics VII

- We consider the momentum change to be equivalent to a force on the rocket that we'll call *momentum thrust*

$$F_{\text{momentum thrust}} = \dot{m} V_{\text{exit}}$$

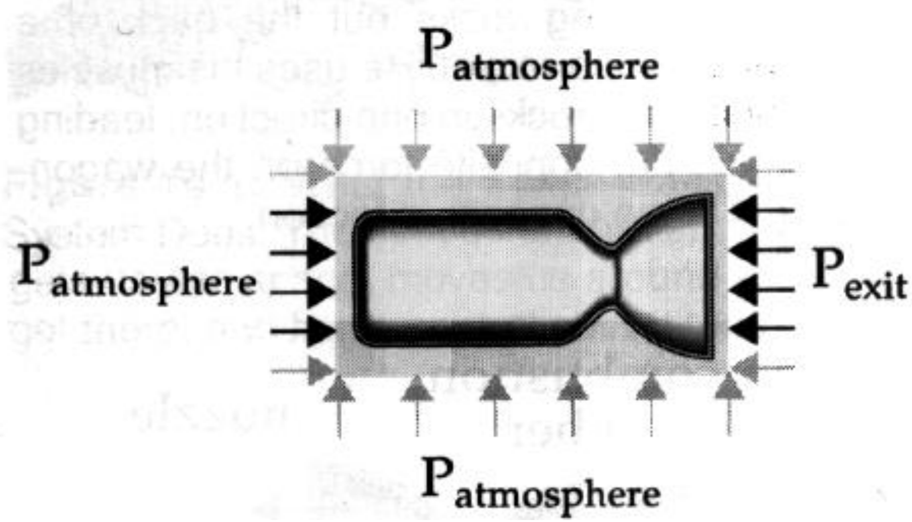
$F_{\text{momentum thrust}}$ = effective thrust on the rocket from momentum change (N)
$\dot{m}$ = mass flow rate of exhaust products (kg/s)
V_{exit} = exit velocity of exhaust products (m/s)

- Change in momentum is not the only consideration when determining the total thrust on a rocket.
- Lets consider a 'control volume' drawn around a rocket
- Consider the atmospheric pressure acting on all sides except at the nozzle ($P_{\text{atmosphere}}$)
- Due to symmetry the atm. pressure cancels out everywhere except in the direction parallel to momentum thrust over an area equal to the nozzle are A_{exit}

Rocket Basics VIII

- The net force exerted on the rocket from this pressure differential is called thrust
- Its equal to the difference between exit pressure, P_{exit}, and atmospheric pressure, $P_{atmosphere}$, times the exit area, A_{exit}

$$F_{\text{pressure thrust}} = A_{exit}\,(P_{exit} - P_{atmosphere})$$



Rocket Thrust, Exhaust Velocity

where

$F_{pressure\ thrust}$ = pressure thrust (N)
A_{exit} = exit area of nozzle (m^2)
P_{exit} = exit pressure (N/m^2)
$P_{atmosphere}$ = atmospheric pressure (N/m^2)

$$\boxed{F_{thrust} = \dot{m}V_{exit} + A_{exit}(P_{exit} - P_{atmosphere})}$$

where

F_{thrust} = total rocket thrust (N)
$\dot{m}$ = mass flow rate (kg/s)
V_{exit} = exit velocity of exhaust products (m/s)
A_{exit} = exit area of nozzle (m^2)
P_{exit} = exit pressure (N/m^2)
$P_{atmosphere}$ = atmospheric pressure (N/m^2)

$$\boxed{c \equiv V_{exit} + \frac{A_{exit}}{\dot{m}}(P_{exit} - P_{atmosphere})}$$

c = effective exhaust velocity (m/s)
V_{exit} = exit velocity of exhaust products (m/s)
A_{exit} = exit area of nozzle (m^2)
$\dot{m}$ = mass flow rate (kg/s)
P_{exit} = exit pressure (N/m^2)
$P_{atmosphere}$ = atmospheric pressure (N/m^2)

Total Thrust or Rocket Equation

$$\boxed{F_{thrust} = \dot{m}c}$$

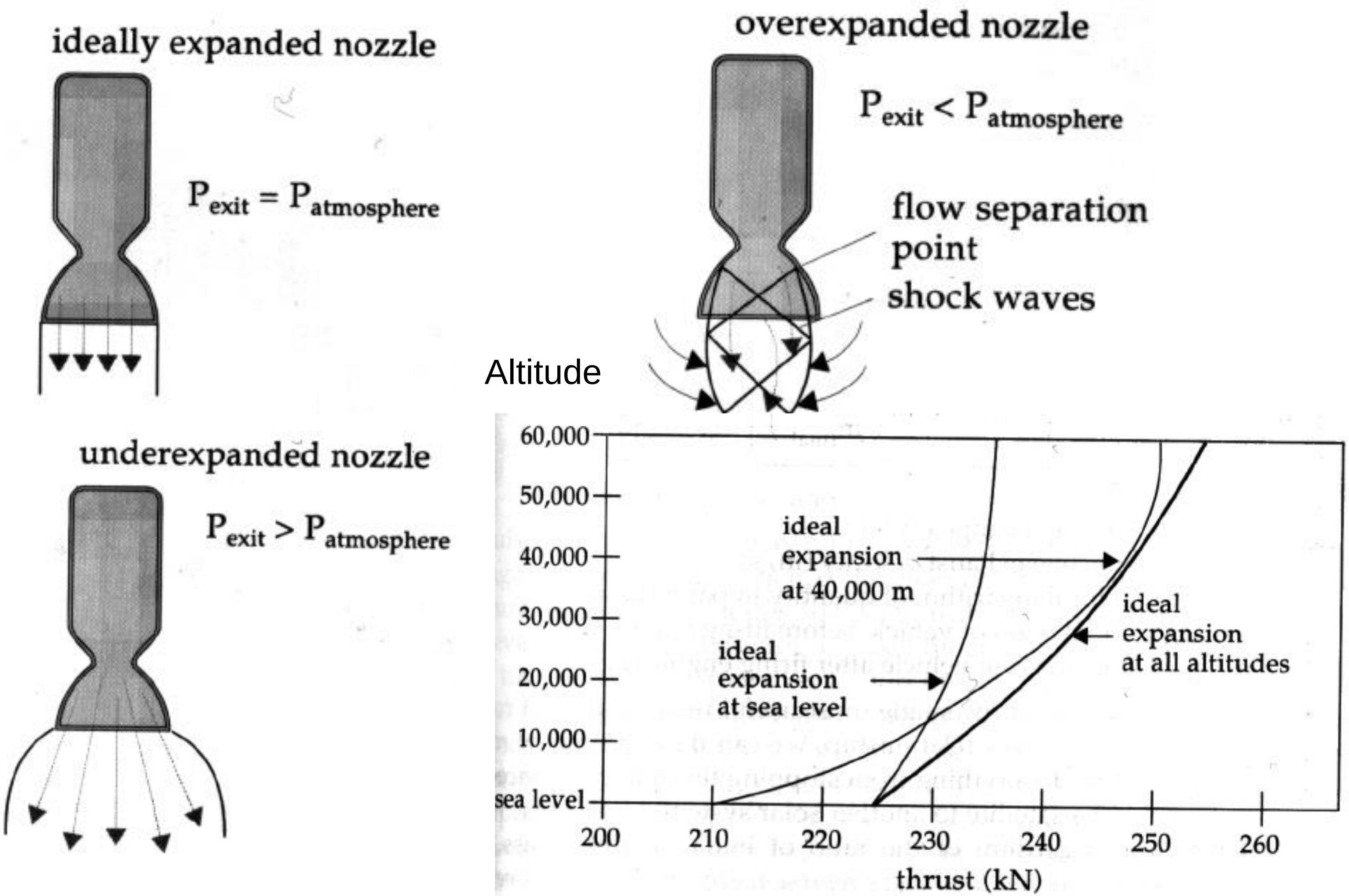
F_{thrust} = total rocket thrust (N)
c = effective exhaust velocity (m/s)
$\dot{m}$ = mass flow rate (kg/s)

- Our student/tomato example:
  - Student can increase the thrust on the wagon by either increasing the rate at which he throws out the tomatoes (higher $\dot{m}$) or throwing the tomatoes faster (higher c). Or both !
- To get a rocket off the ground, the total thrust must be greater than the weight of the entire vehicle
- The ratio of the thrust produced is referred as *thrust-to-weight* ratio (Atlas rocket 1.2, Space Shuttle 1.6)

G-load

- Astronauts feel at the top of a rocket the thrust force as and acceleration or g-load
- A rocket with a constant thrust will make astronauts feel a g-load or acceleration which depends on the rocket's mass
- This mass decreases as propellant burns and is ejected
- This means acceleration will tend to increase over time.
- The overall g-load is kept on the Space Shuttle under 3-g's by throttling down the engines about six minutes into the launch in order to decrease their thrust as propellant burns

Nozzles



ideally expanded nozzle
$P_{exit} = P_{atmosphere}$
overexpanded nozzle
$P_{exit} < P_{atmosphere}$
flow separation point
shock waves
underexpanded nozzle
$P_{exit} > P_{atmosphere}$
Altitude
60,000
50,000
40,000
30,000
20,000
10,000
sea level
200
210
220
230
240
250
260
thrust (kN)
ideal expansion at 40,000 m
ideal expansion at sea level
ideal expansion at all altitudes

The Ideal Rocket Equation (I)

- When you take a long trip in your car one of your main concerns is having enough gas in you tank
- Sam goes for space; how do we determined ‘gas’ for space?
- For rockets the mass expelled out the back is referred to as *propellant*

$$F_{thrust} = \dot{m}c = \frac{\Delta p_{rocket}}{\Delta t}$$

F_{thrust} = effective thrust on the rocket (N)

$\dot{m}$ = mass flow rate (kg/s)

c = effective exhaust velocity (m/s)

$\frac{\Delta p_{rocket}}{\Delta t}$ = time rate of change of momentum for the rocket (N)

The Ideal Rocket Equation (II)

$$\Delta V = c \ln\left(\frac{m_{initial}}{m_{final}}\right)$$

ΔV = velocity change (m/s)
c = effective exhaust velocity (m/s)
$\ln$ = natural logarithm of quantity in parenthesis
$m_{initial}$ = initial mass of vehicle before firing engine (kg)
m_{final} = final mass of vehicle after firing engine (kg)

- One of the most important equations
- We can determine how much propellant we need to do anything from stopping a spacecraft spinning in orbit to launching a spacecraft to another solar system.

Total Impulse

- From Newton's 2nd law:

$$F = \frac{\Delta p}{\Delta t}$$

- Multiply both sides with Δt: $F\Delta t = \Delta p$
- In order to achieve some change in momentum we can use a large force acting over a short time or a smaller force acting over a longer time:

$$\boxed{I \equiv F\ \Delta t = \Delta p}$$

I = total impulse $(N \cdot s)$
F = force (N)
Δt = time (s)
Δp = momentum change $(N \cdot s)$

Specific Impulse (I)

- Specific Impulse I_{sp} is the ratio of the total impulse to the rocket's change in weight as propellant is ejected
- It is used to compare the performance of different types of rockets
- Using weight rather than mass to calculate I_{sp} is simply a convention established by the founders of rocket science and, as a result, I_{sp} has the unusual units of seconds

$$\boxed{I_{sp} \equiv \frac{I}{\Delta W}}$$

where

I_{sp} = specific impulse (s)

I = total impulse (N · S)

ΔW = change in weight (N)

Specific Impulse (II)

- Substituting for total impulse:

$$I_{sp} = \frac{F_{thrust}\,\Delta t}{\Delta W}$$

$$\boxed{I_{sp} = \frac{F_{thrust}}{\dot{W}}}$$

where
I_{sp} = specific impulse (s)
F_{thrust} = force of thrust (N)
$\dot{W}$ = weight flow rate (N/s)

- I_{sp} represents the ratio of what you get (momentum change) to what you spend (propellant)
- The bigger the I_{sp} the more efficient the rocket
- A more useful equation is:

Specific Impulse (III)

$$I_{sp} = \frac{F_{thrust}}{\dot{m} g_o}$$

$\dot{m}$ = mass flow rate
g_o = gravitational acceleration constant = 9.81 m/s^2 (sea level)

- Using: $\boxed{F_{thrust} \doteq \dot{m} c}$ we get:

$$\boxed{I_{sp} = \frac{c}{g_o}}$$

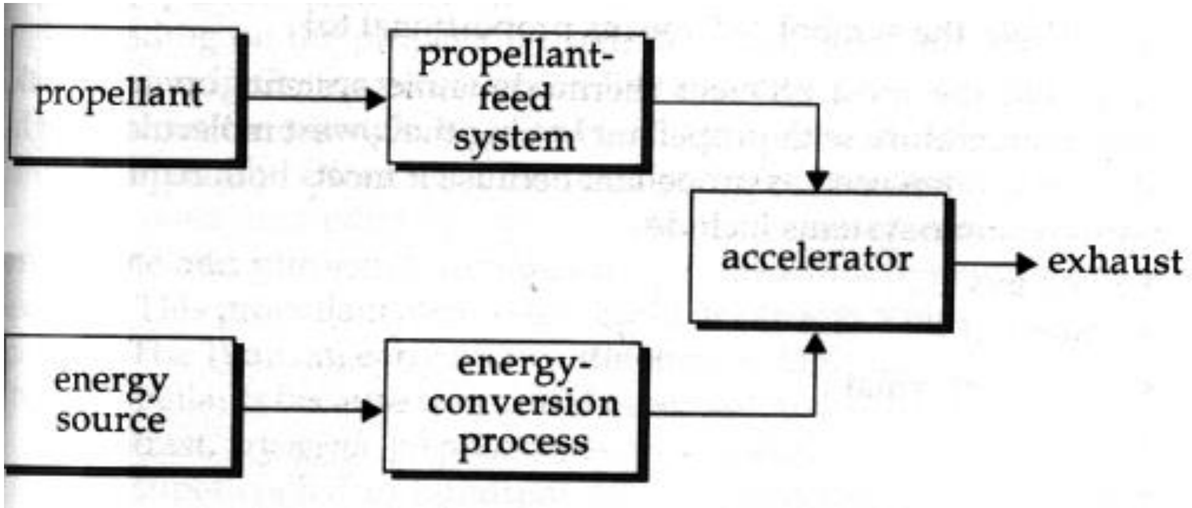
I_{sp} = specific impulse (s)
c = effective exhaust velocity (m/s)
g_o = gravitational acceleration at sea level (9.81 m/s^2)

- Note that g_0 is a constant value representing the acceleration due to gravity at sea level.
- This means no mater where we go in the universe we humans will use the same value of g_0 to measure engine performance

