Launch vehicle design and sizing

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Revision record

Revision	Date	Changes
1.00	Feb. 2010	Initial release.
1.01	July 2010	Worked in some errata; Total page count is 96 pages.
1.02	Oct. 2011	Added solutions to the problems included in the syllabus; Added two examples demonstrating the calculation of mass fractions of staged vehicles; Some minor corrections; Total page count is 102 pages
1.03	Feb. 2013	Revision record added; Changed course code; Added calculation examples on structural design of launcher stage tanks, stage structure and an integrated design of tanks and stage; Annex added on Nose cone volumes; Some minor corrections. Total page count is 107 pages.
1.04	Sep. 2013	Changed symbols used to denote mass of various launcher elements; Omitted [SSE] as study material. Some materials are now included in this document; Contents list extended; Updated slides and converted various slides to text; Added text on fairing sizing; Total page count is 116 pages.
1.05	Aug. 2015	Updated front page; Content is more detailed; Order has been changed to reflect better the design process followed; List of tables and list of figures added Equation numbering added; Section on launch vehicle concepts extended; Section on trade studies extended; Updated reliability data in Appendix B; Total page count is 129 pages.
1.06	Aug. 2017	Various small updates to main text Appendix E has been updated Appendix F on size data of rocket stages has been added. Total page count is 138 pages.
1.06.1	Sep. 2017	Appendix E: Some relations in subsection B have been updated as well as the example results in subsection E (mainly results presented under model C and D). Total page count remains unchanged.
1.06.1	Nov. 2019	Some mistakes have been corrected for. Total page count remains unchanged.
1.07	Sep. 2022	Minor changes to main text and appendices Appendix E: Relations in section B have been updated allowing for an improved estimation of LV mass.

Vehicles on cover page include European Ariane 5 and 6 rocket launch vehicle (courtesy ESA), USA Apollo command module (CM) and service module (SM) and Mars Sample Return ascent module lifting off from Mars (courtesy NASA).

Preface

In this lecture series, which forms part of the course Aerospace Design and Systems Engineering Elements I (AE1110-II), some basics of launch vehicle design and sizing are dealt with. Launch vehicles have been discussed earlier in the curriculum. Main focus in this earlier course is on the history of launch vehicles and the main development trends. Also a typical flight profile of a launch vehicle has been discussed and characteristic parameters, including vehicle mass at lift off, payload mass to orbit, mass ratio, etc., have been defined and typical values discussed. Finally also the various vehicle elements (vehicle stages, payload fairing and attach fitting) making up a launch vehicle have been introduced and explained.

In the current lecture series, we will spend 4 lecture hours of 45 minutes each to explain launch vehicle design issues. As (space) launch vehicles to a large extent have the same subsystems as spacecraft, we tend to focus on issues specific for launchers. For more general issues relating to the design of launch vehicle subsystems, you are referred to the material on spacecraft and/or aircraft design earlier dealt with in this course.

Learning goal
The student shall be able to conduct all steps necessary to perform a conceptual design (for a definition, see earlier in this course) of a rocket launch vehicle and its (sub-) systems

Learning objectives
The student shall be able to
- Describe/explain what a launch vehicle is and what it does (ae1110-II)
- Describe the launch vehicle design process
- List important issues for launch vehicle design
- Apply a simple method to perform a first design of a launch vehicle
    - Discuss/explain the main vehicle subsystems and their functions as well as the metrics for defining a good design
    - Calculate mass ratios (stage and (sub)rocket)
    - Generate a vehicle concept
    - Perform propulsion system sizing
    - Perform sizing for aerodynamics
    - Ensure static stability
    - Perform structural sizing
    - Perform sizing of other subsystems
    - Calculate the effect of staging on vehicle mass & payload (optimum staging)

Prerequisites
Student should have mastered the learning goals of the course ae1110-II and the learning goals related to spacecraft and aircraft design as taught earlier in the course ae1222-II. Students should also be able to determine center of pressure, center of mass and area/mass moments of inertia of simple bodies and to combine these estimates to represent a body of a more complex shape.

Study material
- This reader (including annexes)
- Mobius problems available via Brightspace
- Tutorial (for exercising)

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1 Introduction

This work is about the conceptual/preliminary design[1] of (space) launch vehicles (SLVs). An SLV is a system/vehicle that is designed to launch a payload safely, intact, and accurately from the surface of some planet or other celestial body to some target orbit about this planet. SLVs destined to fly at high altitude most commonly are propelled by one or more rocket motors/engines[2] and hence are referred to as rockets[3]. They can be distinguished depending on the mission. We consider ballistic missiles, sub-orbital rockets or sounding rockets[4] and orbital launch vehicles (for spaceflight). Orbital launch vehicles or more specifically space rockets are the focus of this work, but some of the material presented can also be applied to other launch vehicles.

An overview of current and past, rocket propelled, SLVs used to bring a useful load (payload) from Earth into space is given in Figure 1.

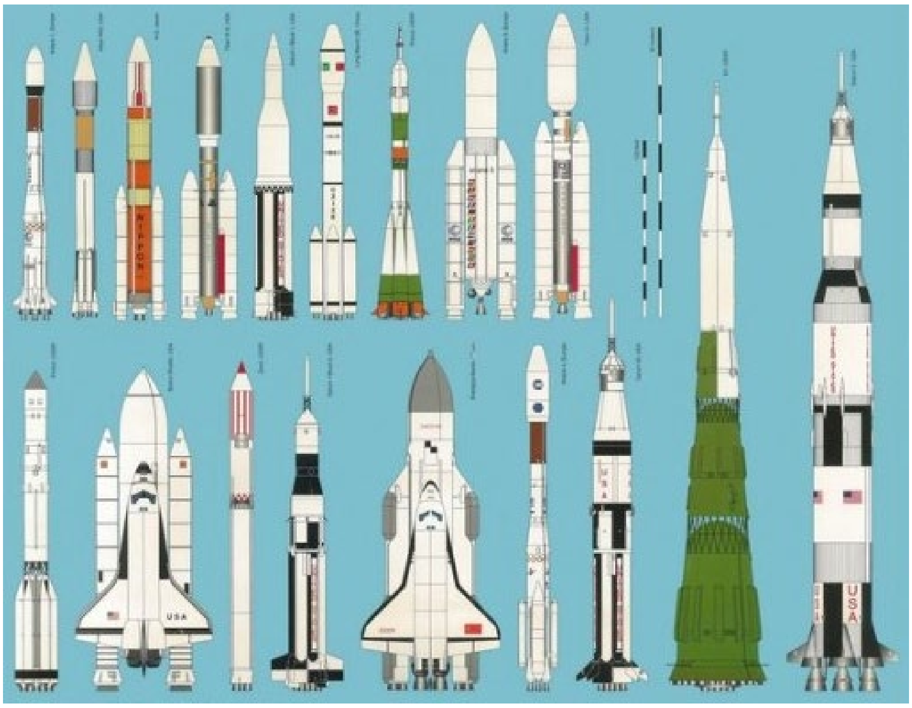


Figure 1: Overview of historical rocket launch vehicles (on relative scale; courtesy NASA)

The rockets shown in figure include such SLV as the European Ariane 1, Ariane 4 and Ariane 5, the American Titan Heavy, Space Shuttle and Saturn V (Vehicle on far right), the Russian Soyuz, Buran and N rocket (vehicle next to Saturn V), the Chinese Long March rocket and the Japanese H1 and H2 rocket. The SLVs shown are all expendable, multi-staged, rocket-propelled SLVs, with the term <u>expendable</u> meaning that these rockets are designed to be used only once (i.e. they are "expended" during a single flight and then discarded), and their components are not recovered after launch. This of course leads to a significant waste of effort and materials. In case easy re-use is

[1] Conceptual- and preliminary design have been defined earlier when discussing spacecraft design, but it essentially means that the models used for the design are limited in the number of details they can handle and as such the outputs/results are not accurate to a high degree. Hence taking into account margins is really important to ensure a proper design.

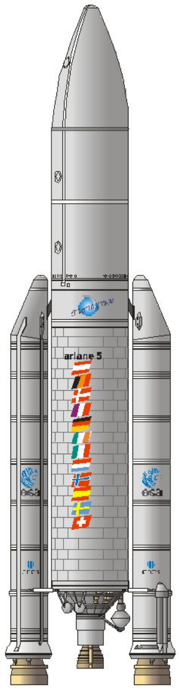
[2] A rocket engine/motor is a reaction engine that carries all the substances necessary for its operation within and hence does not require the use of substances such as atmospheric oxygen, that are to be drawn from the surrounding medium, and thus is capable of operating in outer space as well as at high altitude (above about 20-30 km altitude).

[3] A rocket (from the Italian word rocchetta = spindle) is a missile, spacecraft, or other vehicle that obtains thrust from a rocket engine/motor.

[4] Apogee of a sounding rocket may be up to about 1500 kilometres, but any value in excess of say 120 km is OK.

possible, this can lead to significant cost savings as claimed by for instance SpaceX. Figure 1 also shows that most rocket SLVs are of a long cylindrical design. This is to offer high volume and limit air drag while providing for adequate strength.

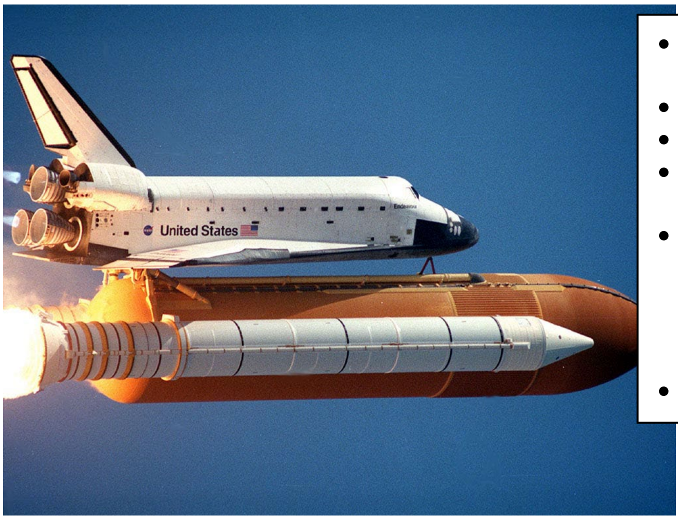
An enlarged view of the European Ariane 5 SLV is shown in Figure 2. This vehicle is used to deliver payloads from Earth surface into geostationary transfer orbit (GTO) or low Earth orbit (LEO). It essentially is a 3-stage (including the boosters) rocket-propelled vehicle with a mass at lift off of 746 metric ton. Payload mass in sun-synchronous (Low Earth) orbit is 10.8 ton, which shows a payload mass to total mass ratio of less than 1.5%. Typical launch costs are about 124 M$ (Fiscal Year 2000[5]) or about 11500 $/kg in LEO or about 20,000 $/kg in GTO. It is the result of an 8 billion US dollar (FY 1996) or development by 12 nations.