Rocket Systems (I)

- Rockets use an energy source to accelerate mass of propellant to high velocity and expel it out the back producing thrust
- To do this some form of energy conversion process takes place converting energy to acceleration
- For example burning hydrogen and oxygen converts chemical energy to thermal energy
- A propellant-feed system moves propellant from the storage tanks to the accelerator
- For liquid propellants this is done using pumps and other plumbing

Rocket Systems (II)



- Thermodynamics systems: create thrust by converting thermal energy (heat) into kinetic energy (mass moving at high velocity)
- Electrodynamics systems: create thrust by using electric or magnetic fields to accelerate charged particles and give them kinetic energy
- Exotic systems: create thrust through some means other than thermodynamic and electrodynamics impulse (work in theory)

Thermodynamic Propulsion (I)

- Thermodynamics systems: create thrust by converting thermal energy (heat) into kinetic energy (mass moving at high velocity)

- They usually burn something (chemical reaction) to produce exhaust products with high thermal energy

- A nozzle converts thermal energy in the exhaust products into organised kinetic energy in one direction to propel the rocket

- Efficiency of a thermodynamics system (I_{sp}) depends on the combustion temperature and the propellant's molecular weight

- Molecular weight is a measure of the weight per molecule of propellant

- To improve efficiency, we increase combustion temperature or decrease the molecular weight (hydrogen is the lightest element)

Thermodynamic Propulsion (II)

$$\boxed{I_{sp} \propto \sqrt{\frac{T_{combustion}}{M}}}$$

I_{sp} = specific impulse (s)
$T_{combustion}$ = Combustion temperature
M = molecular weight
[*Note:* the symbol "$\propto$" means proportional to]

- As a result the most efficient thermodynamic systems operate at the highest temperature having the lowest molecular weight
- Hydrogen is often used as propellant because it meets both requirements

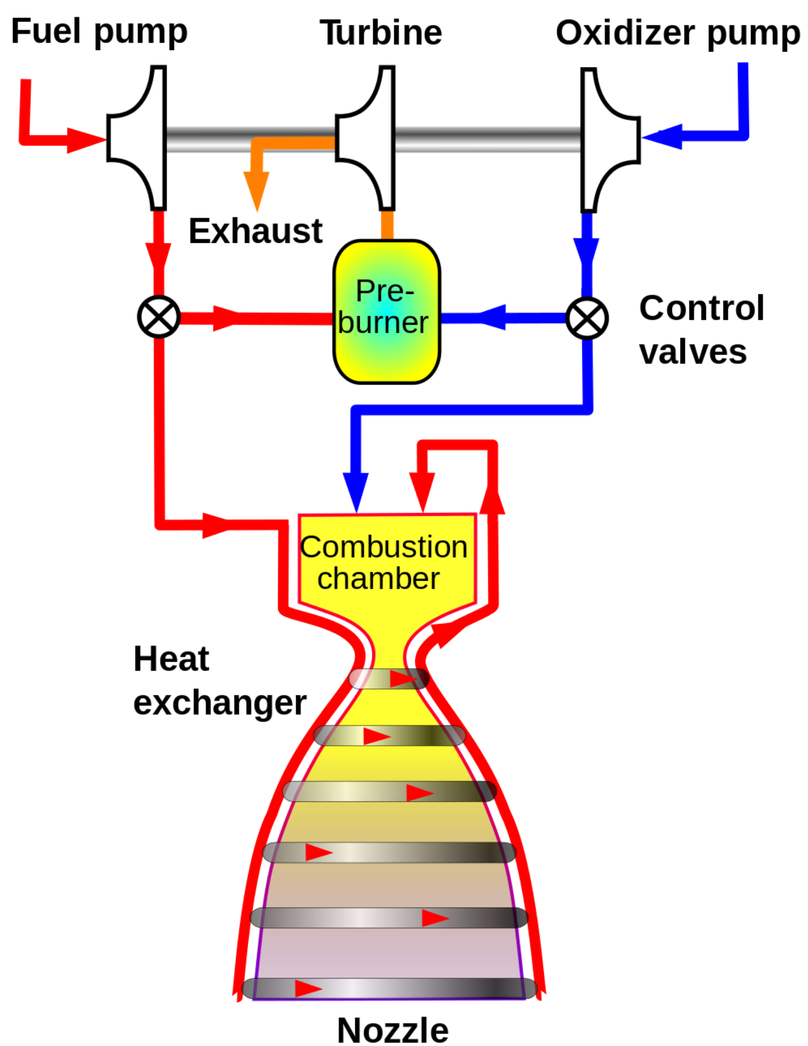
Thermodynamic Propulsion (II)

- Thermodynamic systems include:
    - Cold gas, chemical, nuclear thermal, arc-jet, resistojet
- Cold gas is the simplest one, work like a balloon, no combustion chamber needed, very reliable and are switch-able (on/off), produce small pulses at precise levels (e.g. manned manoeuvring unit on astronauts of the space shuttle)
- Liquid propulsion systems:
    - Simplest is a monopropellant, a catalyst is used to decompose the monopropellant such as hydrazine and expel by-products of decomposition via a nozzle
    - More complex are rockets using 2 propellants: biprop's

Chemical Propulsion

- Bi-prop's:
  - They combine a fuel such as liquid hydrogen (LH_2) and an oxidiser such as liquid oxygen (LOX) in a combustion chamber, thus they chemically react to form heat plus reactant products
  - In order to do this fuel and oxidiser must be stored in tanks and moved in the chamber and injected in the right proportion
  - In small rockets we force the propellant into the combustion chamber using an inert gas such as helium or nitrogen
  - For large engines such as the space shuttle we use powerful pumps which are driven by turbines fuelled by burning the same oxidiser and fuel used by the engines

Pump-Fed Liquid Rockets



How a Rocket Works

Πως Εκτοξεύονται οι Δορυφόροι?

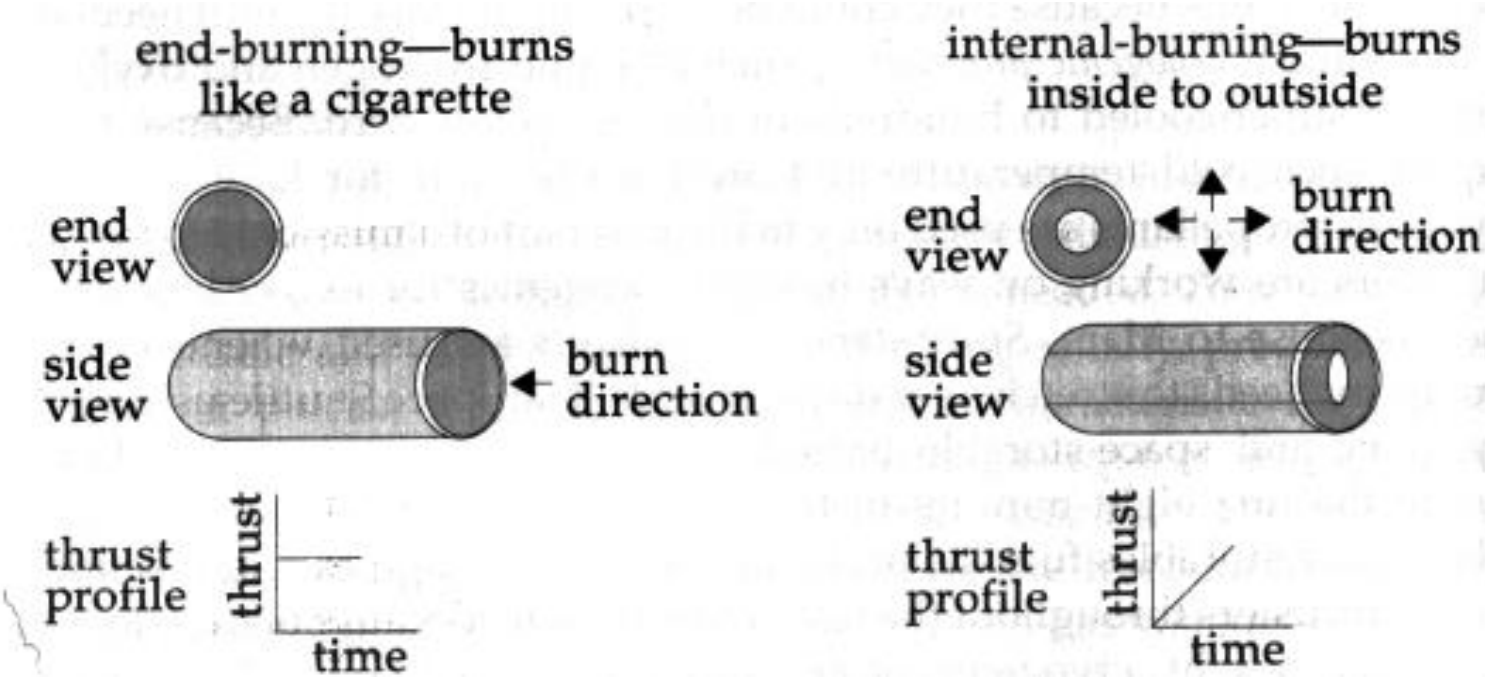
Hypergolic and Cryogenic Propellants

- Other systems use hypergolic propellant- the fuel and oxidiser react immediately upon contact so no ignition is required
- Hydrazine and nitrogen tetroxide are examples of a hypergolic fuel and oxidiser
- This propellant is storable at near room temperature for long periods used on ballistic missiles (ICBMs) because they could be stored on a missile when needed
- Cryuogenic propellants such as liquid hydrogen and oxygen must be super-cooled to hundreds of degrees below zero
- They are hard to store for long periods and are used for the initial phase of a mission
- Space shuttle uses cryogenic propellant for its launch and storable propellants for attitude/orbit control maneuvers.

Solid Rocket Motors (I)

- Fireworks: developed by the Chinese thousands of years ago
- Solid rocket is called a motor because it is self contained: its consists of the propellant (solid) fitted ina container along with an igniter and a nozzle
- Because they are simple SRM are inherently more reliable and cheaper to produce than liquid engines BUT they have LOW I_{sp}
- SRM consist of a mixture of fuel and oxidiser blended together in the right proportion and then solidified
- The larger the surface area burning the faster the propellant will be used up and the higher the thrust
- The simplest type of burning profile is like a cigarette- producing a flat thrust profile.

Solid Rocket Motors (II)



- Space Shuttle SRMs have a core shape designed so the thrust can be decreased 55s into the flight to lower aerodynamic forces on the vehicle.

Type	Typical Propellants	Thrust (N)	I_{sp} (s)	Advantages	Disadvantages
Cold Gas	N_2 He H_2	0.2 – 2700	60 – 225	• Very simple • Ideal for attitude control • "Unlimited" cycling	• Low I_{sp} • Redundancy required
Liquid	H_2/LOX Kerosene/LOX N_2H_4 N_2H_4/N_2O_4	$10 - 6 \times 10^6$	270 – 530	• Can throttle • Very high thrust	• Complex • Expensive • Explosive hazard
Solid	Al/NH_4ClO_4 Asphalt/NH_4ClO_4 HTPB* binder	$0.1 - 1.2 \times 10^7$	200 – 300	• Very simple • Very high thrust • Inexpensive • Storable	• Can't throttle • Can't stop/restart • Difficult to vector thrust • Significant explosive hazard
Hybrid	Plexiglass/LOX HTPB*/LOX	$10 - 3.0 \times 10^5$**	250 – 350	• Can throttle • Simple • Non-explosive • Environmentally safe	• Difficult to build very high-thrust motors • Lower I_{sp} than liquid systems
Nuclear Thermal	H_2	$3000 - 6 \times 10^5$	700 – 1100	• High I_{sp} • High thrust • Can cycle • Can throttle	• Nuclear contamination • Shielding required • Lower thrust-to-weight than chemical systems
Arcjet	H_2 NH_3	0.05 – 40	500 – 2500	• High thrust (electric) • High I_{sp}	• Limited life-span • High power requirement
Resistojet	NH_3 H_2	$5 \times 10^{-4} - 10$	250 – 900	• Simple • Long life	• High power requirement • Lowest I_{sp} electric system
Ion	Cs Hg Xe C_{60}	0.02 – 2.0	5,000 – 10,000	• Very high I_{sp} • Very long useful life • "Unlimited" cycling	• Very low thrust • High power requirement
Plasma	H_2 Ar	0.2 – 200	2,000 – 10,000	• Relatively high thrust • Very high I_{sp}	• Limited life-span • High power requirement

Staging (I)

- Rockets/boosters have in common (most of the time) two characteristics:
  - Chemical rockets systems
  - Multiple stages
- Chemical rockets aren't as efficient as some options discussed in previous slides they offer higher thrust and more importantly very high thrust to weight ratios
- Only chemical rockets can produce thrust to weight ratios > 1
- When you see a rocket what you see is the tank (cylinder)
- Why carry all that extra tank weight along once part of the propellant has been used?
- Why not split the propellant into smaller tanks and drop them when they are empty like planes (bombers) do?
- That concept is called staging

Staging (II)

- Following example shows how staging can increase the amount of payload delivered in orbit
- Payload mass is very small 5% of total mass
- The other 15% is structure avionics etc.