- Payload:
  - 6820 kg in 7° GTO
  - 10800 kg in sun-synchronous orbit
- Launch cost: 124 M$ (FY 2000)
- Success rate: 95% (end 2005)
- Number of stages: 2 core stages (in series) + 2 boosters (parallel staged)
- Length: 51.4 m
- Diameter: 5.4 m
- Lift off mass: 746 metric ton
- Payload volume: ≈ 243 m^3

Next to expandable SLVs also (partially) reusable SLVs exist. These are designed to be recovered intact and used again for subsequent launches. For spaceflight, the US Space Shuttle, see Figure 3, the Russian Buran (1 flight only) and the US SpaceShipOne are the only examples of such vehicles flown so far. Of these the Russian Buran was only used once and SpaceShipOne never really reached space orbit (sub-orbital flight only) and hence is more of a sounding rocket design. Only the US Space Shuttle has flown multiple missions.

Figure 2: Ariane 5 launch vehicle (courtesy ESA)



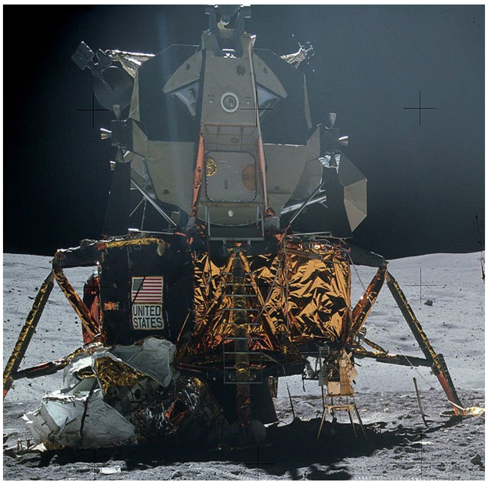
- Vertical take-off, horizontal landing (unpowered descent)
- Payload to LEO: 23090 kg
- Launch cost: 470 M$ (FY 2001)
- Production cost of one Space Shuttle: 1.7 G$ (Endeavor)
- External dimensions (comparable to MD80 (DC9) aircraft)
  - Wing span 23.79 m
  - Length overall 56.14 m
  - Height overall 23.35 m
- Total Mass: 2010.6 mt (metric ton)

Figure 3: Space Shuttle (courtesy NASA)

[5] To learn about current year money values consider correcting the value using average inflation rate. For this various inflation rate calculators can be found on the internet.

Space Shuttle payload mass in Low Earth orbit is 23.09 metric ton and vehicle mass at lift off is 2010.6 metric ton, which gives a payload to total mass ratio of less than 1.1%. Typical launch costs are about 470 M$ or about 20,000 $/kg in LEO (about twice the specific launch cost of an Ariane 5 launch). Recent NASA estimates peg the shuttle program's cost through the end of last year at $209 billion (in 2010 dollars), yielding a per-flight cost of nearly $1.6 billion.

Some SLVs show a quite different design. This is related to them lifting off from a celestial body other than Earth. For instance, Figure 4 shows the Lunar module with on top the ascent stage and an artist concept of the Altair Ascent stage[6] as currently under development by the USA. Both are lunar launch vehicles/ascenders, meaning that they are used to transport some payload from the lunar surface to lunar orbit. The shape of the vehicle as opposed to the launch vehicles shown in Figure 4 clearly demonstrates the effect of absence of a lunar atmosphere on the vehicle shape. Both designs are also single stage rockets, as the required velocity increment for launch into lunar orbit from lunar surface is much less demanding (in terms of Δv) due to the much lower gravitational acceleration associated with ascent flight on the Moon.



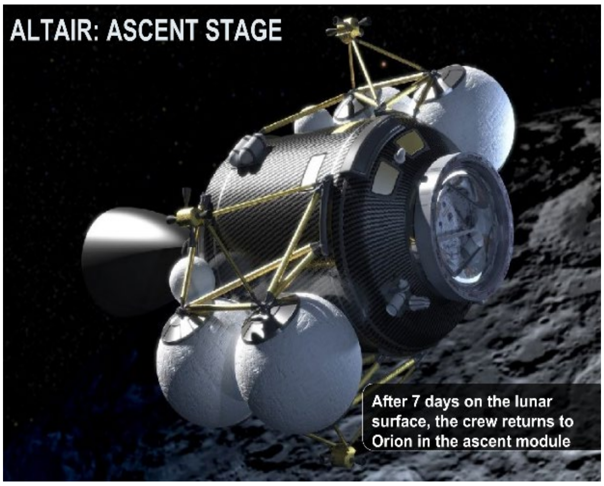


Figure 4: Lunar module with ascender on top (l) and Altair ascent stage (r) (courtesy NASA)

Both vehicles are also manned, meaning that they require a crew cabin and a life support system to enable the crew to survive the harsh space environment. Next to a life support system many systems are needed that can also be found on ordinary spacecraft. From Wikipedia, it follows that the Ascent stage of the lunar module contains next to a crew cabin and an environmental control (life support) system also an instrument system, overhead hatch/docking port; forward hatch; reaction control system; rendezvous radar; VHF and S-band communications equipment and antennae; guidance and navigation systems (primary and backup); active thermal control system, ascent rocket engine; and enough propellant, battery power, cooling water, and breathing oxygen to return to lunar orbit and rendezvous with the mother ship. Its total mass at lift-off is 4,547 kg. Payload mass is 160 kg crew (2 crew) and 108 kg samples. This gives a payload to total vehicle mass ratio of less than about 5.9%. For further details on the Apollo lunar ascent stage, see Wikipedia.

The Altair stage houses a crew of 4, life-support equipment, and propellant for the ascent stage motor and steering rockets. Its mass at lift-off will be 10,809 kg. Payload mass as of yet is unknown. Other systems are considered to be similar to those of the Apollo lunar module.

The main problem of designing SLVs is that a huge amount of energy is needed to reach space. This leads to the need of using large amounts of high energetic materials with as a consequence a quite massive and complex design. SLVs take years and billions of dollars to develop and may cost

[6] The development of this stage has been cancelled by the US in 2010-2011.

many millions of dollars to launch. Because of the complexity of the vehicle, the high development costs and the enormous risks associated with the development of such a vehicle, it is required that the design is done in a proper way to ensure that the design meets all the needs in the best possible way. For this to happen a structured design process is needed otherwise the result may be flawed.

Now, having set the stage, we can start considering how to design an SLV? The next figure gives a schematic of the approach followed in this work.

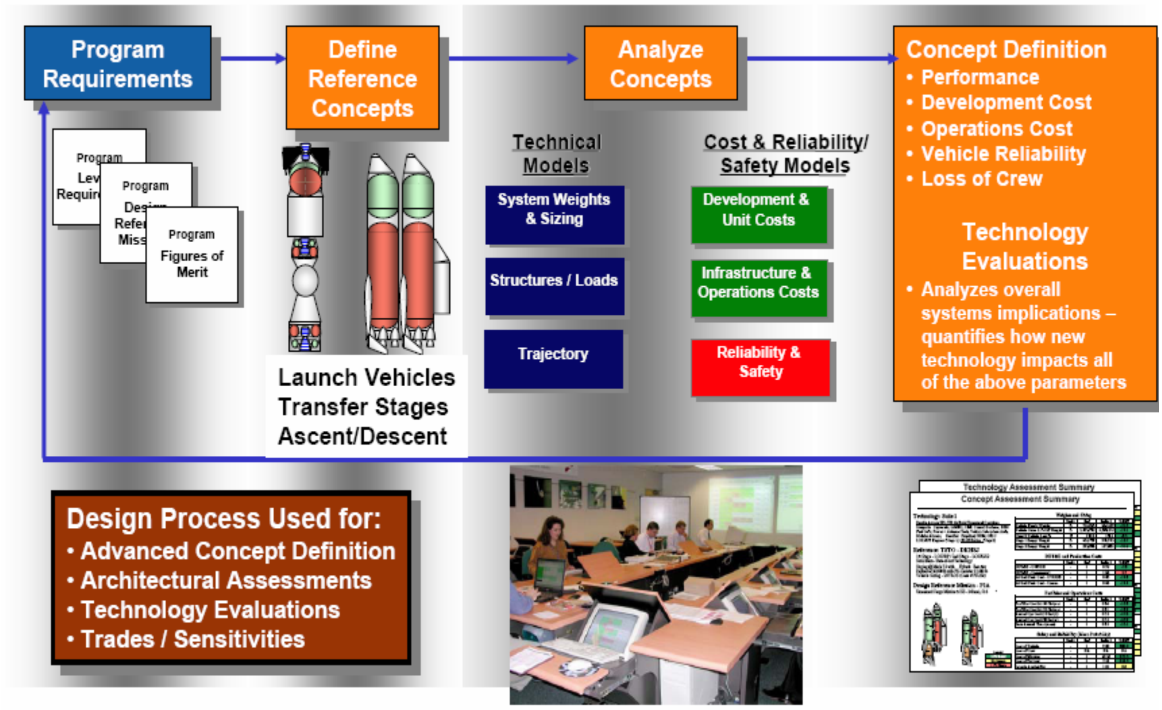


Figure 5: The launch vehicle design process (unknown source)

The design process used in this work essentially consists of 7 steps that together will help to realize a proper design. These 7 steps include:

1. Define the need/problem
2. Establish requirements
3. Set up options/concepts
4. Analyze options/concepts using e.g.
    a. Technical models
    b. Cost, reliability & safety models
5. Compare concepts
6. Make choice (select best concept(s))
7. Evaluate outcome and/or Iterate

The first letters of the 7 steps together make up the acronym DESACME/I. In the ensuing chapters, we will discuss the above steps in some detail with most attention on analysis methods for conceptual and preliminary design.

2 Mission objectives

The design of any launch vehicle actually starts with a need. This need is to be converted in to a mission statement and one or more mission objectives. Generally speaking, the mission (the need) of a space launch vehicle (or space rocket) is to carry a payload from a planetary surface (like Earth) to its destination into space. This need is amongst others illustrated by Euroconsult in its 2014 space market overview wherein an estimated 1155 satellites[7] are forecasted to be launched in the period 2014-2023 or an average of 115.5 satellites per year [Euroconsult]. Likewise the US Federal Aviation Authority (FAA) in its 2015 commercial[8] space transportation forecast [FAA, 2015] forecasts for the years 2015-2017 a steady 25 satellites globally to be launched commercially on a yearly basis in Geostationary Earth Orbit (GEO). This comes down to 250 GEO satellites over a period of 10 year. For Non-GEO orbits (including deep space missions, MEO & LEO) the FAA forecasts for the period 2015 to 2024, 986 payloads to launch commercially.

The need to carry a payload from a planetary surface to its destination into space is central in the mission of a launch vehicle. Hence, the first things that should be decided upon are the planet at which the SLV should be operating, the target orbit, and the mass and size of the payload. Next to that, one should consider that this need should not be realized at any cost and/or risk. For instance, consider the following statement as expressed by the French president Nicolas Sarkozy on June 20, 2009 about the need to replace the Ariane 5 SLV.

> **France wants replacement for Ariane 5 space launcher**
>
> **PARIS (Reuters) - France wants Europe to start looking into a space rocket launcher to replace Ariane 5 at some point between 2020 and 2025, President Nicolas Sarkozy's office said on Saturday.**
>
> The Ariane-5, which is billed as a cost-effective launcher for large satellites, has launched satellites for European telecoms operators, telescopes and scientific space observatories.
>
> But it was time to start working on Ariane 6, the president's office said in a statement.
>
> "(Sarkozy) hopes that the first studies will begin on this new launcher, in cooperation with our European partners and the European Space Agency (ESA) with regard to decisions at the ESA ministerial meeting in 2011," the statement said.
>
> "Ariane 6 should be a very robust launcher with modules and optimized in terms of cost in order to best respond as much to government and commercial needs in a context of increased competition."
>
> A strong euro has led many European operators to launch aboard American and Russian rockets offering lower prices than Europe's Ariane rocket series.

The above statement shows that, with regards to fierce competition in the launch market for lager satellites, attention must be given to the design of a cost-effective SLV as well as that the launch vehicle design should be robust, meaning that the SLV is insensitive to variation in parts, environmental conditions, wear of machines and or ageing of materials (fault tolerant) and flexible to the market.

When considering the success of an SLV design, we find that at least the following elements need to be considered critically:

1. Cost of a single launch or the cost price per kilogram of payload orbited.
2. The design life of the launch vehicle; how many years the system will be active?
3. Robustness of the design; how good it can adapt to changes in the market?
4. Reliability of the launch vehicle[9] or the ratio of successful launches to total launches.

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[7] Notice that this figure does not include microsatellites weighing less than 40 kg at launch, nor do they include classified military satellites.
[8] Commercial forecast excludes military and government satellite launches not open to international competition.
[9] See for a definition of **reliability** the section on spacecraft design and sizing

5. Availability[10] or the total time that the system is ready for action.

In general the following trends (future mission challenges) can be distinguished:
- Increasing payload mass
- Reducing launch cost while maintaining performance (payload mass into orbit)
- Increasing flexibility to allow for a design that can adapt to the market
- Increasing reliability
- Increasing (operational) availability

Currently, focus is not so much on increasing payload mass, but rather on reducing launch cost and enhancing reliability. This is especially the case for commercial launch applications. Reducing launch cost is a goal set already a long time ago with the development of the US Space Shuttle, where it was boldly stated that the design should aim for reducing the launch cost to low Earth orbit with about a factor 10 from say 10000 US$/kg to 1000 US$/kg. It is considered that reducing the cost of launch with a factor 10 is very likely to spur a growth in space applications, with remote sensing and navigation being important candidates for growth. The importance of enhancing reliability is illustrated in the design of the Falcon 9 SLV. This SLV was designed from the ground up for maximum reliability.

Typical goals with respect to some of the above mentioned metrics can be obtained from current and past SLV designs. The (non-exhaustive) list below provides an overview of the objectives of some recent US and European SLVs developments.
- ArianeSpace is developing a replacement for Ariane 5. The replacement (Ariane 6) should be capable of lifting 6.5 tonnes to the Geo-synchronous transfer orbit for €70 million (FY 2014) at a launch rate of 9 per year with maiden flight in 2021. Focus is on a cost-effective and robust SLV (see also the earlier given statement by (former) president N. Sarcozy of France).
- SpaceX has developed the Falcon 9 and Falcon Heavy rocket. Falcon 9 target launch price was set in 2012 at around M$ 54. Its main objective was to launch 8.5 – 10.5 ton in LEO and 4.5 ton in GTO. The Falcon Heavy objectives were set at lifting 54 ton in LEO and 13.2 ton in orbit to Mars at a target price in range of M$ 93-128 M$ (also at 2012 price levels). Both Falcon launch vehicles are to be designed from the ground up for maximum reliability.
- ArianeSpace in 1998 started the development of a small launch vehicle (Vega) capable of launching 300 to 2,500 kg satellites into low Earth orbits. Its aim is to make access to space easier (for small launch masses), quicker and cheaper. The reference mission is a polar orbit bringing a spacecraft of 1,500 kilograms to an altitude of 700 kilometres at a cost (2012 year money) of €32 million/launch at 2 launches/yr or €22 million at 4 launches/yr.
- Boeing in 2012 proposed to develop a Small SLV, with as purpose to launch small payloads of 45 kg into low-Earth orbit. The program is proposed to drive down launch costs for small satellites as low as US$ 300,000 per launch ($7,000/kg) and could be fielded by 2020.
- NASA is currently undertaking the development of the NASA Space Launch System (SLS). This rocket will be the most powerful rocket ever built, and will allow NASA to land astronauts on Mars and captured asteroids, and perhaps other planets and moons throughout the Solar System as well. At the time of publishing first plans, the objective was for the first SLS mission to lift off no later than 2018 (note that a first launch is now planned for 2022, showing delays in the design of the vehicle), sending a capsule (designated Orion) around the Moon. Asteroid- and Mars-bound missions should follow a few years after that.

Once the mission is clear and the main objectives defined, requirements can be generated. This is the subject of the next chapter.

[10] **Availability** = uptime/(uptime + downtime); Uptime is time the system is ready for action even though it is not actually working.

3 Requirements generation

Requirements are expressions that express what is needed. Without requirements no proper design can be made. Hence all designs start with the generation of the top-level requirements. For an SLV, this is best performed by asking questions on what the SLV shall do and how well. Consider e.g.:

- What is the target orbit that the SLV shall reach and what is the related delta-V (ΔV)?
- What payload mass into orbit?
- What payload volume/dimensions?
- What success rate?
- What cost?
- What launch loads?
- What injection accuracy?
- What number of launches per year?
- What time period/frame (when)?
- What payload support?
- Etc.

An important way to obtain answers (quantification) on above questions is to collect data from current and past SLV, to learn what is achievable when and at what cost. It prevents one from setting too extreme requirements or to select those areas where one wants to excel. Data from current and past SLV can be obtained from amongst others the internet (e.g. Encyclopedia Astronautica, and Spaceflight 101 Launch Vehicle Library), Isakowitz International Reference Guide to Launch Systems, or ESA's Launch Vehicle Catalogue. A collection of SLV data is given in appendix A. Hereafter; we will discuss some of the above questions and illustrate how requirements are derived/generated.

Target orbit

Several categories of target orbits can be distinguished.

1. Suborbital flight (like for SpaceShip One, see Wikipedia, and for most ballistic missiles[11])
2. LEO (Low Earth Orbit)
3. MEO (Medium Earth Orbit)
4. GTO (Geostationary Transfer Orbit)
5. GEO (Geostationary Earth Orbit)
6. Deep space

Which target orbit is selected usually depends on the market that is being targeted, but when the target orbit is known, the mission characteristic velocity, important for the design of the launch vehicle (see later in this lecture series), can be determined. Some typical values depending on the target orbit can be obtained from the next table.

Table 1: Typical ΔV value(s) for rocket launch vehicles depending on target orbit and vehicle size

Maneuver	ΔV, km/s
Flight of Spaceship One to 100 km altitude	1.4 km/s
V2-rocket	1.6 km/s
Earth surface into LEO	9.1-10.2 km/s (Scout: 9.1 km/s, Vega: 10.2 km/s)
LEO to Earth surface	Orbital maneuvering burn to lower perigee into the atmosphere, atmospheric drag takes care of the rest.
Moon surface into Low Lunar Orbit (LLO)	2.0-2.6 km/s (2.2 km/s Apollo ascent stage)
LLO to Moon surface	1.6-2.5 km/s (2.5 km/s Apollo descent stage)
Mars Surface to low Mars orbit	4.1-5.7 km/s
Low Mars orbit to Mars surface	~4.7 km/s (atmospheric drag helps to slow down)

[11] A **ballistic missile** is a missile that follows a sub-orbital ballistic flight-path with the objective of delivering one or more warheads to a predetermined target.

Values in the foregoing table include losses associated with gravity, aerodynamic drag and steering. For instance, to reach LEO, ideally a velocity increment of 7.8 km/s is needed. Because of losses, the ideal velocity increment that should be delivered by the launch vehicle is 1.4-2.2 km/s higher. Information on how to obtain more detailed values of the mission characteristic velocity, as well as values that apply to missions not included in the table, see the material presented as part of the course AE1110-II or consult the simple model entitled "Velocity increment needed to reach orbit" available on Brightspace, see the tab leading to the reader of ADSEE I.

Mass into orbit

Figure 6 provides an FAA overview of number of spacecraft destined for GEO in the period 2004-2017 distinguished after spacecraft mass. Payload mass classes are defined as:

- Small: below 1200 kg
- Medium: 1201 to 2500 kg
- Intermediate: 2500 to 4200 kg
- Heavy: 4201 to 5400 kg
- Extra heavy: above 5400 kg

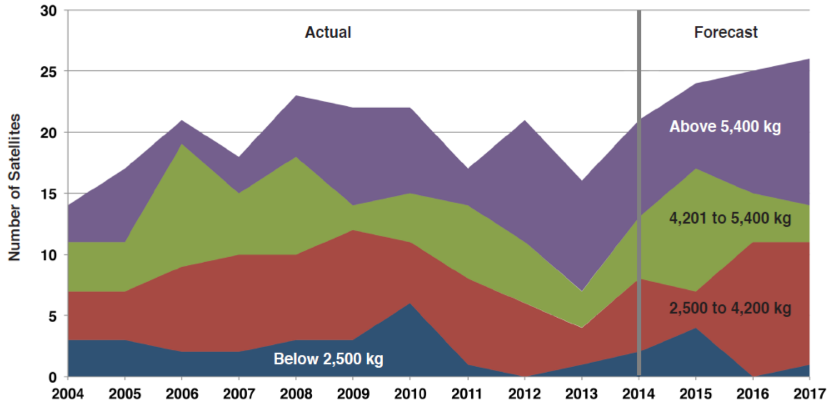


Figure 6: Trend in launch mass of spacecraft into GEO orbit [FAA, 2015]

Typical values of payload mass into orbit are given in the annex A for various categories of orbits. For interplanetary orbits, data is included providing mass as a function of the characteristic energy C3. This is a characteristic value which is equal to the square of the hyperbolic excess velocity (Earth escape velocity). Notice that the payload mass reduces with increasing distance to Earth as more energy is required to reach those orbits. Highest payload masses are for launch into Low Earth Orbit. Largest mass ever orbited into Low Earth orbit was around 118 ton by a Saturn V rocket.

Payload volume/dimensions

The payload bay under the fairing/shroud is to a large extent determined by the (stowed) size of the spacecraft (and vice versa). For most launchers the payload volume is mostly cylindrical with a more or less conical cap to allow for a proper streamline shape of the fairing. Figure 7 shows the Vega fairing with its usable volume (inner line). The usable volume consists of a cylindrical shape of diameter 2.38 m and height 3.515 m. On top of this cylinder a truncated conical volume is available of height 2.00 m and of smallest diameter 1.06 m. Diameter of payload bay is smaller than the outside diameter of 2.60 m as to allow for some thickness of the fairing and to make sure that during the launch the spacecraft never gets in contact with the fairing. The lower portion of the fairing volume

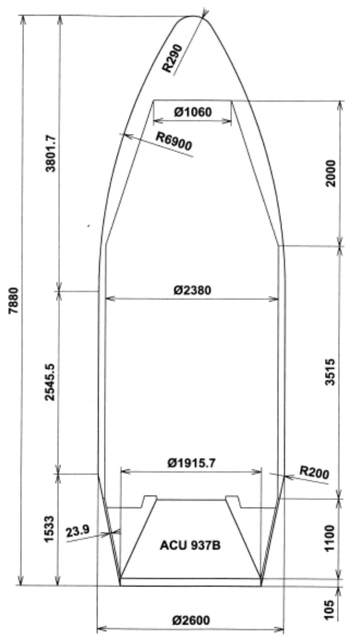


Figure 7: Vega Fairing external dimensions and usable volume [LVC]

is to hold the payload adapter which connects the spacecraft to the launcher. Typical payload dimensions can be obtained from, for instance, appendix A, tables A-4 and A-5.