Booster	Parameters	Payload to Orbit
Single Stage  $m_{payload} = 84$ kg $m_{structure} = 250$ kg $m_{propellant} = 1500$ kg engine $I_{sp} = 480$ s	$\Delta V_{design} = 8000$ m/s $I_{sp} = 480$ s $m_{structure} = 250$ kg $m_{propellant} = 1500$ kg	$m_{payload} = 84$ kg
Two Stage  $m_{payload} = 175$ kg $m_{propellant} = 750$ kg $m_{structure} = 140$ kg engine $I_{sp} = 480$ s $m_{propellant} = 750$ kg $m_{structure} = 140$ kg engine $I_{sp} = 480$ s	$\Delta V_{design} = 8000$ m/s **Stage 2** $I_{sp} = 480$ s $m_{structure} = 140$ kg $m_{propellant} = 750$ kg **Stage 1** $I_{sp} = 480$ s $m_{structure} = 140$ kg $m_{propellant} = 750$ kg	$m_{payload} = 175$ kg

Staging (III)

- Use rocket equation for analysis:

$$\Delta V = I_{sp} g_o \ln\left(\frac{m_{intial}}{m_{final}}\right)$$

- Each stage has an initial and final mass
- I_{sp} might be different per stage
- To get the total ΔV we must add the ΔV for each stage

$$\Delta V_{total} = \Delta V_{stage\ 1} + \Delta V_{stage\ 2} + \ldots + \Delta V_{stage\ n}$$

$$\boxed{\begin{aligned}\Delta V_{total} &= I_{sp\ stage\ 1}\ g_o \ln\left(\frac{m_{initial\ stage\ 1}}{m_{final\ stage\ 1}}\right) \\ &+ I_{sp\ stage\ 2}\ g_o \ln\left(\frac{m_{initial\ stage\ 2}}{m_{final\ stage\ 2}}\right) + \ldots \\ &+ I_{sp\ stage\ n}\ g_o \ln\left(\frac{m_{initial\ stage\ n}}{m_{final\ stage\ n}}\right)\end{aligned}}$$

ΔV_{total} = total ΔV from all stages (m/s)
$I_{sp\ stage\ n}$ = specific impulse of stage n (s)
g_o = gravitational acceleration at sea level (9.81 m/s^2)
$m_{initial\ stage\ n}$ = initial mass of stage n (kg)
$m_{final\ stage\ n}$ = final mass of stage n (kg)

Staging (IV)

- What is the initial and final mass of stage 1? Initial is easy, it's the mass of the entire vehicle at lift-off, but how about the final mass?
- Final mass of any stage is the initial mass of that stage (including the mass of subsequent stages) less the mass of propellant burned in that stage, so for stage 1:

$$m_{\text{final stage 1}} = m_{\text{initial vehicle}} - m_{\text{propellant stage 1}}$$

- For stage 2,3...

$$m_{\text{initial stage 2}} = m_{\text{final stage 1}} - m_{\text{structure stage 1}}$$

$$m_{\text{final stage 2}} = m_{\text{initial stage 2}} - m_{\text{propellant stage 2}}$$

- Staging REDUCES the vehicles total weight for a given payload and ΔV requirement

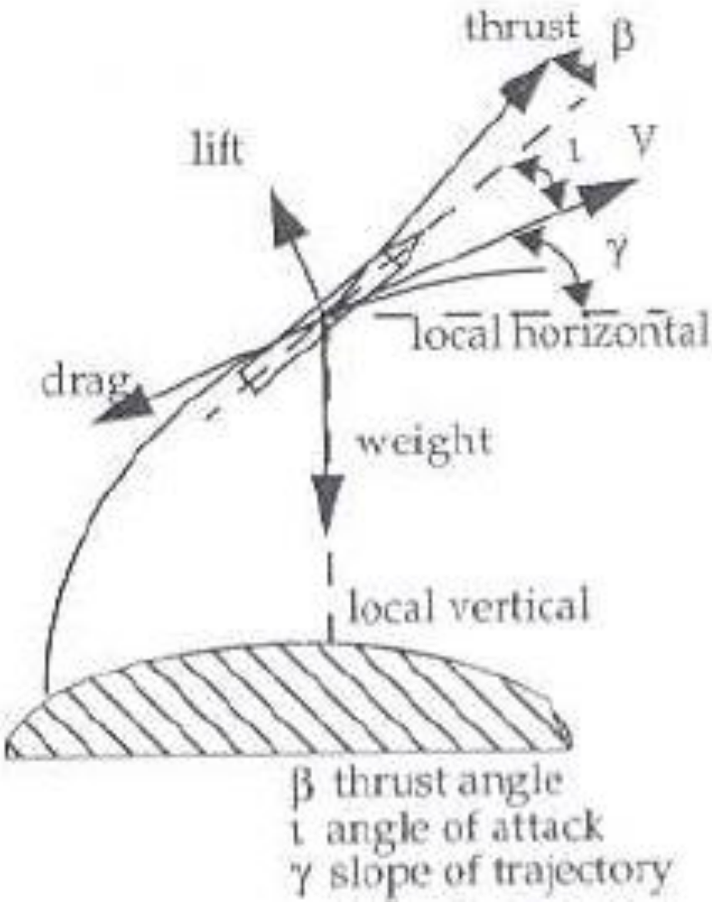
Staging (V)

- Advantages
  - Increases the total payload mass we can deliver to space for the same-sized vehicle
  - Increases the total velocity we can achieve for the same-sized vehicle
  - Decreases the engine efficiency (I_{sp}) required to deliver a given-sized payload to orbit
- Drawbacks (there ain't no such thing as a free lunch or launch!)
  - Increased complexity because of the extra set of engines and plumbing
  - Decreased reliability because we're depending on extra sets of engines and the plumbing to work
  - Increased total cost because more complex vehicles cost more to build

Staging (VI)

- Another interesting limitation of staging has to do with the law of diminishing returns (‘φθίνουσα απόδοση’)
- It may be concluded that if two stages are good, four stages must be good as well, but this is not necessarily true
- Although the 1st added stage significantly improves performance, each additional stage enhances it less
- By the time we add a 4th or 5th stage, the increased complexity offsets the small extra gain in performance. That’s why most boosters have only two or three stages

Forces Acting on a Launcher



$$\Delta V_{design} = \Delta V_{mission} + \Delta V_{losses} = g_0 I_{SP} \ln \frac{M_i}{M_f}$$

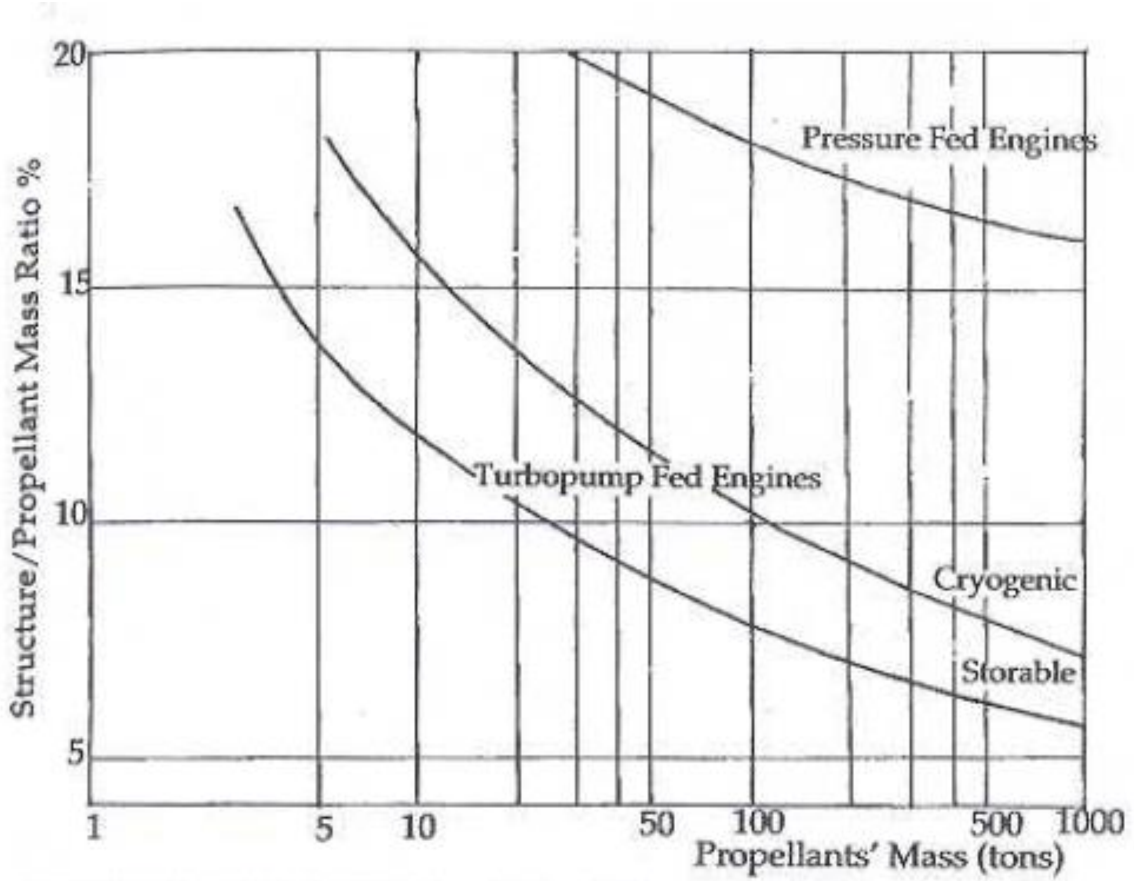
- Velocity Losses are important
- Launch Pad selection

Source	Loss
$\Delta V_{gravity}$	830 m/s
$\Delta V_{guidance}$	800 m/s
ΔV_{drag}	120 m/s
ΔV_{lift}	1 m/s
$\Delta V_{mission}$	~ 8000 m/s

Structural Factor

- Large part of a launch vehicle is taken up by its tanks
- Size and mass depends on the mass of propellant
- Structural Factor S: ratio of the total structure's mass to the propellant's mass
- S decreases as the mass of propellant increases: this results in the *size effect*, a big tank being relatively lighter than a small one
- Pressurized tanks which need thicker walls to stand the pressure lead to heavier launchers

Structural Factor/Staging



- Biggest part of a launcher's mass is represented by it's propellants: typical values of mass for a GTO mission are 90% for propellants, 9% for structure and only 1% for payload

Structural Factor/Staging

I_{SP}=450 s			
		Operation of	
		stage 1	stage 2
Initial mass (t)	200	200	55
ΔV (m/s)	9500	4750	4750
Mass at burn out (t)	23	68	19
Mass of propellant used (t)	177	132	36
Structural factor	0.09	0.10	0.12
Structure mass (t)	16	13	4.3
Payload mass	7	14.7	

- Payload Mass is doubled for the same take-off mass, going from one to two stagεs

Types of Launch Systems

- Human Space Flight
- Unmanned Space Flight
- Expendable/Reusable/Part Reusable
  - Single Stage to Orbit.(SSTO)/ Two Stage to Orbit (TSTO), Multiple Stage to Orbit (MSTO)
  - Payload Mass, Propellant/Propulsion System
  - Civilian/Military, Space Exploration

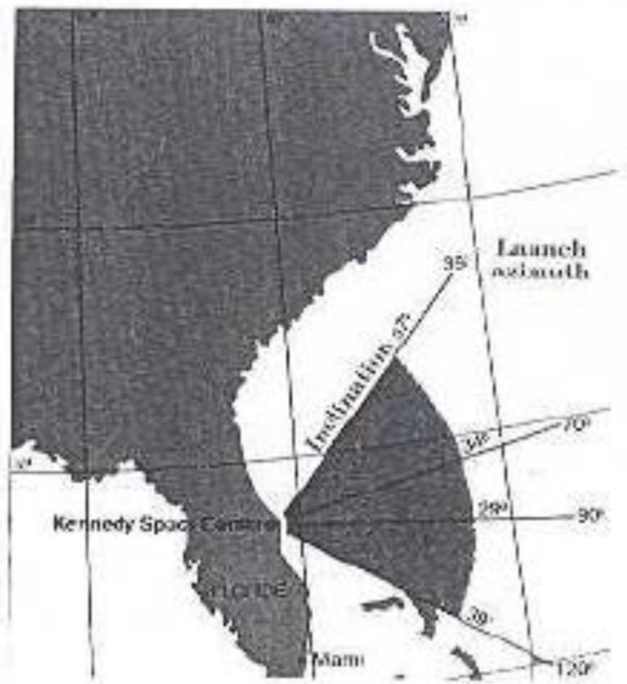
Selection of Launch Systems

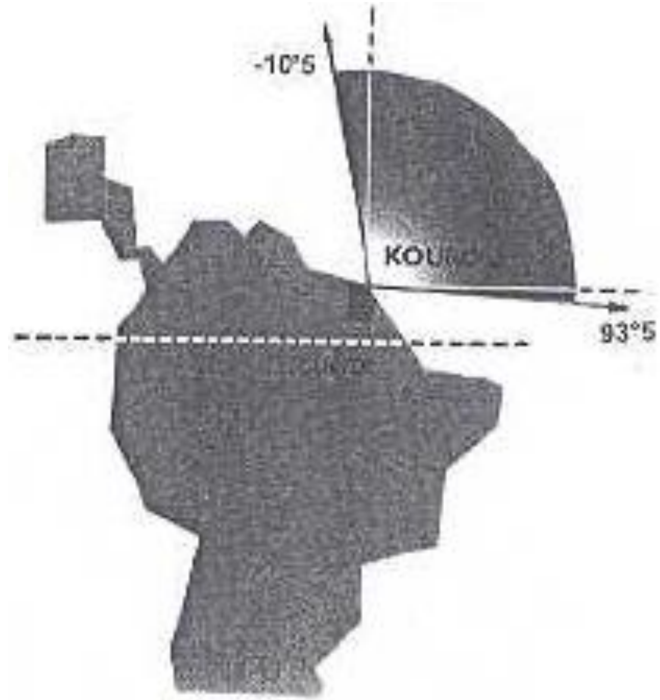
- Selection of a launch system depends on the mission′s technical characteristics
- Others factors are risk, cost, political or security considerations
- First criterion for selection is the capability of launcher to handle payload requirements in terms of weight and dimensions and orbit parameters (altitude, inclination)
- Other factors are launcher availability, infrastructure availability (launch pad, communication systems), launch rate
- Customer decides on a dedicated or shared launch

Selection of Launch Systems (II)

- Optimisation of the spacecraft system has to be worked out
- Payload mass can be augmented by using the launcher or by using its own propulsion system
- GEO comm's satellites have to use their own propulsion system as a 4th stage in order to reach a geostationary transfer orbit
- Location of a launch site is important: if close to the equator one can use to his advantage due to the high velocity derived from the earth's rotation
- Main advantage launching from equator is for geostationary orbits for which no orbit change is necessary

Selection of Launch Systems (III)