Injection accuracy

Important for the design of an SLV is the injection accuracy it should be able to reach. In case the accuracy is too low, spacecraft designers need to incorporate in their design features that allow for reducing the injection errors made. In this respect, lower injection errors will bring benefits in terms of, amongst others, mass, cost and complexity of the spacecraft. Typical orbit injection accuracy for a range of space launch vehicles is given in Table 2. Injection accuracy for some SLVs depends amongst others on the target orbit selected.

Table 2: Injection accuracy (3-sigma) [Falcon], [LVC]

Launch vehicle	Target orbit	Inclination [deg]	Perigee [km]	Apogee [km]	Mean altitude [km]	Semi major axis [km]	Period [s]
Falcon 9	LEO	0.1	10	10			
	GTO	0.1	7.4	130			
Falcon 1	200 km circular	0.1	10	20			
H2	GTO	0.03	220	250			
	450 km circular	0.03			20		
Ariane 5G	GTO	0.06				120	
Proton M	200 km circular	0.025	6	16			8
	1000 km circular	0.05	10	10			100
	GTO	0.5	400	150			550
Soyuz	200 km circular	0.06					22
Taurus	400 km circular	0.15			30		
Pegasus XL	740 km circular	0.15	19	90			
Pegasus XL with HAPS	740 km circular	0.08	15	15			

Injection accuracy requirements may e.g. determine the selection of the type of rocket power-plant and/or require some means of thrust control, see later in this work.

Launch loads

During launch, loads are transmitted to the spacecraft through the spacecraft interface (launch vehicle or payload adapter). These loads include steady state accelerations, low frequency accelerations, vibrations (including acoustic loads associated with sound waves produced by rocket boosters/engines reflected from the surroundings), and shocks. For instance, Figure 8 shows typical launch loads as produced during the ascent flight of a rocket launcher. High loads occur especially at burn out of stages. For the example shown in the figure maximum load occurs at burn out of the third stage and approaches roughly 7g.

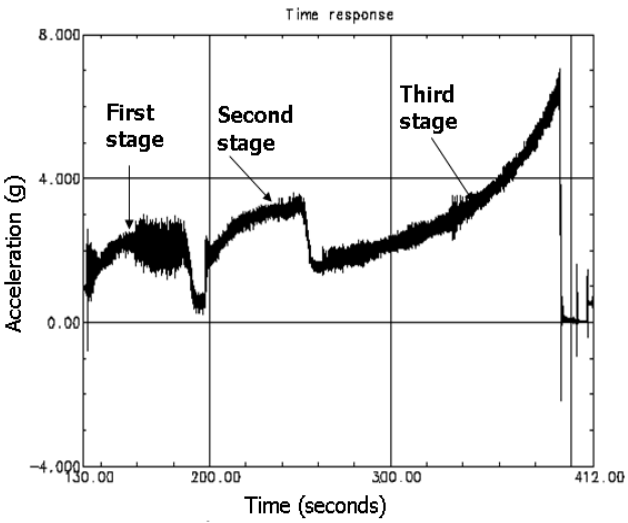


Figure 8: Typical launch loads [LVC]

In an earlier section in this course we learned about the importance of designing a spacecraft for the proper frequency. In this respect, it is mentioned that also the launch vehicle structure itself must be designed to withstand the random vibration loads generated by the vehicle.

From existing SLVs a good overview can be obtained what is currently to be expected with respect to the various (acceleration and vibration) loads. Such data can for instance be found in the respective launch vehicle's User Manual. From known data, values that should be endured by the spacecraft can be derived. For an SLV, limiting the launch loads too much, may lead to a costly vehicle design because of excessive gravitational loss. On the other hand if launch loads are too high, this may limit the number of spacecraft that can endure the launch environment. It is the task of the vehicle designer to make sure that the loads produced by the vehicle are acceptable for the payload at the right cost, etc. Anyway, from a viewpoint of competition, the lower the loads the better this is to attract payloads for the specific launch vehicle.

Cost
It is important for the designer to control the launch cost as much as possible. Best would be if the launch cost are reduced to zero, but this is not realistically feasible. Information on launch costs can be derived from various sources, see for instance [Futron]. Some typical data have been collected in Appendix A.

However, next to launch costs we also need to take into account other aspects like payload mass. For instance, [Futron] reports for the US Delta 2 rocket a launch cost of 55 million US$ as compared to 165 million US$ for an Ariane 5G (all cost expressed in year 2000 money). This indicates that the Delta 2 rocket is much cheaper than the Ariane 5G. However, according to data also taken from [Futron], Delta 2 is only capable of launching 5144 kg into LEO, whereas Ariane 5G is capable of launching 18000 kg into LEO. For Delta 2, this comes down to 10690 $/kg versus 9100 $/kg for Ariane 5G. This is due to the economies of scale that a larger vehicle provides (consider e.g. a small ship versus a large ship).

A better measure for cost is considered the specific launch cost, i.e. cost per kg of payload into orbit.

Example: Ariane 5G specific launch cost
Ariane 5G is capable of lifting:
- *18000 kg into a 550 km, 28.5° inclined orbit*
- *6640 kg into 560-35890 km, 7° GTO*

Given a total launch cost of 165 M$, this leads to a specific launch cost (cost per kg) of:
165 M$/18000 kg = 9100 $/kg into LEO (550 km, 28.5° inclined)
165 M$/6640 kg = 25000 $/kg into GTO

Typical specific launch costs for Western SLVs (year 2000[12] money) are [Futron]:
- LEO: 8000-16000 US$/kg
- GTO: 20000 US$/kg for heavy SLVs and 36000 US$ for small SLVs

For non-Western SLVs (China, Russia, and Ukraine), specific launch cost is roughly 50% lower.

That specific launch cost increases with increasing distance (orbital altitude) is because the further a rocket travels, the lower the payload mass that can be carried onboard. So, when reporting specific launch cost, it is good practice to also report the payload mass and target orbit considered.

The challenge for the rocket engineer is to control the cost of the launcher. That usually means he has to control both the cost of the rocket itself (the hardware), but also the cost of launch services, government support and range safety. The former makes up roughly 60-80% of the launch costs, whereas the latter contributes roughly 20-40%, see Table 3.

[12] To convert year 2000 US$ to year 2022 US$, by correcting for average inflation, multiply the numbers given by a factor 1.72. The latter has been determined using an inflation calculator available on the internet.

Table 3: US Expendable launch vehicles-Historical recurring cost breakdown (%) [Sackheim]

LAUNCH VEHICLE COST SUMMARY				
COST ELEMENT BY PERCENT	DELTA 7925	ATLAS CENTAUR	TITAN III	TITAN IV
HARDWARE	66	78	82	59
LAUNCH SERVICES	20	16	12	22
GOVERNMENT SUPPORT	12	6	6	19
RANGE SAFETY	2	Included above	Included above	Included above
TOTAL	100	100	100	100

<u>Reliability</u>

SLVs are to be designed for a certain reliability. Nobody is willing to put an expensive payload on an SLV that has a low reliability. Historical SLV reliability figures are in the range of 90% and better, see annex B. For instance, of the 4378 space launches conducted worldwide between 1957 and 1999, 390 launches failed (success rate is 91.1%):

$$R = (4370-390)/4378 = 0.911 = 91.1\%$$

However, this also includes failures that have nothing to do with the SLV and/or outright design failures. Typical reliability figures used in design are much higher and may easily be 0.98 or 98% (only 2 failure on every 100 launches) for unmanned SLV and 0.998 or 99.8% (or higher) for manned SLV. The challenge for the engineer is to come up with a design that has a high reliability. In case of a manned SLV, this may require the addition of a launch escape stage, see later in this work. For more details on launch vehicle reliability, see later section on SLV reliability.

<u>Number of launches per year</u>

According to [Clark], a total of 92 space launches were recorded worldwide in 2014 and 81 in 2013. Clark also indicates that Russia, USA and China are responsible of 80% of the launches. Russia had the most liftoffs with 36 orbital launch attempts — 34 were deemed complete successes — and the United States came in second with 23 space launches, with all but one reaching its intended target. Chinese rockets were responsible for 16 launches. Of these 80-90 launches per year roughly 30-40% is considered open to internationally competitive launch service procurement, meaning about 24-36 commercial launches per year launching roughly 115 spacecraft. Of these the US Federal Aviation Authority (FAA) predicts 17-18 commercial launches to GEO per year using dual-manifesting[13]. For Non Geostationary Earth orbits (including deep space missions, MEO & LEO) the FAA forecasts for the period 2015 to 2024, 986 payloads to launch commercially, driving 131 launches with multi-manifesting, reflecting an industry planning to launch more medium and small-class payloads in clusters, instead of increasing the demand for individual launches. Multi-manifesting is also applied for micro-spacecraft (not included in above figures). For illustration, we mention the QB50 mission, where it is planned to launch a constellation of 50 small satellites (of mass less than 3 kg) with a single launch, see https://www.qb50.eu/.

[13] Dual-/multi-manifesting is the phenomenon that a single launcher orbits two (dual) or more (multi) satellites at the same time.

4 Define Concepts

Defining concepts actually is about generating design solutions. It is not about what the system is supposed to do and how well, but it is about how the system shall be designed to achieve the targets (requirements) set. Defining concepts generally starts with a listing of all possible design options (including technology options) and the generation of possible vehicle configurations including a baseline vehicle that is based on an existing vehicle for benchmarking. Next to a baseline concept a few advanced concepts using advanced technology (advanced materials, improved rocket engines, etc.) may be added. One also may consider adding a breakthrough concept that partially uses technology that has only been proven in laboratories or in other fields. Typical such technologies could include applying air-breathing propulsion and/or wings to the launcher. However, the more concepts are generated the more concepts need to be analyzed, which may take a lot of time.

4.1 Current space launch vehicles

A range of current launch vehicles has already been introduced in Figure 1. All vehicles shown in this figure are rocket propelled multi-stage vehicles, where the term multi-stage refers to a vehicle driven by several rocket systems (referred to as stages[14]) in sequence. The lowest, or first stage, ignites and then lifts the vehicle at increasing velocity until exhaustion of its propellants. At that point the first stage drops off, lightening the vehicle, and the second stage ignites and accelerates the vehicle further and so on. In case of a series staged (single-body) rocket, stages are mounted on top of each other like for the European Vega , see Figure 9 (left), and Ariane 1, US Saturn 1 and Saturn V, Ukranian Zenit 2, and the Russian Zenit 3SL and Moon rocket. Other vehicles show a combination of series and parallel staging, i.e. stages which are attached alongside another stage, like for the European Ariane 5, US Titan IV, US Space Shuttle, Russian Buran, Proton, Soyuz, Japanese H2, Chinese Long March 2 and Indian PSLV. As an example, Figure 9 (right) shows the Soyuz SLV. Clearly can be seen 4 booster stages at the bottom of the vehicle that form the first stage of the vehicle.