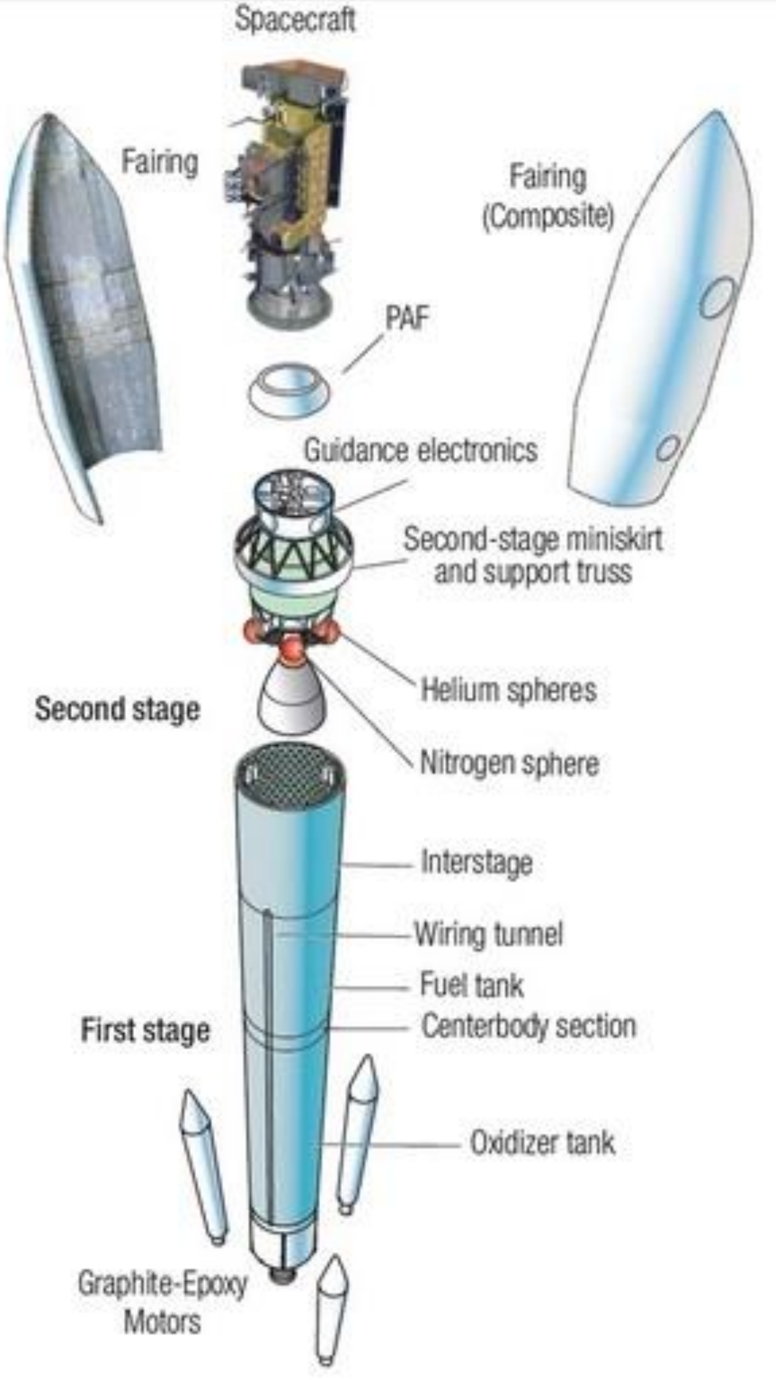


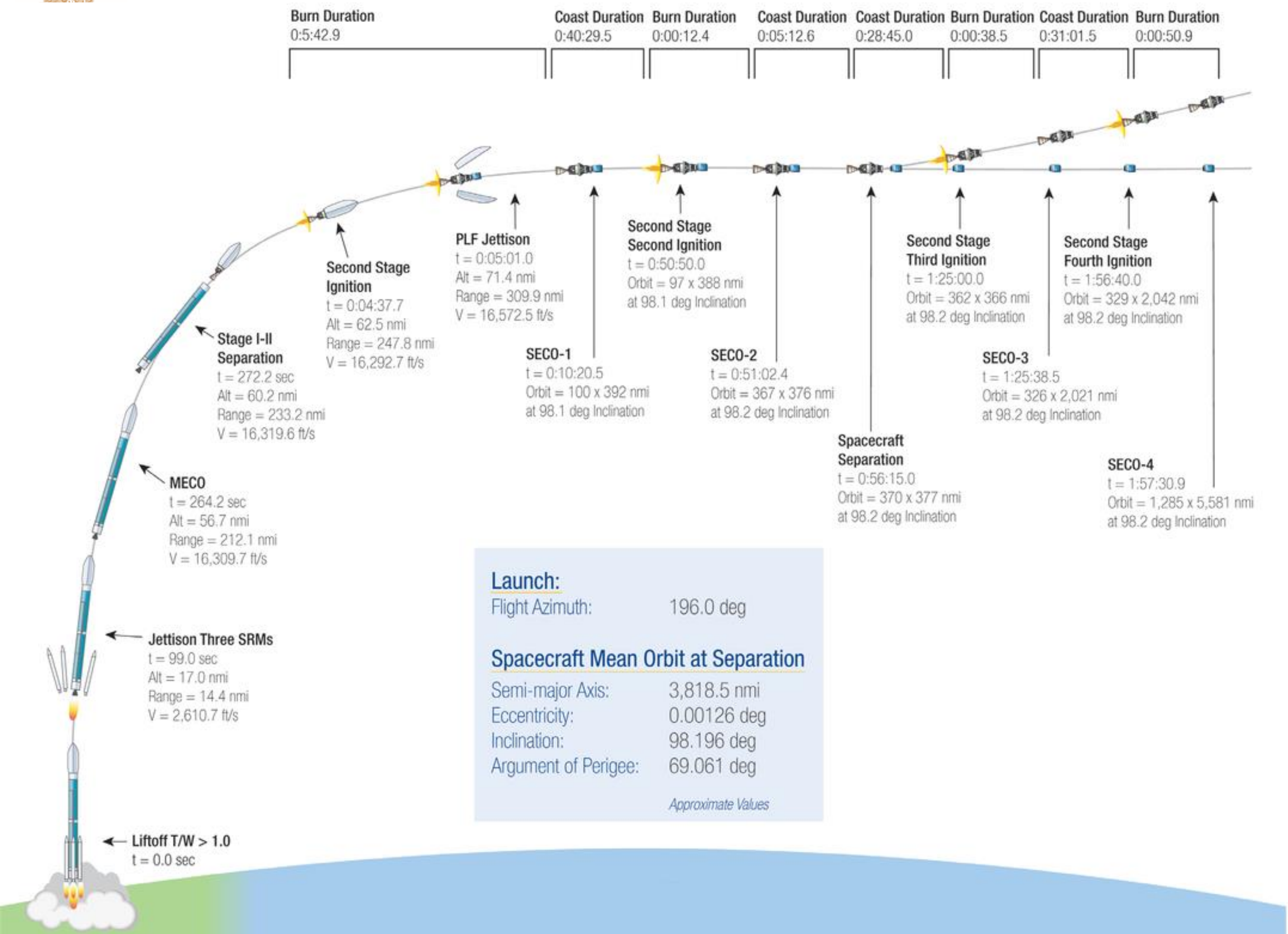


- A plane change requires a lot of energy: in LEO a ΔV=200 m/s is needed for an inclination change of 1 deg
- From an equatorial launch site any orbit inclination can be chosen
- For other locations the original orbit inclination can’t be lower than the latitude of the launch site

Overview of Launchers

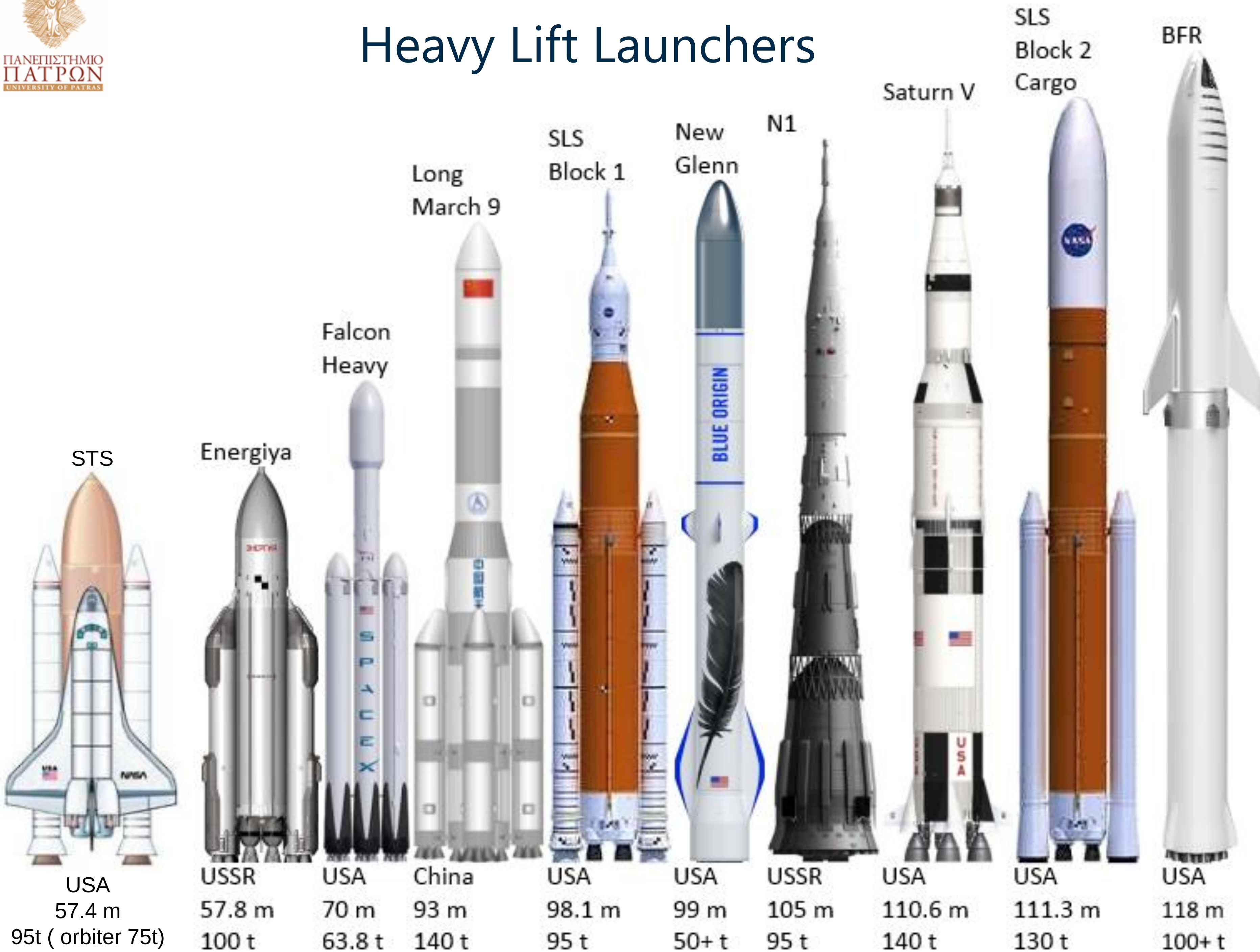
Launch Vehicle: Delta II 7320



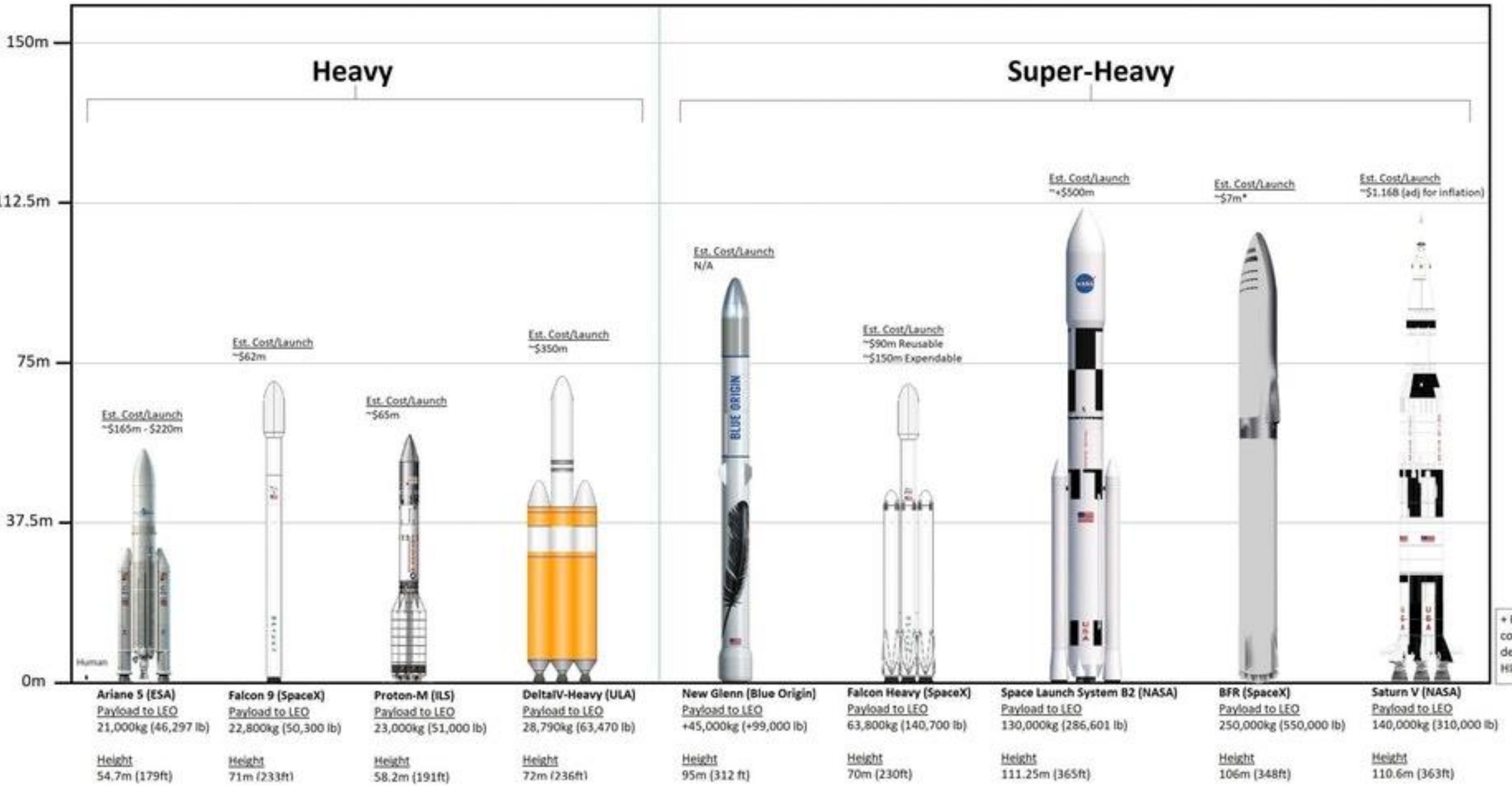


Burn Duration
0:5:42.9
Coast Duration
0:40:29.5
Burn Duration
0:00:12.4
Coast Duration
0:05:12.6
Coast Duration
0:28:45.0
Burn Duration
0:00:38.5
Coast Duration
0:31:01.5
Burn Duration
0:00:50.9
PLF Jettison
t = 0:05:01.0
Alt = 71.4 nmi
Range = 309.9 nmi
V = 16,572.5 ft/s
Second Stage
Second Ignition
t = 0:50:50.0
Orbit = 97 x 388 nmi
at 98.1 deg Inclination
Second Stage
Third Ignition
t = 1:25:00.0
Orbit = 362 x 366 nmi
at 98.2 deg Inclination
Second Stage
Fourth Ignition
t = 1:56:40.0
Orbit = 329 x 2,042 nmi
at 98.2 deg Inclination
Second Stage
Ignition
t = 0:04:37.7
Alt = 62.5 nmi
Range = 247.8 nmi
V = 16,292.7 ft/s
Stage I-II
Separation
t = 272.2 sec
Alt = 60.2 nmi
Range = 233.2 nmi
V = 16,319.6 ft/s
SECO-1
t = 0:10:20.5
Orbit = 100 x 392 nmi
at 98.1 deg Inclination
SECO-2
t = 0:51:02.4
Orbit = 367 x 376 nmi
at 98.2 deg Inclination
SECO-3
t = 1:25:38.5
Orbit = 326 x 2,021 nmi
at 98.2 deg Inclination
Spacecraft
Separation
t = 0:56:15.0
Orbit = 370 x 377 nmi
at 98.2 deg Inclination
SECO-4
t = 1:57:30.9
Orbit = 1,285 x 5,581 nmi
at 98.2 deg Inclination
MECO
t = 264.2 sec
Alt = 56.7 nmi
Range = 212.1 nmi
V = 16,309.7 ft/s
Jettison Three SRMs
t = 99.0 sec
Alt = 17.0 nmi
Range = 14.4 nmi
V = 2,610.7 ft/s
Liftoff T/W > 1.0
t = 0.0 sec
Launch:
Flight Azimuth: 196.0 deg
Spacecraft Mean Orbit at Separation
Semi-major Axis: 3,818.5 nmi
Eccentricity: 0.00126 deg
Inclination: 98.196 deg
Argument of Perigee: 69.061 deg
Approximate Values

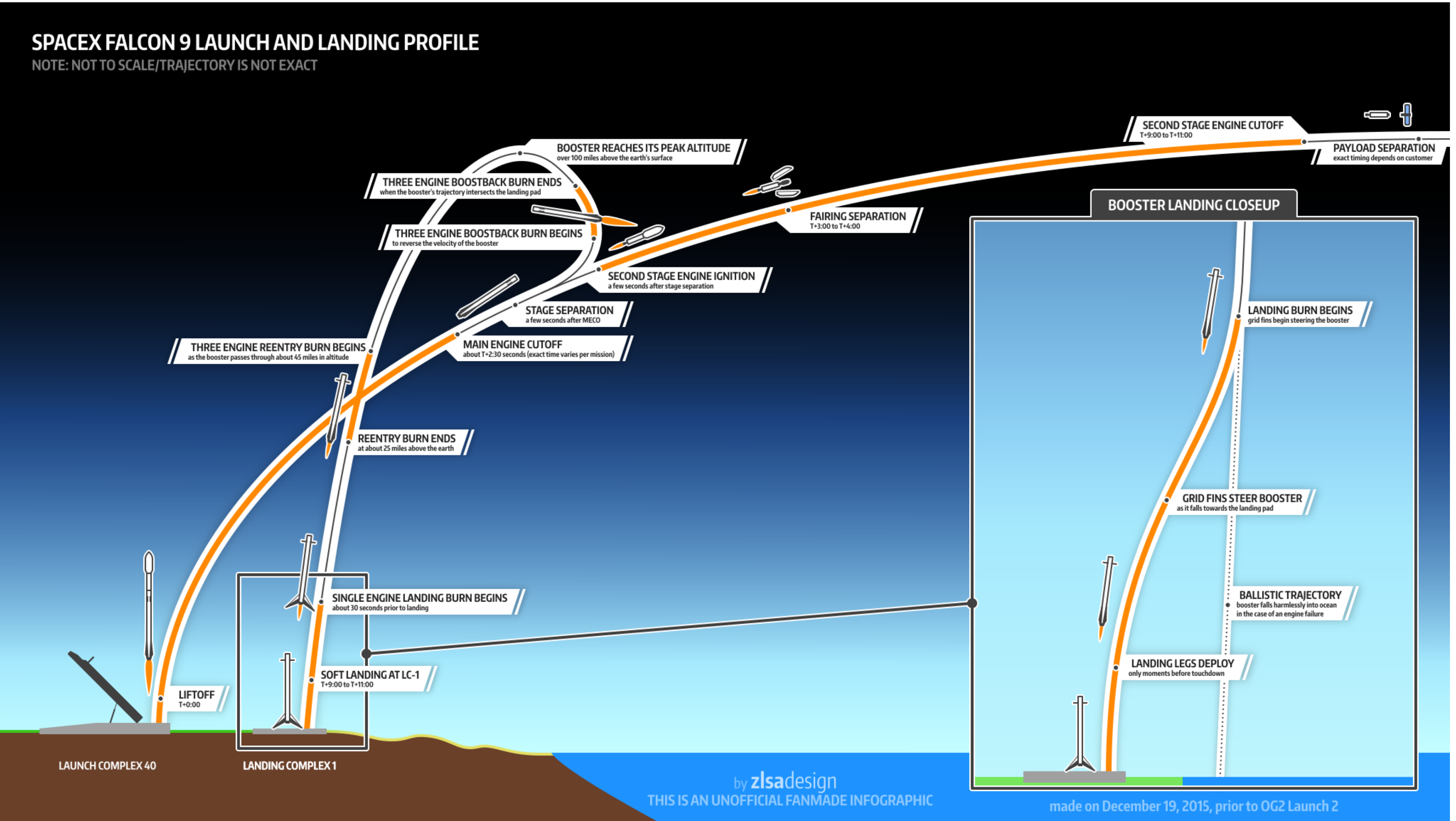
Heavy Lift Launchers



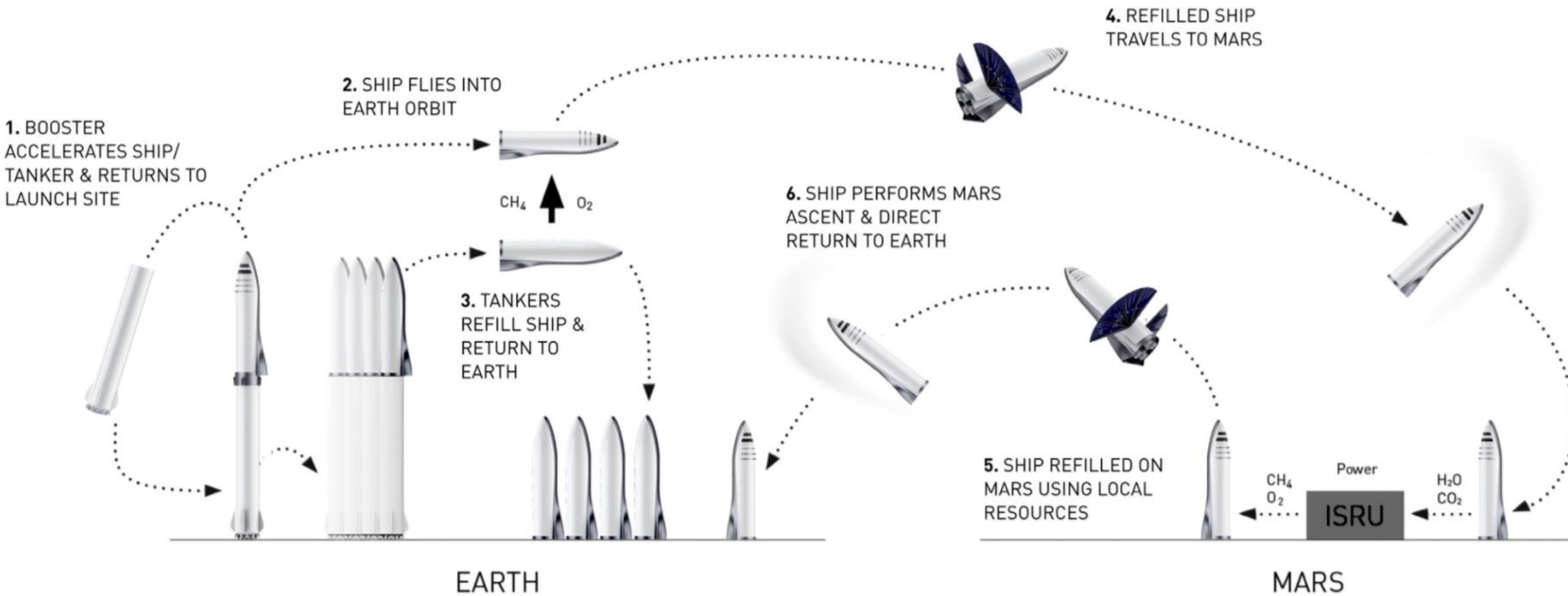
Heavy/Super Heavy Launch Costs (Estimates)