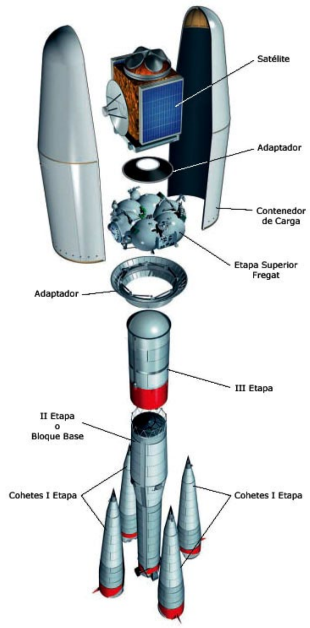


Figure 9: Vega (left) and Soyuz rocket exploded view (right) (courtesy ESA)

Two of the vehicles shown in Figure 1 are the US Space Shuttle and the Russian Buran. The latter two are semi-re-usable vehicles as part of the vehicle (the orbiter) returns to Earth at end of mission for re-use. Clearly visible for the two vehicles are the aircraft like features which allows for the

[14] A stage is a separable part of a rocket which contains its own engines and propellant.

vehicles to realize braking in the atmosphere and to provide the vehicle with a cross and down-range capability, just like a sail plane.

4.2 Near term future launch vehicle concepts

In this section we will discuss Ariane 6, the US SLS and Falcon 9 as examples of near term future SLV concepts. The vehicles are referred to as near term concepts as the design of these vehicles will be based on applying and improving existing technologies and not on developing new technologies in a major way.

Ariane 6

For the successor of Ariane 5 over 700 rocket-propelled, expendable SLV concepts have been evaluated by ESA on several different factors including cost, payload mass, reliability, development cost and risk. The main concepts differed in terms of propellants used, the number of engines applied, series staging only or with boosters (parallel staging) and in the size of the stages. The main concepts considered were:

- KH: Kerosene-liquid oxygen propelled first stage (as compared to Ariane 5's liquid hydrogen-liquid oxygen first stage).
- HH: First stage without the solid rocket boosters as used on Ariane 5. First stage is comparable to the current Ariane 5 core stage, but with 3 main engines instead of 1 as for Ariane 5.
- HH-PPB: First stage with various small (P20; number indicates propellant mass in ton) solid rocket boosters. First stage equipped with 2 main engines
- PPH: Solid propelled first and second stage: P340 first stage, P110 second stage.
- Multi-P: A modular design using common solid propelled propulsion units
- PPH-PPB: Fully solid propelled design with P180 first stage, P110 second stage, P40 boosters
- KH-CCB/HH-CCB: Hydrogen or Kerosene first stage (with liquid oxygen as oxidizer) using common core boosters (CCB).

The one common propulsion unit in all of these concepts is the use of a liquid hydrogen-liquid oxygen upper stage with a single engine.

The concept shown in Figure 10 (left) is surprising in that it has little similarity to the above concepts. It uses four common rocket stages carrying 145 tons of solid rocket propellant, with two or three of these "cores" making up the first stage and a single one making up the second stage. An upper stage would use a single cryogenic Vinci engine and use hydrogen and oxygen to propel the payload into the target orbit. The payload to GTO is 6.5 tons for the full configuration, and a version using two solids in the first stage would bring 3.4 tons to GTO. The goal is to cost only €70 million (FY2012), or about $95 million dollars. A significant cost reduction is expected from having common rocket stages, which would allow for increasing series production size.

The current (yr. 2015) baseline concept is shown in Figure 10 (middle and right). Like Ariane 5, it will use parallel staging with the core rocket consisting of two rocket stages. The first one will use a Vulcain engine and hydrogen and oxygen as propellants similar to the cryogenic main stage of the Ariane 5 rocket. The upper stage would again use a single cryogenic upper stage based on the use of the Vinci engine. The Ariane 6 will come in two variants — one with two strap-on solid rocket boosters named the Ariane 62 and another with four auxiliary rocket motors called the Ariane 64. This will allow addressing two different payload mass classes in a more optimum way. The strap-on solid rocket boosters will be of an identical design as used for the European Vega rocket to allow for reducing costs.

Figure 10: Two different Ariane 6 concepts (courtesy ESA)

<u>Space Launch System (SLS)</u>

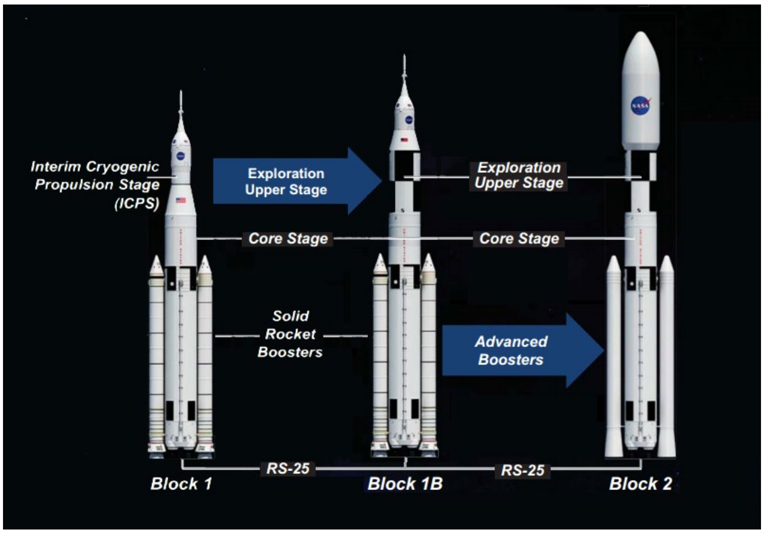


Figure 11: Space Launch System (NASA)

The Space Launch System is also of a relatively conventional expendable design. Block I, the first and most simple design, consists of a core stage that's lifted almost straight from the Space Shuttle: It has two Space Shuttle Solid Rocket Boosters (SSRBs), and a first stage that's fashioned out of a converted Space Shuttle External Tank (that big red cylinder thing, see Figure 3, — but on the SLS it'll be painted white). Together with a modified Delta Cryogenic Second Stage (modified from the Delta IV launch vehicle), Block I will be able to lift around 70 metric tons into low-Earth orbit. There is only expected to be one launch of the Block I variant. If all goes to plan, it will launch sometime in 2017 or 2018 and send an un-crewed Orion capsule on orbit around the dark side of the Moon.

The next variant of the SLS, Block IB, will use the same core stage as Block I — but instead of the modified Delta IV second stage, it'll have the brand-new Exploration Upper Stage. The EUS has a lot of fuel and four RL10 rocket engines, boosting the total payload capacity to around 110 metric tons to LEO. Finally, at some point in the 2030s, Block II will arrive, which replaces the two SSRBs with new, "advanced boosters." Block II will be capable of lifting around 155 metric tons to LEO.

Falcon 9

Falcon 9 is a two-stage, expendable, liquid oxygen and rocket grade kerosene (RP-1) powered launch vehicle. As the first rocket completely developed in the 21st century, Falcon 9 was designed from the ground up for maximum reliability. Falcon 9's simple two-stage configuration minimizes the number of separation events -- and with nine first-stage engines, it can safely complete its mission even in the event of an engine shutdown. Diameter of the fairing is 5.2 m. The first stage of the vehicle has been designed to be reusable. For that, the vehicle has a coating ablative cork and a landing system to land gently on a recovery vessel. Also materials used on the vehicle are resistant to salt-water corrosion. For the first several flights, the first stage will not be re-used as recovery technology is evaluated and improved to lead to a re-usable stage in a later stage of the Falcon 9 operational life.

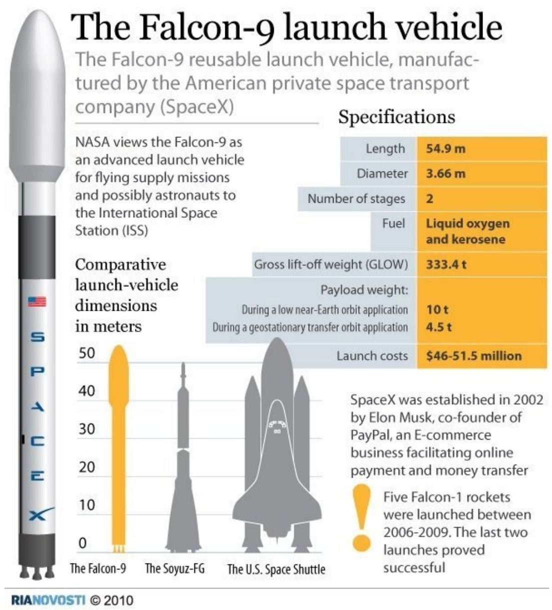


Figure 12: Falcon 9 launch vehicle [Arora]

4.3 More advanced vehicle concepts

In this section, a number of advanced concepts are discussed that introduce more airplane-like features in an attempt to reduce cost of access to space through reusability. The concepts introduced include Boeing Small Launch Vehicle, LauncherOne, and an advanced successor of Ariane 5, which did not make it.

Boeing Small Launch Vehicle

An even more futuristic concept for a small launch vehicle is the Boeing Small Launch Vehicle as shown in Figure 13. The figure shows an air-launched three stage to orbit launch vehicle concept. The first stage of the three-stage launcher would be an air-launched supersonic aircraft accelerating to a speed of Mach 4.5 (5,512.7 km/h) at 19,000 m, while the second stage would be a hypersonic aircraft which would accelerate the vehicle to Mach 10 (12,250 km/h) at an altitude of 29,000 m. Both of the first two stages would be reusable to reduce launch cost, and both stages would carry only fuel, and obtain their oxygen for combustion from the Earth's atmosphere. The third stage would be powered by a rocket, roughly 4.9 m long, to complete the acceleration of the 53 cm × 97 cm payload to orbital velocity. The carrier aircraft is projected to be a Scaled Composites WhiteKnightTwo.

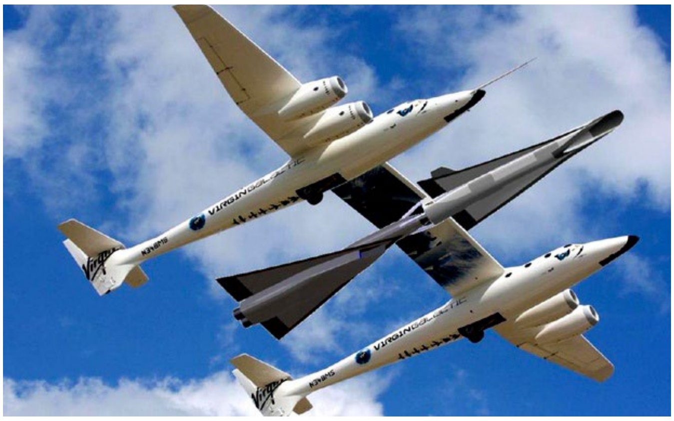


Figure 13: Boeing Small Launch Vehicle [Wikipedia]

LauncherOne [Virgin Galactic]

Virgin Galactic is developing a small satellite launch vehicle dubbed LauncherOne, see Figure 14. It is a two-stage vehicle capable of carrying up to 225 kilograms to a low inclination low Earth orbit for prices below $10 million. The rocket will be launched from Virgin Galactic's proven WhiteKnightTwo aircraft. Due to the extreme flexibility of air launch, LauncherOne potentially offers reduced infrastructure costs in addition to a wide range of possible launch locations tailored to individual mission requirements and weather conditions in this mission. The use of WhiteKnightTwo for other activities, such as SpaceShipTwo's tourism and microgravity science flights, will spread the cost of the carrier aircraft in order to achieve the $10 million launch price target for LauncherOne.

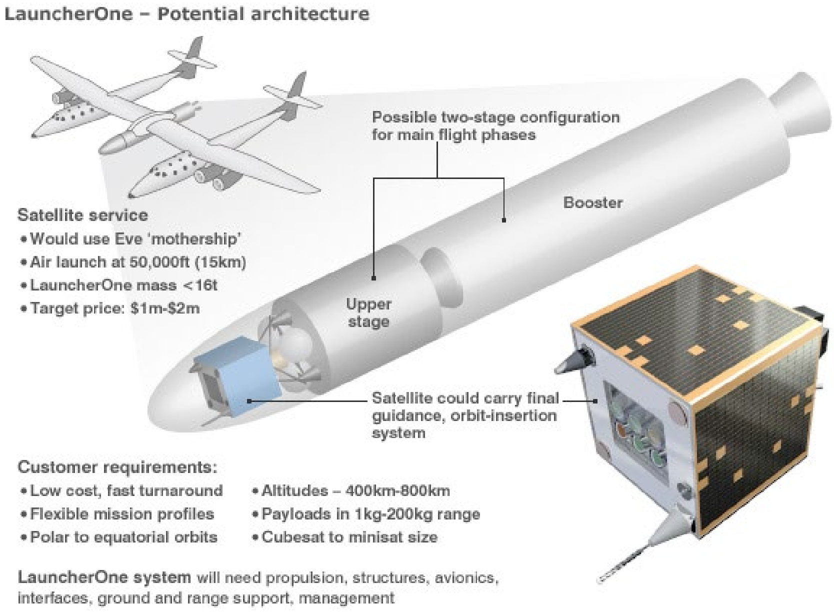


Figure 14: LauncherOne small satellite launch vehicle [Virgin Galactic]

Ariane 5 successor

A more futuristic successor for Ariane 5 than the expendable rocket vehicle shown in the previous section vehicle is shown in Figure 15. It shows a liquid fueled fly-back booster concept using a cluster of Vulcain engines (The same engine that now drives the core stage). This design allows for the booster to fly back to some landing site, where they can be refurbished for the next flight, thereby providing savings in booster production costs.

Figure 15: Future Ariane 5 successor [Prampolini et al]

More advanced concepts

Figure 16 shows a further collection of advanced launcher concepts studied including some that offer truly air-plane like features, such as:

- Wings to allow for return flight to the launch site and or to provide for lift to assist in attaining altitude
- Air-breathing engines (only fuel mass has to be carried)
- Undercarriage to allow for horizontal landing and take-off from "conventional" airports.

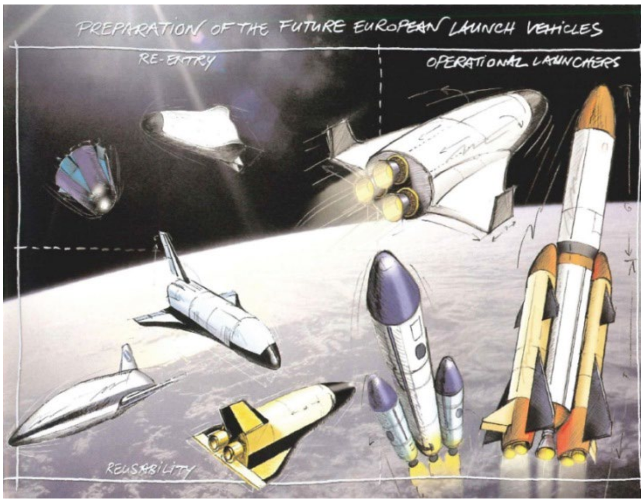


Figure 16: Future launcher concepts (courtesy ESA)

For instance, the potential benefit of using air-breathing propulsion is that it allows for much higher values of specific impulse[15] as compared to rocket propulsion, see Figure 17.

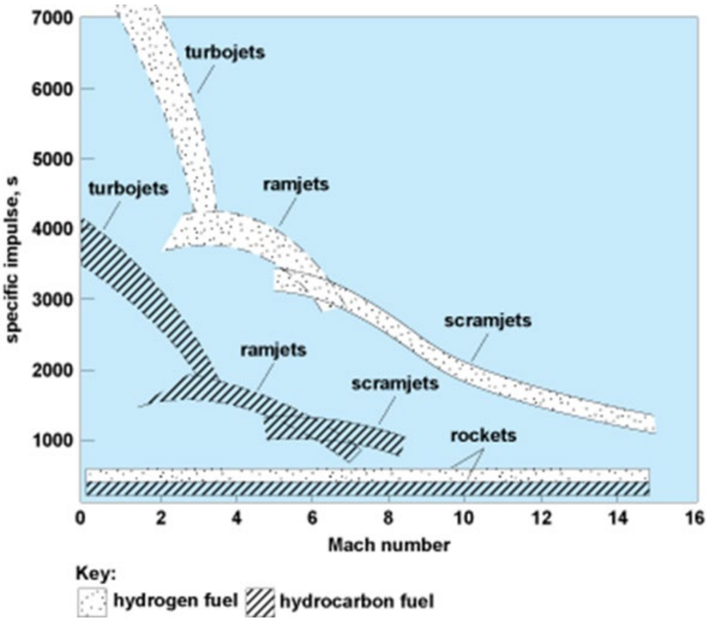


Figure 17: Specific impulse of air-breathing propulsion versus rocket propulsion

The figure clearly shows that up to about Mach 14 a significant improvement in terms of specific impulse and hence reduction in propellant mass is possible. However, next to specific impulse, we should also consider the attainable thrust level, which is much less for air-breathing propulsion than for rocket propulsion. Moreover, the thrust of an air-breathing jet engine drops both with increasing altitude and flight velocity, whereas rocket thrust is only weakly dependent on altitude and invariant with flight velocity. Above about 14-17 km, air-breathing propulsion is not able to provide the acceleration power needed for space launch.

[15] Specific impulse is a measure for the amount of expellant to be carried on board. The higher the value of the specific impulse, the lower this mass is.

5 Analyze Concepts

All design concepts need to be analyzed in some detail to allow determining the effect of design changes on the functional[16] and operational[17] performance of the vehicle as well as on the vehicle configuration and such characteristics like mass, size, cost, reliability, development schedule, operations, characteristics, etc.

In general, this means that we need to have (or develop) analysis methods/models for evaluating each of the aforementioned aspects (performance, cost, etc.). A typical schematic of such an analysis tool is given in next figure.

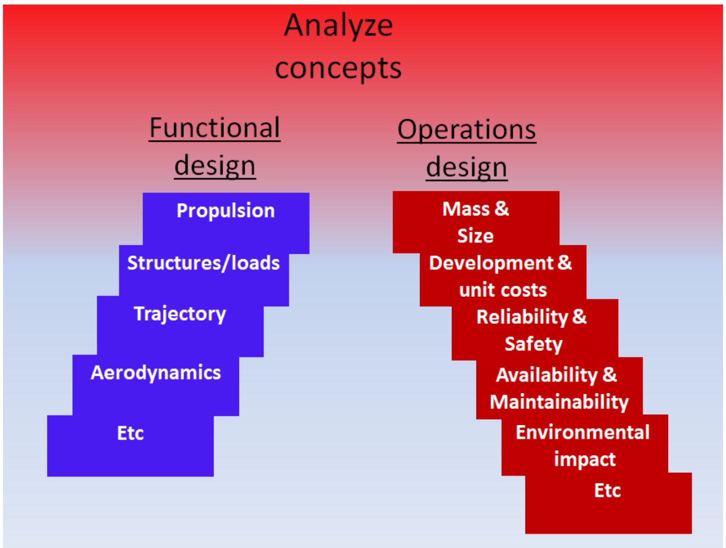


Figure 18: Typical analysis models needed to evaluate launch vehicle designs

First models generally tend to be quite simple as little design information is known. For instance, one would like to know the relation between payload mass and SLV Gross Lift-Off Mass (*GLOM*) or SLV initial mass $M_{initial}$ or M_o for an SLV that is to launch some payload from Earth surface into LEO.

Data on payload mass and GLOM for such SLV are for instance given in Table A-1 of Appendix A. From this data, it follows that current SLV are capable of launching 0.68-3.91% of total SLV as useful/payload mass into orbit, with the higher percentages for the larger SLVs. The ratio of payload mass to GLOM is ratio is defined as the payload fraction (λ) of the vehicle and is given by:

$$\lambda = \frac{M_u}{M_o} \quad [1]$$

When we plot payload mass versus vehicle initial mass, see Figure 19, the data points show a more or less linear relationship between the two parameters. Adding a linear curve fit leads to the following relationship between payload mass and Gross Lift-Off Mass (GLOM) (all values in kg):

$$GLOM = 26.04 M_{Payload} + 88235 \quad [2]$$

The goodness of fit, see section on spacecraft design and sizing, is demonstrated by $R^2 = 0.987$ and RSE (Relative Standard Error[18]) = 38.8%. Using this relationship a quick estimation of SLV GLOM can be made for a given payload mass.

[16] Functional performances are related to the functions the launcher is supposed to perform.
[17] Operational performances are related to how well the launcher can be operated. It typically is measured in terms of (operational) costs, reliability, availability, maintainability, and safety.
[18] This term is sometimes also referred to as Standard Error of the Estimate (SEE).

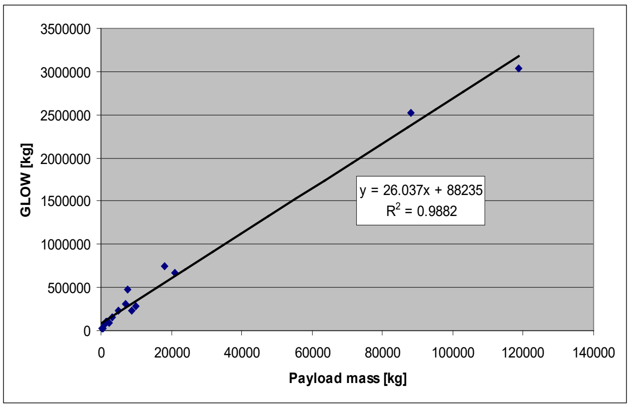


Figure 19: Gross Lift-Off Mass (GLOM) versus payload mass

> ***Example: Estimation of GLOM***
> *Given a payload mass of 160 kg, we estimate a GLOM of 92.4 ton with an RSE of 38.5% of the estimated value or 35.6 ton. It follows with an 68% probability that the actual vehicle mass is within ±35.6 ton of the Most Likely Estimate (MLE), whereas we have a probability of about 95% that actual vehicle mass will be within range 92.4 ton ± 2 x 35.6 ton (21.2 - 163.6 ton).*

Some more detailed relations of GLOM and payload mass can be found in appendix E. section A.