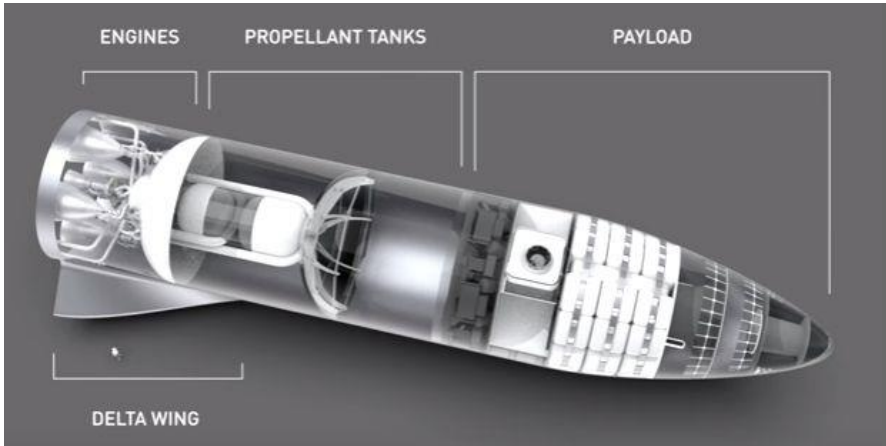


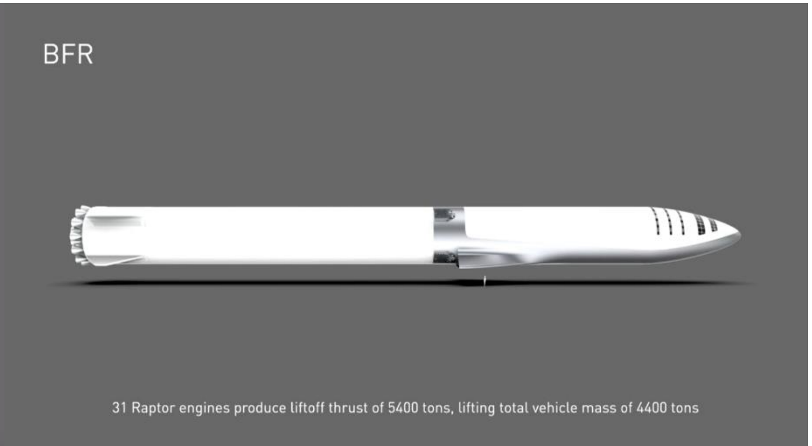
SpaceX Falcon 9 Flight Profile



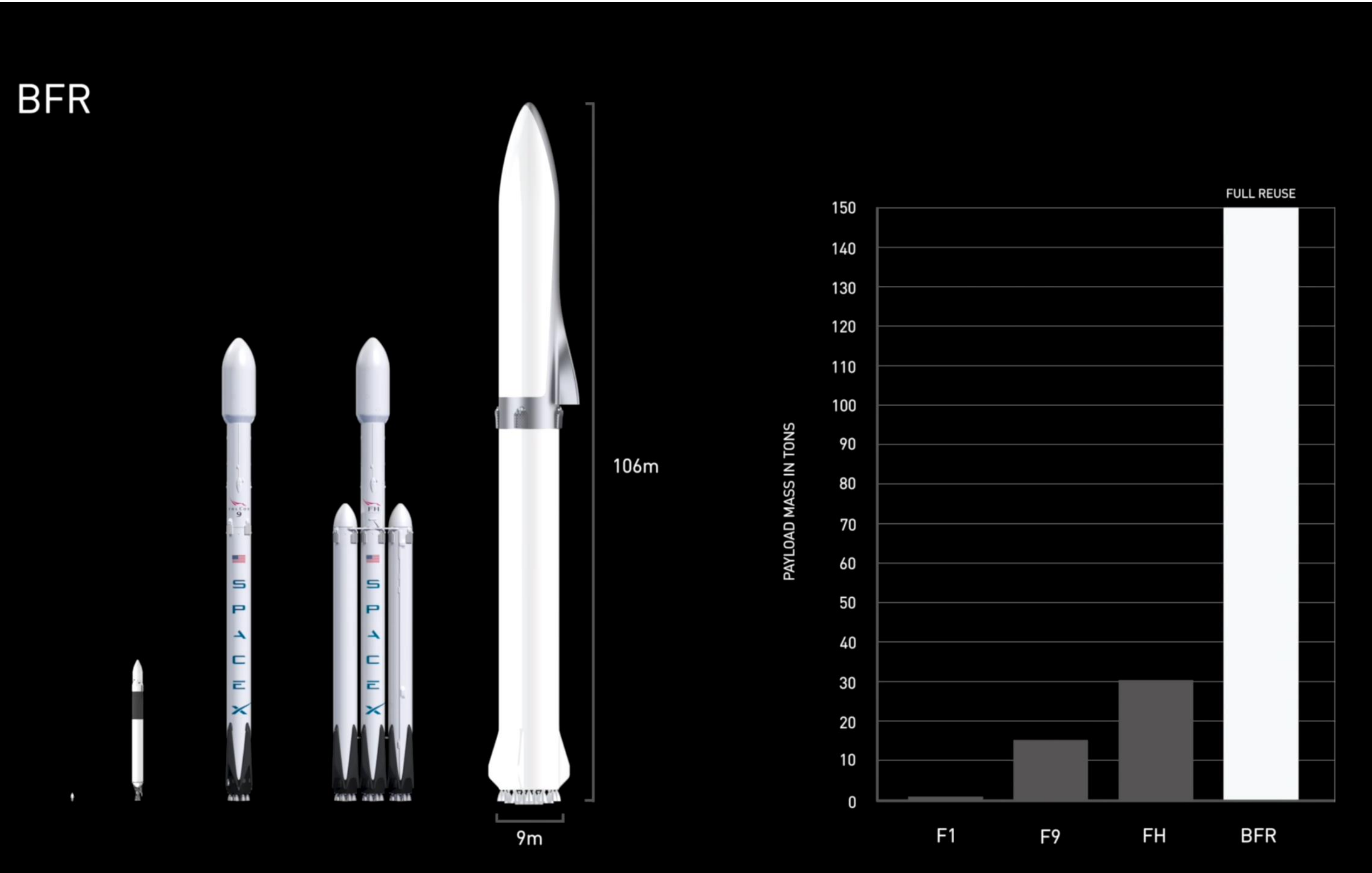
Big Falcon Rocket (BFR) - Mars



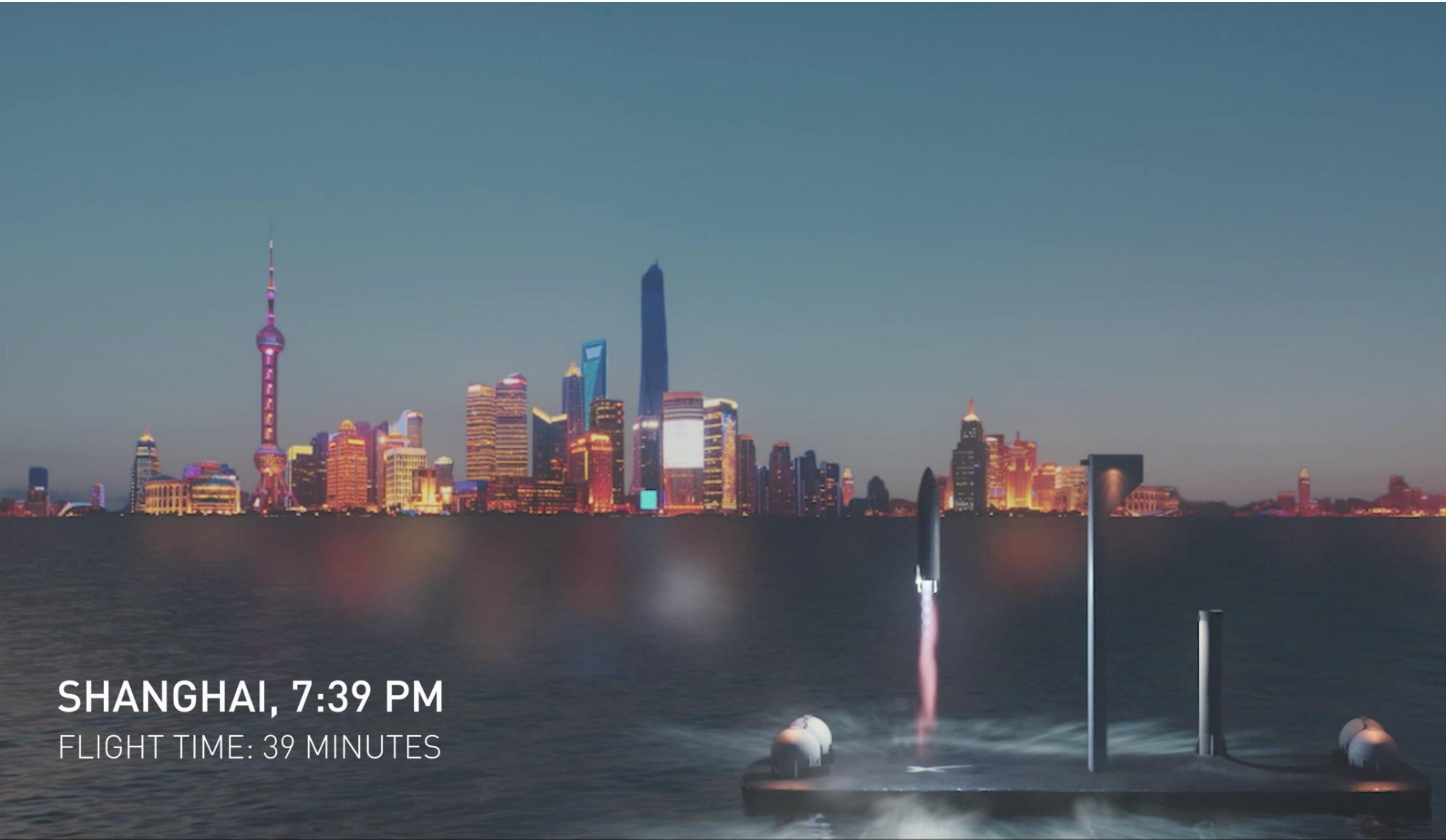




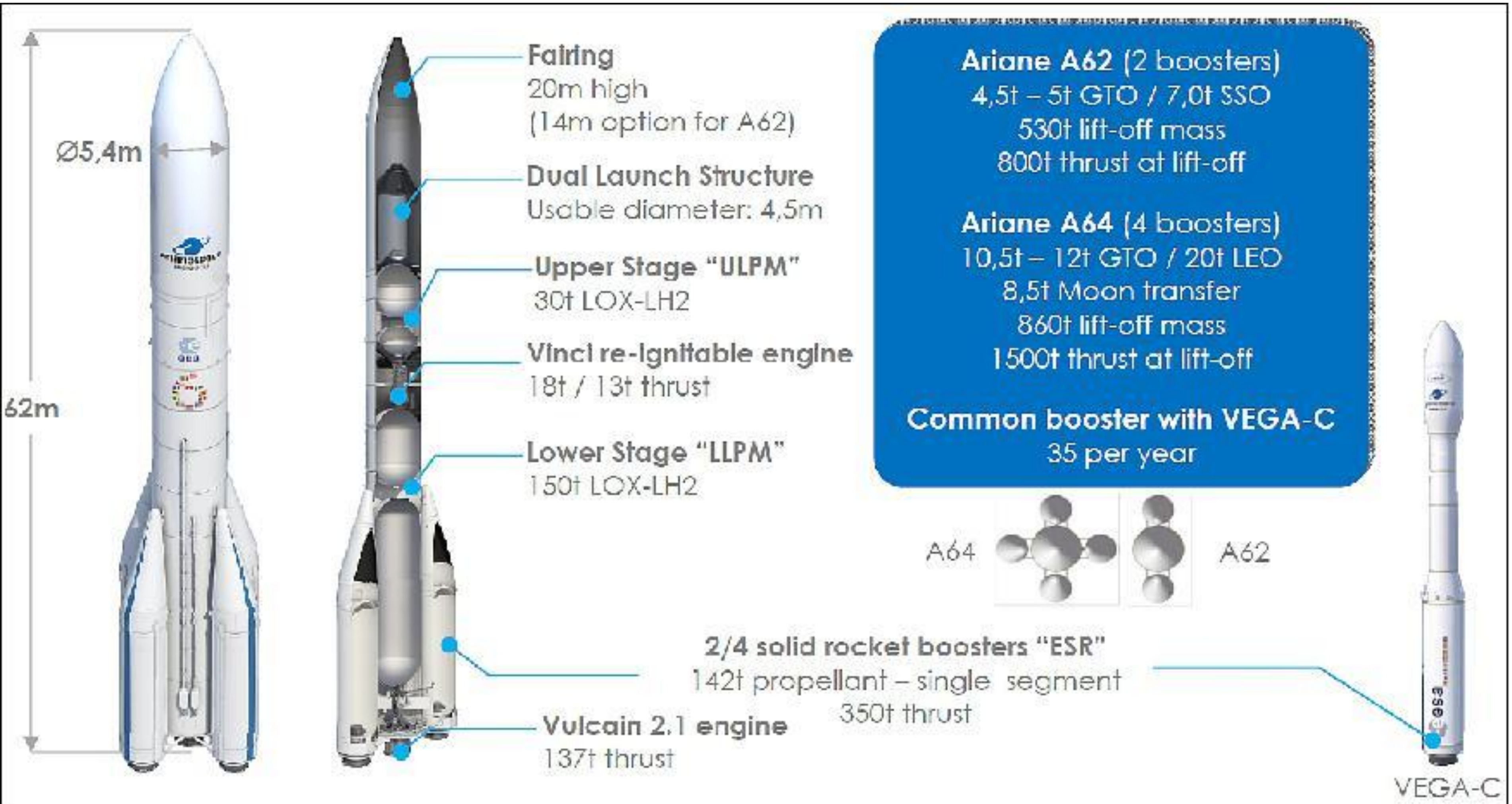
Space X BFR Reusable Heavy Launcher



BFR: New York to Shanghai in 40 Minutes!

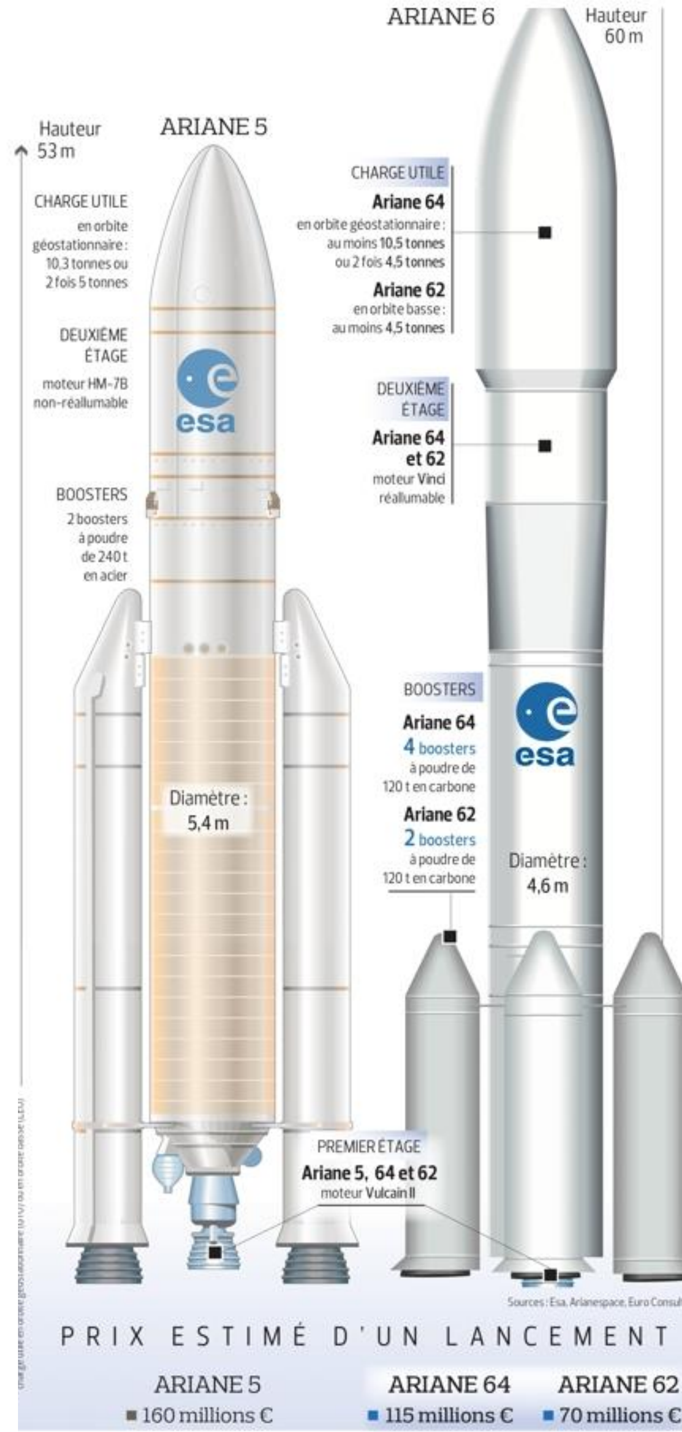


Ariane 6



Ø5,4m
62m
Fairing
20m high
(14m option for A62)
Dual Launch Structure
Usable diameter: 4,5m
Upper Stage "ULPM"
30t LOX-LH2
Vinci re-ignitable engine
18t / 13t thrust
Lower Stage "LLPM"
150t LOX-LH2
Ariane A62 (2 boosters)
4,5t – 5t GTO / 7,0t SSO
530t lift-off mass
800t thrust at lift-off
Ariane A64 (4 boosters)
10,5t – 12t GTO / 20t LEO
8,5t Moon transfer
860t lift-off mass
1500t thrust at lift-off
Common booster with VEGA-C
35 per year
A64
A62
2/4 solid rocket boosters "ESR"
142t propellant – single segment
350t thrust
Vulcain 2.1 engine
137t thrust
VEGA-C

Ariane 6



ARIANE 6
Hauteur 60 m
Hauteur 53 m
ARIANE 5
CHARGE UTILE
en orbite géostationnaire : 10,3 tonnes ou 2 fois 5 tonnes
DEUXIÈME ÉTAGE
moteur HM-7B non-réallumable
BOOSTERS
2 boosters à poudre de 240 t en acier
Diamètre : 5,4 m
CHARGE UTILE
Ariane 64
en orbite géostationnaire : au moins 10,5 tonnes ou 2 fois 4,5 tonnes
Ariane 62
en orbite basse : au moins 4,5 tonnes
DEUXIÈME ÉTAGE
Ariane 64 et 62
moteur Vinci réallumable
BOOSTERS
Ariane 64
4 boosters à poudre de 120 t en carbone
Ariane 62
2 boosters à poudre de 120 t en carbone
Diamètre : 4,6 m
PREMIER ÉTAGE
Ariane 5, 64 et 62
moteur Vulcain II
Sources : Esa, Arianespace, Euro Consult
PRIX ESTIMÉ D'UN LANCEMENT
ARIANE 5
160 millions €
ARIANE 64
115 millions €
ARIANE 62
70 millions €

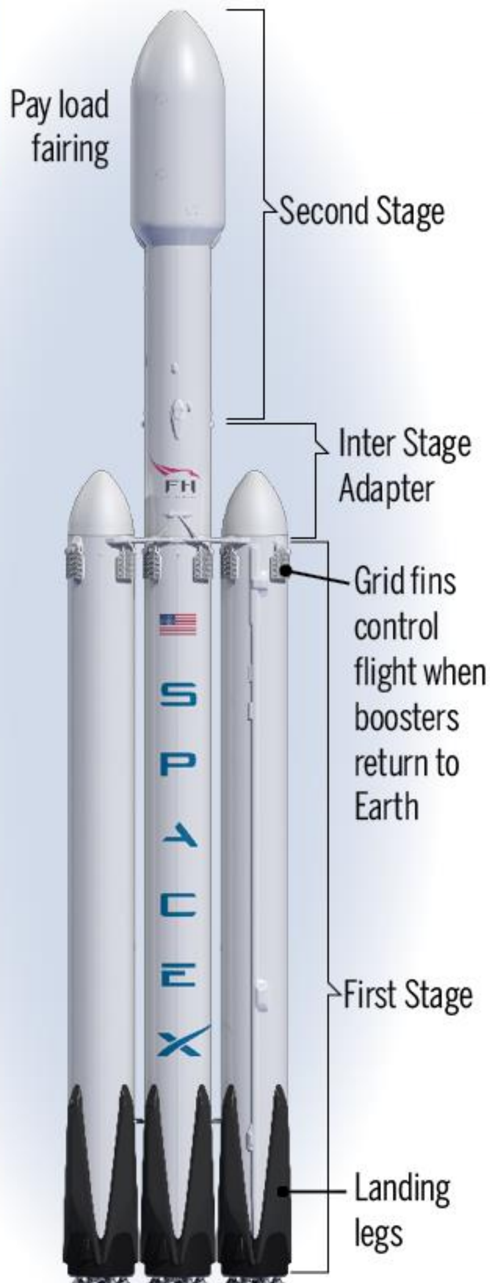
FALCON HEAVY

The world's most powerful rocket

Height: 70 meters (229.6 feet)
Stages: Two
Boosters: Two
Re-usable Cores: Three
Engines: 27
Total width: 12.2m
Mass: 1,420,788kg
Total thrust at lift-off: 22,819 kilonewtons