Next to a model for estimating GLOM, other equally simple models can be developed that either estimate vehicle cost, size, reliability, availability, etc. allowing for a fast estimate of important vehicle characteristics. A drawback of using simple models is that the answer may hold quite some uncertainty (a large spread). To reduce the spread and hence to increase accuracy more detailed modelling is needed.

In the next few sections, we will introduce some more advanced (but still simple) methods that will allow for analyzing the performance (delta-V capability) of single stage rockets (launch vehicles and landers) and multi-stage vehicles based on the rocket equation and information on the basic SLV configuration, effective exhaust velocity and some general mass characteristics. In a final section, we will discuss the effect of thrust level on mission characteristic velocity by limiting the gravity loss.

5.1 The rocket equation

The rocket- or Tsiolkowsky equation can be written as:

$$\Delta V = w \ln \frac{M_{initial}}{M_{final}} \qquad ; \qquad M_{initial} / M_{final} = \Lambda \qquad [3]$$

It essentially relates the velocity change *ΔV* that can be realized by a rocket propelled vehicle to its initial and final mass (*M*) and some effective velocity (*w*) with which the (rocket) propellants are expelled. The latter hence is also referred to as effective rocket exhaust velocity or shortly effective exhaust velocity. The ratio of initial to final vehicle mass, here denoted by Λ, is referred to as the vehicle mass ratio while the inverse of this ratio is referred to as vehicle inert mass fraction (a value smaller than one). A derivation of the rocket equation is given in AE1110-II. In this course it was also introduced a relation between the specific impulse (I_{sp}), i.e. the total impulse delivered by the rocket per unit of propellant weight, and the effective exhaust velocity:

$$w = I_{sp}\, g_0 \qquad [4]$$

Here g_o is Earth gravitational acceleration at sea level. Both specific impulse and the exhaust velocity depend on the type of rocket propulsion selected. This will be explained later in more detail.

Another aspect introduced in AE1110-II is that with varying altitude the exhaust velocity, and hence also the specific impulse, of a rocket tends to change. This is related to the change in atmospheric pressure with altitude. As an example, it is mentioned that at sea level (high atmospheric pressure) the Space Shuttle Solid Rocket Booster produces an I_{sp} of 242 s and in vacuum (absence of atmosphere) 268.6 s. We write $I_{sp,sealevel}$ = 242 s and $I_{sp,vacuum}$ = 268.6 s. For the V2 rocket, these values are 270 s at sea level and 312 s in vacuum. The difference between the values for the Space Shuttle and the V2 can be explained from the use of different propellants, which will be highlighted later in this course in more detail, and from differences in engine design. The latter will be dealt with later in the curriculum.

Notice that the effective exhaust velocity of the Space Shuttle rocket booster varies from about 2375 m/s (242 s x 9.81 m/s^2) at sea level to 2635 m/s in vacuum. For the V2 rocket these values are 2650 and 3060 m/s, respectively.

Because of the specific impulse (and hence also the effective exhaust velocity) changing with altitude, we use an average value over the burn/action time of the rocket to obtain a "mission average" value:

$$\overline{I}_{sp} = \frac{\int_0^{t_b} T\, dt}{\int_0^{t_b} m\, g_0\, dt} = \frac{\int_0^{t_b} m\, w\, dt}{\int_0^{t_b} m\, g_0\, dt} = \frac{\overline{w}}{g_0} \qquad [5]$$

Here T and w are instantaneous rocket thrust and effective exhaust velocity, respectively and m is propellant mass flow rate. Notice that if a rocket stays at constant altitude, both specific impulse and exhaust velocity remain constant unless we change the settings of the rocket motor. The latter will be dealt with in later courses.

Next figure shows some results obtained from the rocket equation. It shows the velocity change as a function of rocket mass ratio for two different (effective) exhaust velocities.

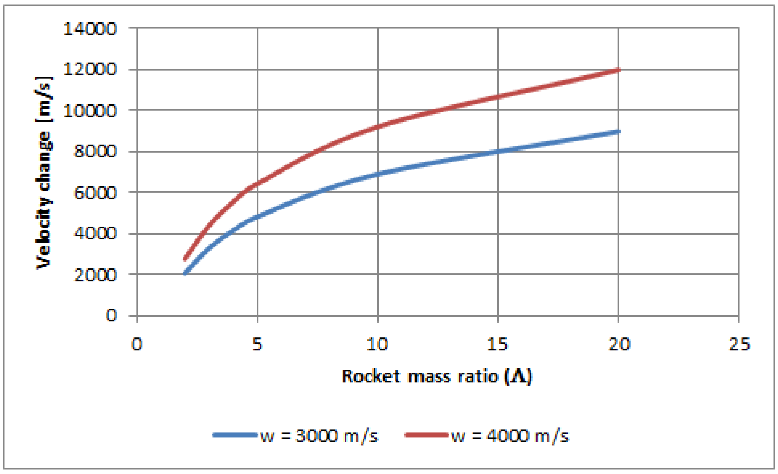


Figure 20: Velocity change (delta-V) as a function of rocket mass ratio for two different values of the average (effective) exhaust velocity

From the rocket equation we learn that to realize a large velocity change (meaning we can travel deeper into space), we need to realize, see also above figure:

1. large vehicle mass fraction
2. high average (effective) exhaust velocity

How we can engineer the vehicle mass ratio (or fraction) and the exhaust velocity will be dealt with in some detail hereafter.

5.2 Single stage rockets

Next figure shows a typical layout of a single stage rocket. It essentially consists of a rocket stage and a payload (also referred to as useful load) with a nose cone on top. For reasons of simplicity, we consider the rocket stage to consist of (usable) propellant and structure (everything else, including engines, stabilization fins, tanks, piping, and aerodynamic control surfaces).

For this rocket, we can write the initial (at start of the propelled phase) and final (or empty) mass (at end of propelled phase) of the rocket as:

$$M_{initial} = M_o = M_u + M_s + M_p$$
$$M_{final} = M_f = M_u + M_s \quad [6]$$

Here M_u is mass of useful load (payload) including the mass of the nose cone, M_s is structural mass of rocket stage and M_p is propellant mass carried on board of the stage.

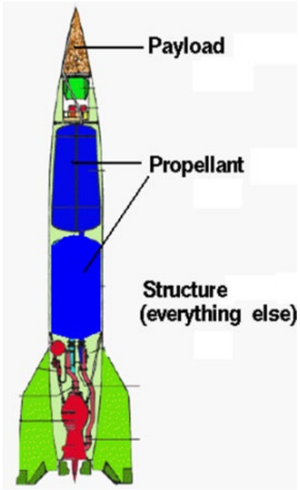


Figure 21: Basic lay-out of single stage rocket

Rocket propellant mass then follows from:

$$M_{propellant} = M_p = M_o - M_f \quad [7]$$

Earlier we found that to attain a high delta-V, the mass ratio as defined in relation [2] needs to be maximized.

Using above relations in combination with the rocket equation, it can easily be found that to attain a high mass ratio (for a given payload mass) and hence maximizing the propellant mass the structural mass should be minimized.

Taking the relation for initial mass and dividing both sides by M_o, we obtain:

$$1 = \frac{M_u}{M_o} + \frac{M_s}{M_o} + \frac{M_p}{M_o} \quad [8]$$

The first ratio is the vehicle payload fraction ($f_u = \lambda$) as defined earlier, the second is the vehicle structural mass fraction (f_s) and the third one is the vehicle propellant mass fraction (f_p). It follows that all three fractions have a value smaller than 1.

A further useful mass fraction is the stage structural coefficient (sometimes also referred to as stage structural efficiency) σ. This is defined as the ratio of stage structural mass divided by the stage propellant mass:

$$\sigma = \frac{M_s}{M_p} \quad [9]$$

A low value of this coefficient indicates that little structural mass is needed to store a certain amount of propellant. Hence this indicates an efficient design. It is actually the technology selected for the stage design that determines the value of the structural coefficient.

Starting from the vehicle mass ratio and applying (some of) the above relations, a relation can be derived[19] that relates vehicle mass ratio directly to payload ratio and structural coefficient. It follows:

$$\Lambda \text{ (or R)} = \frac{1+\sigma}{\lambda+\sigma} = \frac{1 + structural\ coefficient}{payload\ ratio + structural\ coefficient} \qquad [10]$$

This relation shows that given a certain vehicle mass ratio and payload mass fraction, it immediately follows the value of the structural coefficient needed. Otherwise for a given payload mass fraction and structural coefficient, the vehicle mass ratio follows.

For instance, in case we consider the design of a single stage launch vehicle for which are given a vehicle payload mass ratio of 0.1 and a stage structural coefficient of 0.1, it follows for the vehicle mass ratio:

$$\Lambda = \frac{1+\sigma}{\lambda+\sigma} = \frac{1+0.1}{0.1+0.1} = 5.5$$

Existing launchers can be used to determine realistic values for payload mass ratio, structural efficiency, etc. These values can then be used as starting values for new designs.

For instance for the V2 rocket as described in ae1110-II, we find:
- *Total mass at take-off: 12700 kg*
- *Propellant mass: 8800 kg*
- *Vehicle inert/empty mass: 3900 kg*
- *Structure/construction mass: 2900 kg*

Using these data the following mass fractions/ratios can be determined

---

[19] We start by writing the vehicle initial mass as:

$$M_{initial} = M_o = M_u + M_s + M_p$$

Dividing both sides of the above relation by the propellant mass and expressing the payload mass as the product of the payload fraction and the initial mass gives:

$$\frac{M_o}{M_p} = \frac{\lambda M_o}{M_p} + \frac{M_s}{M_p} + \frac{M_p}{M_p} = \frac{\lambda M_o}{M_p} + \sigma + 1$$

Or:

$$\frac{M_o}{M_p} = \frac{1+\sigma}{1-\lambda} \qquad (a)$$

As a next step we consider that propellant mass can be written as the initial mass minus the final mass:

$$M_p = M_o - M_f \rightarrow \frac{M_p}{M_o} = 1 - \frac{M_f}{M_o} = 1 - \frac{1}{\Lambda} \qquad (b)$$

Combining the above relations a and b gives:

$$\frac{1-\lambda}{1+\sigma} = 1 - \frac{1}{\Lambda}$$

Reworking leads to the asked for relation:

$$\frac{1-\lambda}{1+\sigma} - 1 = \frac{-(-\lambda-\sigma)}{1+\sigma} = -\frac{1}{\Lambda} \rightarrow \Lambda = \frac{1+\sigma}{\lambda+\sigma}$$

- *Structural coefficient σ = 2900/8800 = 0.33*
- *Vehicle structural mass fraction f_s = 2900/12700 = 0.23*
- *Vehicle propellant mass fraction f_p= 8800/ 12700 = 0.69*
- *Vehicle payload ratio λ = 1000/12700 = 0.08*

Typical values for the structural coefficient are in range 4% to 18%. These values follow from the inert mass fraction (δ), which is defined here as the ratio of stage inert mass (i.e. structural mass M_s) and the total mass of the stage (sum of propellant mass and structural mass):

$$\delta = \frac{M_s}{M_s + M_p} = \frac{\sigma}{1+\sigma} \qquad [11]$$

Typical values of the inert mass fraction are shown in next figure. From these values, also the value of the structural coefficient can be obtained using above relation.