PAYLOAD

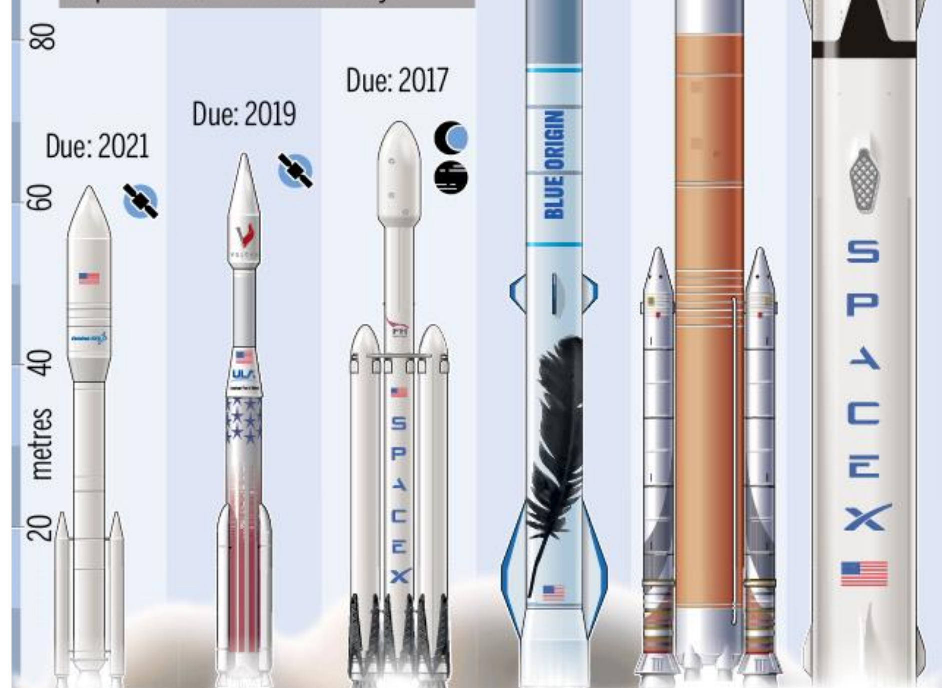
- To Low Earth Orbit: 63,800kg
- To Mars: 16,800kg
- Will allow SpaceX to enter the new satellite launch markets



First payload SpaceX founder Elon Musk's Tesla Roadster bound for Mars orbit

US ROCKETS IN THE RACE TO EXPLORE SPACE

A SpaceX Falcon 9 rocket takes off from Cape Canaveral in Florida on May 1



Sources: Aviation Week, Space.com, NASA, wire agencies; © GRAPHIC NEWS

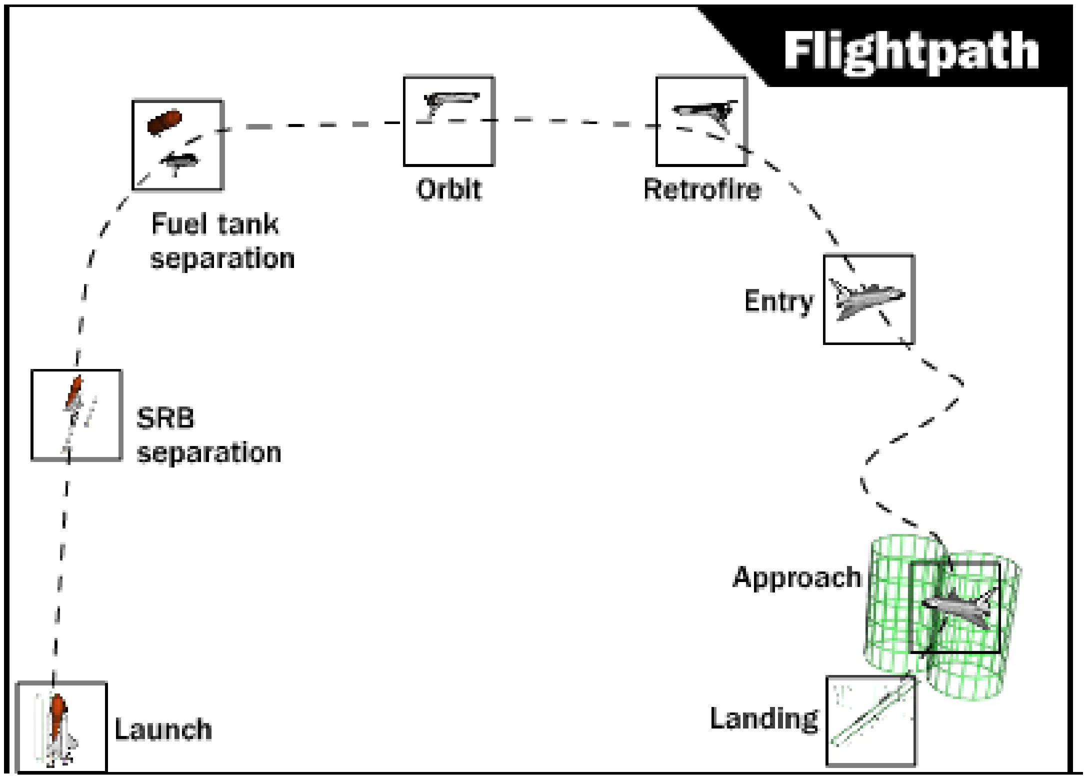
Blue Origin ‘New Glenn’

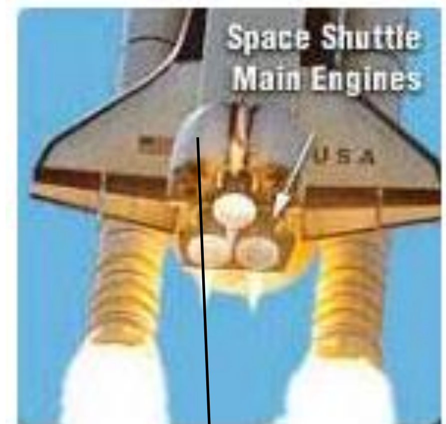
SSTS Development

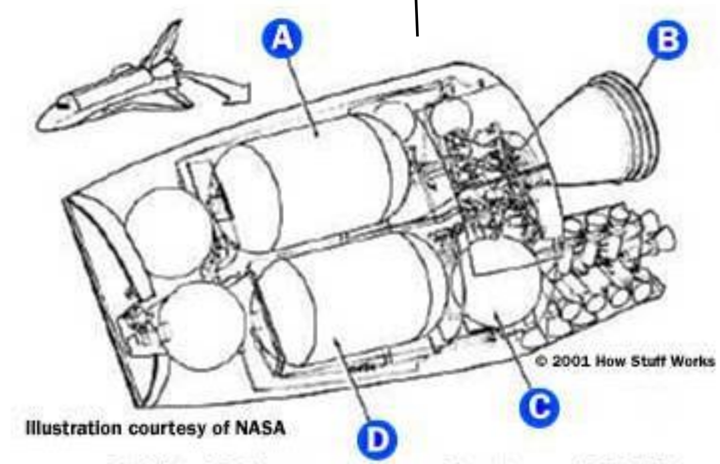
- The Shuttle program was launched on January 5, 1972, when President Richard M. Nixon announced that NASA would proceed with the development of a reusable, low-cost Space Shuttle system.
- The project was already to take longer than originally anticipated due to the year-to-year funding caps. Nevertheless, work started quickly and several test articles were available within a few years.
- Most notable among these was the first complete Orbiter, originally to be known as *Constitution*. However, a massive write-in campaign from fans of the *Star Trek* television series convinced the White House to change the name to *Enterprise*. Amid great fanfare, the *Enterprise* was rolled out on September 17, 1976, and later conducted a successful series of glide-approach and landing tests that were the first real validation of the design

Space Shuttle Transportation System

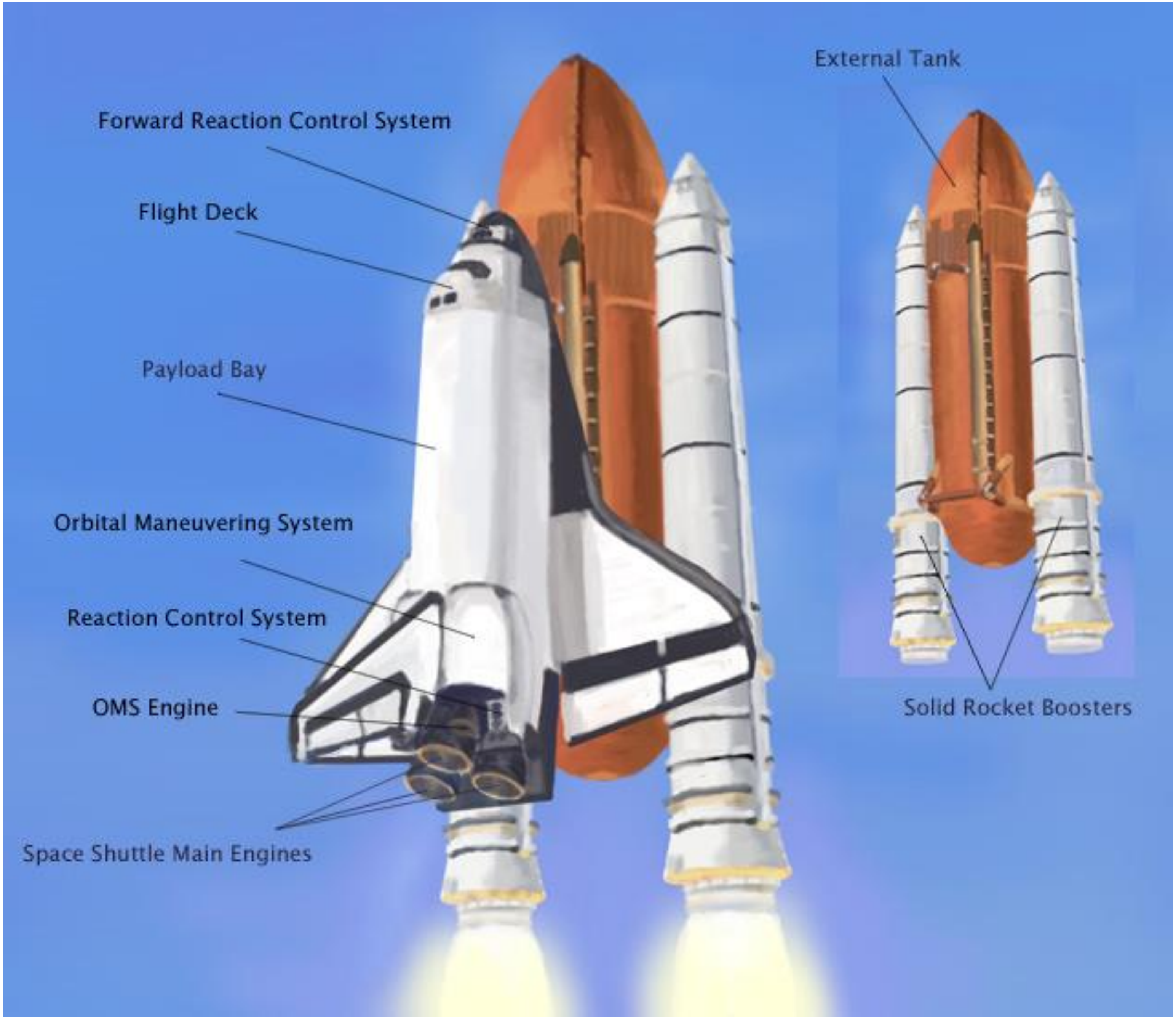
Space Shuttle







http://science.howstuffworks.com/space-shuttle1.htm


External Tank
Forward Reaction Control System
Flight Deck
Payload Bay
Orbital Maneuvering System
Reaction Control System
OMS Engine
Solid Rocket Boosters
Space Shuttle Main Engines


CCCP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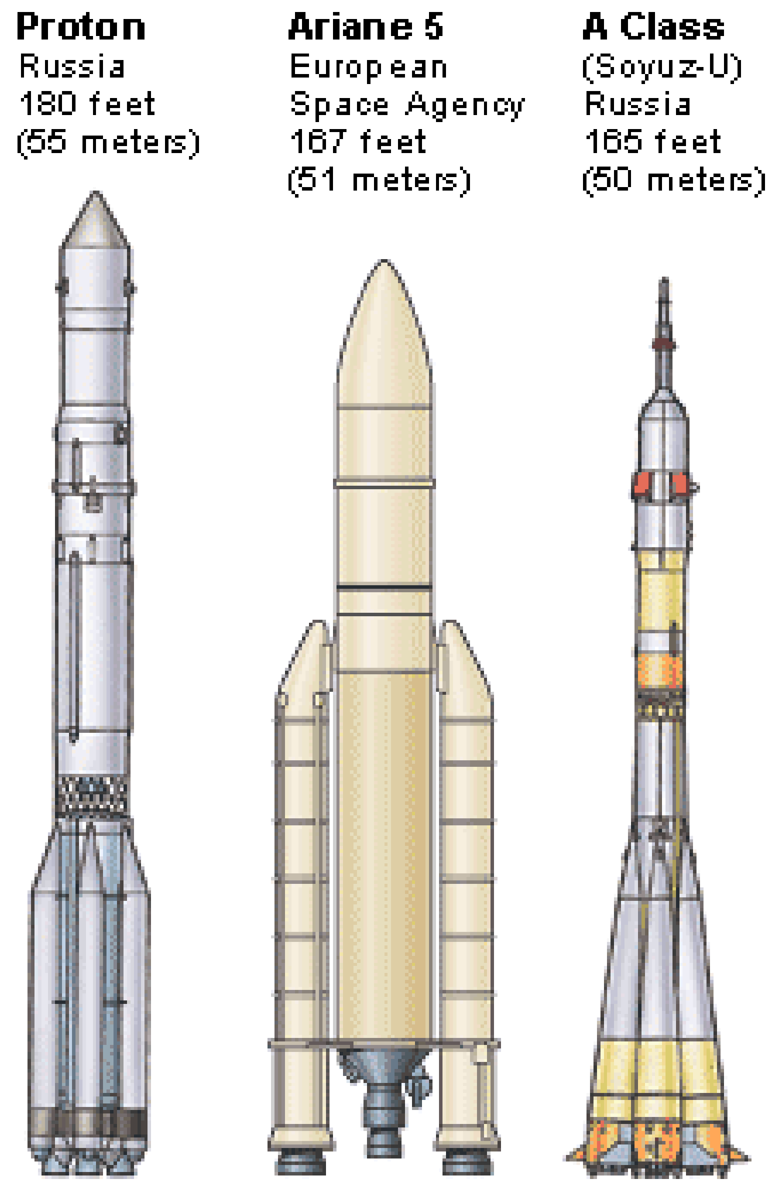
Soviet Space Shuttle: Buran

- Buran was not an integral part of the system, but rather a payload for the Energia launcher.
- Therefore payloads — other than Buran orbiter — with mass as high as 80 metric tons could be lifted to space by Energia, as was the case on its first launch.
- As Buran was designed to be capable of both manned and unmanned flight, it had automated landing capability; the manned version was never operational.
- The orbiter had no main rocket engines, freeing space and weight for additional payload; the largest cylindrical structure is the Energia carrier-rocket, not just a fuel tank.
- The boosters used liquid propellant (kerosene/oxygen).
- Solid Rocket Boosters but requires a new External Fuel Tank for each flight, as the tank is not recovered and is allowed to burn up in the atmosphere.

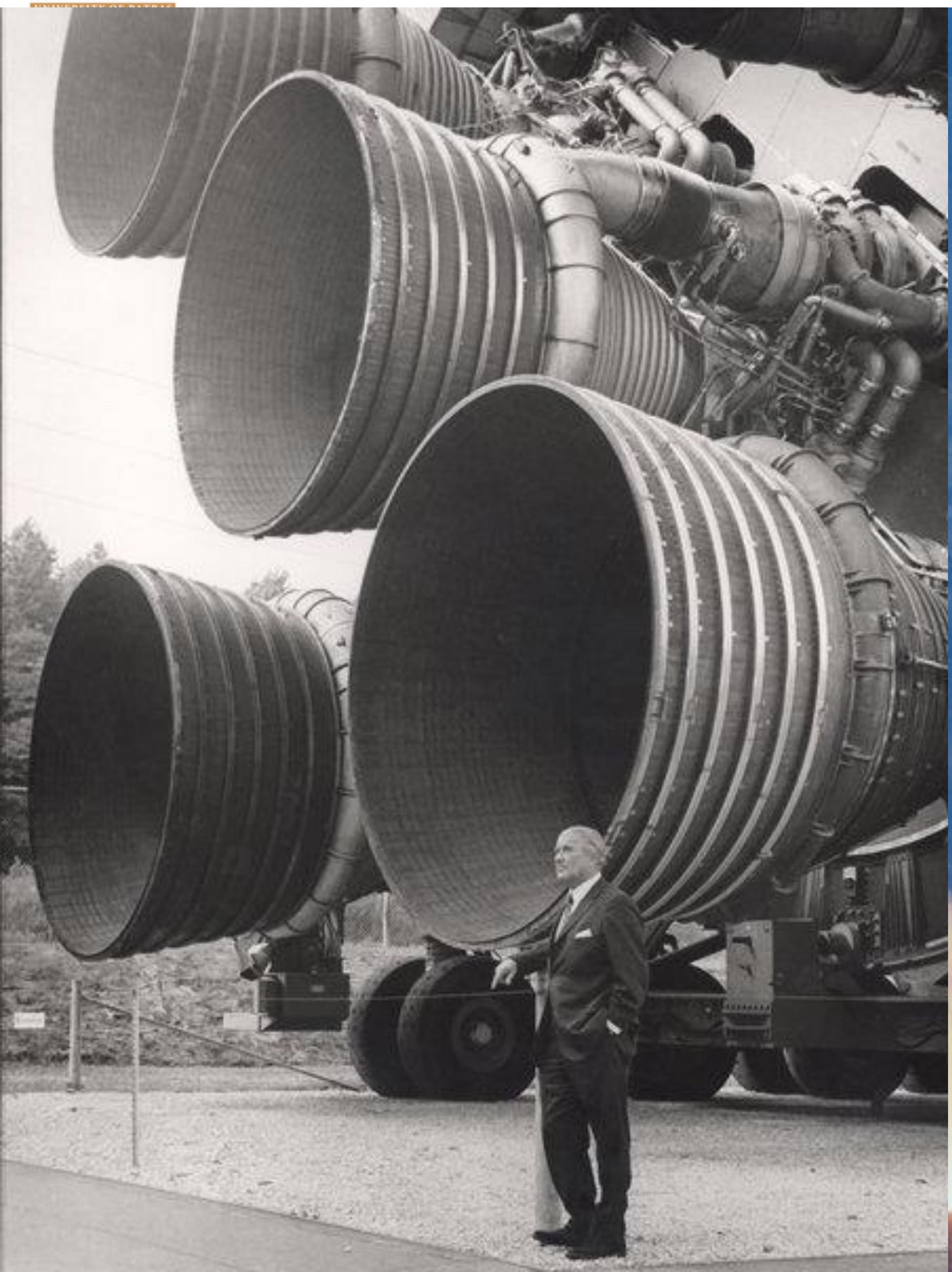
Ariane 5

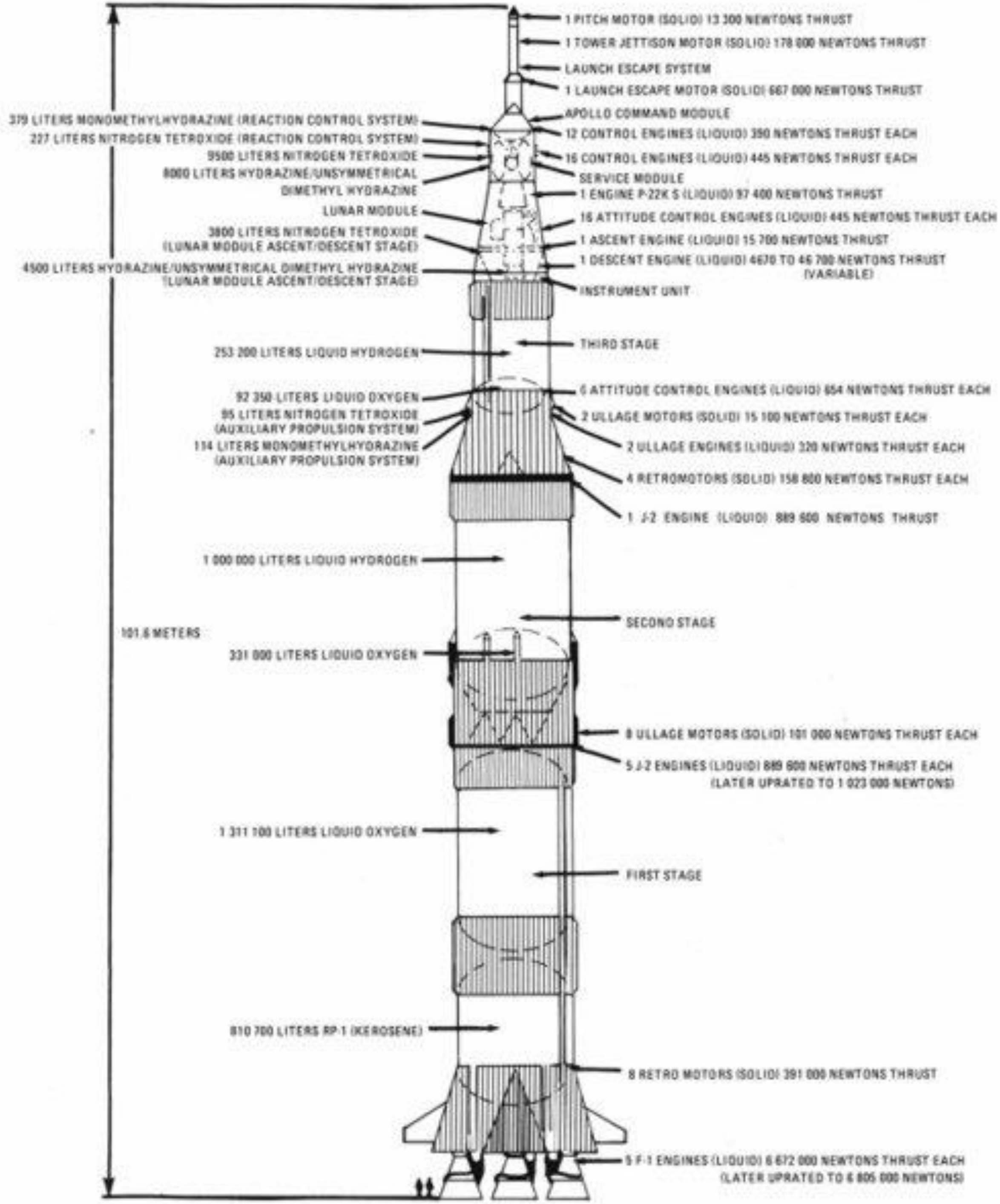

Proton
Russia
180 feet
(55 meters)
Ariane 5
European
Space Agency
167 feet
(51 meters)
A Class
(Soyuz-U)
Russia
165 feet
(50 meters)

Soyuz

Other Launch Vehicles

Saturn 5




1 PITCH MOTOR (SOLID) 13 300 NEWTONS THRUST
1 TOWER JETTISON MOTOR (SOLID) 178 000 NEWTONS THRUST
LAUNCH ESCAPE SYSTEM
1 LAUNCH ESCAPE MOTOR (SOLID) 667 000 NEWTONS THRUST
APOLLO COMMAND MODULE
12 CONTROL ENGINES (LIQUID) 390 NEWTONS THRUST EACH
16 CONTROL ENGINES (LIQUID) 445 NEWTONS THRUST EACH
SERVICE MODULE
1 ENGINE P-22K S (LIQUID) 97 400 NEWTONS THRUST
16 ATTITUDE CONTROL ENGINES (LIQUID) 445 NEWTONS THRUST EACH
1 ASCENT ENGINE (LIQUID) 15 700 NEWTONS THRUST
1 DESCENT ENGINE (LIQUID) 4670 TO 46 700 NEWTONS THRUST (VARIABLE)
INSTRUMENT UNIT
379 LITERS MONOMETHYLHYDRAZINE (REACTION CONTROL SYSTEM)
227 LITERS NITROGEN TETROXIDE (REACTION CONTROL SYSTEM)
9500 LITERS NITROGEN TETROXIDE
8000 LITERS HYDRAZINE/UNSYMMETRICAL DIMETHYL HYDRAZINE
LUNAR MODULE
3800 LITERS NITROGEN TETROXIDE (LUNAR MODULE ASCENT/DESCENT STAGE)
4500 LITERS HYDRAZINE/UNSYMMETRICAL DIMETHYL HYDRAZINE (LUNAR MODULE ASCENT/DESCENT STAGE)
253 200 LITERS LIQUID HYDROGEN
THIRD STAGE
92 350 LITERS LIQUID OXYGEN
95 LITERS NITROGEN TETROXIDE (AUXILIARY PROPULSION SYSTEM)
114 LITERS MONOMETHYLHYDRAZINE (AUXILIARY PROPULSION SYSTEM)
6 ATTITUDE CONTROL ENGINES (LIQUID) 654 NEWTONS THRUST EACH
2 ULLAGE MOTORS (SOLID) 15 100 NEWTONS THRUST EACH
2 ULLAGE ENGINES (LIQUID) 320 NEWTONS THRUST EACH
4 RETROMOTORS (SOLID) 158 800 NEWTONS THRUST EACH
1 J-2 ENGINE (LIQUID) 889 600 NEWTONS THRUST
1 000 000 LITERS LIQUID HYDROGEN
SECOND STAGE
101.6 METERS
331 000 LITERS LIQUID OXYGEN
8 ULLAGE MOTORS (SOLID) 101 000 NEWTONS THRUST EACH
5 J-2 ENGINES (LIQUID) 889 600 NEWTONS THRUST EACH (LATER UPRATED TO 1 023 000 NEWTONS)
1 311 100 LITERS LIQUID OXYGEN
FIRST STAGE
810 700 LITERS RP-1 (KEROSENE)
8 RETRO MOTORS (SOLID) 391 000 NEWTONS THRUST
5 F-1 ENGINES (LIQUID) 6 672 000 NEWTONS THRUST EACH (LATER UPRATED TO 6 805 000 NEWTONS)


USA

Space X Falcon Heavy – Reusable Rockets

Falcon 9: Reusable Rockets

Soyuz Launcher – Failure (2018)

Space Shuttle Columbia Accident

Soyuz Docking with the ISS

Problem Set 1

1. An advanced inertial upper-stage (IUS) is being designed to boost a new cable TV satellite from low-altitude parking orbit to geosynchronous orbit. The ΔV for the first burn of the Hohmann transfer is 3.34 km/s, and the effective exhaust velocity of the IUS is 3000 m/s. If the mass of just the structure of the IUS, without propellant, is 100 kg and the satellite mass is 1000 kg, what mass of propellant should be loaded into the upper-stage?
2. Two rockets are candidates for a space mission. Rocket 1 has an I_{sp} of 300 seconds, and rocket 2 has an I_{sp} of 350 seconds. If the total ΔV needed for the mission is 1000 m/s, how much more propellant will rocket 1 need over the life of the mission? Assume the dry mass of the spacecraft is 1000 kg.

Problem Set 1 (II)

3. An experimental two-stage booster is preparing to launch from the Kennedy Space Centre. The booster must deliver a total ΔV (ΔV_{design}) of 10,000 m/s. The total mass of the 2^{nd} stage, including structure and propellant is 12,000 kg, 9000 kg of which is propellant. The payload mass is 2000 kg. The I_{sp} of the 1^{st} stage is 350 seconds and the 2^{nd} stage is 8000 kg. What mass of propellant must be loaded on the 1^{st} stage to achieve the required ΔV_{design}? What is the vehicle′s total mass at lift-off?