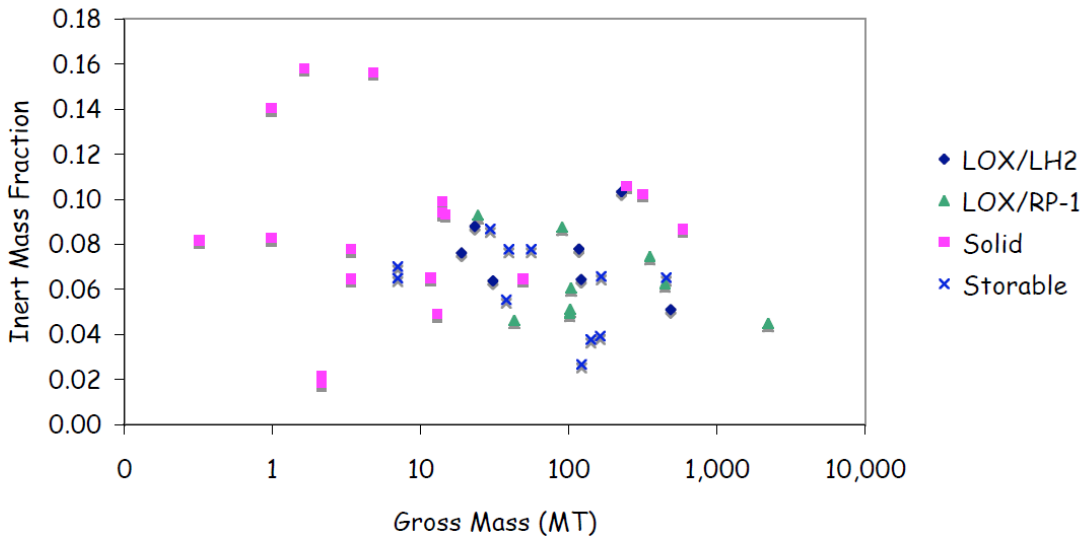


Figure 22: Stage inert mass fraction for various types of rocket stages [Akin]

Clearly it follows that the inert mass fraction and structural coefficient values are within a certain range. This range is because of the different designs. From the figure, we can already see there is an effect of the type of rocket stage considered, including solid and liquid propelled rocket stages, with the latter further subdivided into those using storable liquid propellants and cryogenic (liquid oxygen and liquid hydrogen) and semi-cryogenic propellants (LOX-Rocket Propellant-1 [20]). Additionally, we need to consider that the variation in the inert mass fraction and structural coefficient can also come from the use of different technologies (materials, etc.), different stage sizes, whether the stage is purely propulsive or also fulfils other functions, etc. Some further details follow later in this course.

Next to the stage inert mass fraction (and the structural coefficient), we can also define a stage propellant mass fraction (μ):

$$\mu = \frac{M_p}{M_s + M_p} = \frac{1}{1+\sigma} \qquad [12]$$

Verify that:

$$\delta + \mu = 1 \qquad [13]$$

[20] Some type of kerosene.

Typical values for the propellant mass fraction can be obtained from Figure 22 using relation {12]. Some values are given in Figure 29, which shows propellant mass fractions are in range 0.80 to 0.95 with increasing propellant mass fraction with increasing propellant mass. In other words, larger stages (carrying more propellant) are more efficient than smaller stages, meaning lower stage inert mass fraction and hence a higher structural coefficient.

An increase in vehicle mass ratio, decreasing structural coefficient or inert mass fraction allows for a larger delta-V or a reduction in total vehicle mass, and hence size and cost. The same can be accomplished by increasing the rocket exhaust velocity (or specific impulse). Typical values for rocket exhaust velocity, like for the structural coefficient, can be determined based on existing rockets and is found to largely depend on the propellant selected. It is usually expressed in terms of specific impulse. As indicated earlier, for launchers, specific impulse varies with the pressure altitude at which the rocket is operating, but usually it suffices to distinguish between sea level and vacuum specific impulse. Typical values of specific impulse for a number of rocket propellants can be found later in this work.

5.3 Multi-stage rockets

A rocket stage (also called step in older literature) is a complete propulsion unit together with control equipment, which is discarded completely when all propellants of that stage is consumed (staging). Following stage separation, a smaller rocket (a sub-rocket of the original rocket) results driven by its own propulsive unit (stage). This continues until all stages are spent. Figure 23 shows a schematic of the ARES 1 concept considered by NASA as a replacement for the Space Shuttle. The figure essentially is a two stage rocket[21] consisting of a first stage and an upper stage. First the 1st stage is ignited. After burn-out this stage is discarded off. This leaves a smaller rocket (sub-rocket) that is further accelerated by the upper stage to the final velocity. The upper stage is ignited once the first stage is separated. For a given (stage) structural coefficient and propulsion system performance, staging allows for higher ΔV's to be reached and/or a higher payload mass to be carried in the designated target orbit at identical total vehicle mass (at lift-off).

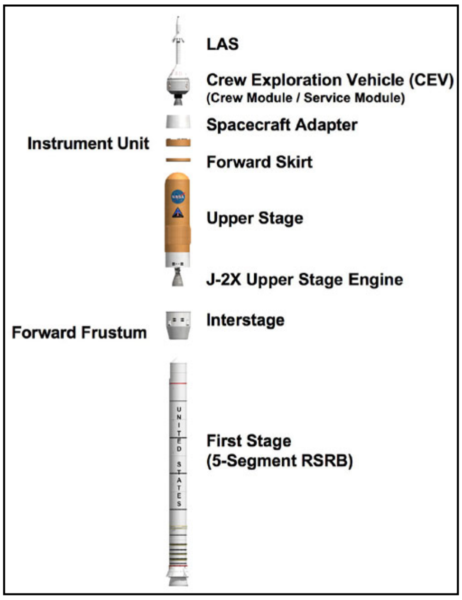


Figure 23: Schematic of ARES 1 launch vehicle showing 3 propulsive stages (courtesy NASA)

Figure 24 shows a typical staged flight of a staged launcher. At various instants in time, no longer useful mass, a stage (and even the fairing), is jettisoned and the remainder of the rocket (a sub-rocket) continues its way to orbit. Notice that here we distinguish between rocket stages and sub-rockets. Also notice that a next stage is not necessarily ignited right after separation of the previous stage. This is done to allow for maximizing the payload mass into orbit. This will be dealt with in more detail in later courses.

[21] Figure shows that also CEV is equipped with a propulsion unit. However, the CEV is considered a payload of the ARES 1 vehicle.

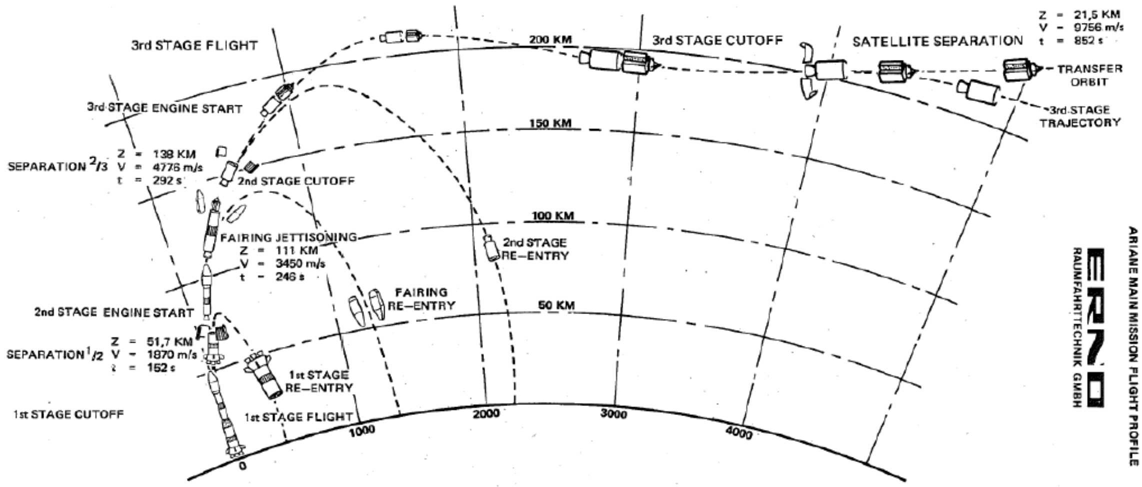


Figure 24: Typical launch vehicle ascent trajectory profile [LVC]

Two types of staging can be distinguished, being serial staging and parallel staging.

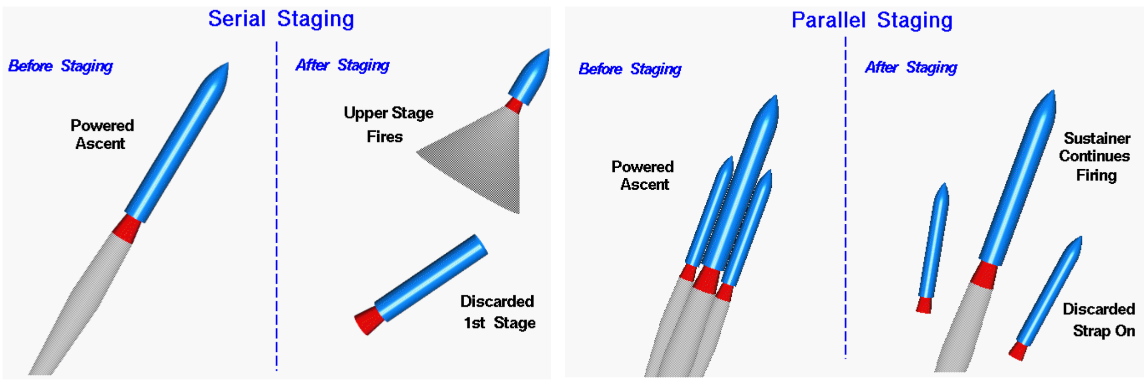


Figure 25: Serial (l) and parallel (r) staging

In case of serial staging, the stages work in series. In case of parallel staging, one or more propulsion units (booster stages) work in overlap with a core stage.

We now define a sub-rocket as a complete rocket vehicle, consisting of an assembly of one or more stages with a payload and an avionics system, (see later in this course) as it occurs during the launch of a satellite. By convention, the nth sub-rocket is defined as rocket section for which the nth stage is bottom stage. Booster stages operating in parallel with a core first stage usually are referred to as zeroth stage.

Now the purpose of staging is to increase the payload ratio. Considering the payload ratio of the original rocket (at lift-off) in some detail, it can be shown that payload ratio of original rocket is product of payload ratios of individual sub-rockets:

$$\lambda_{total} = \prod^{N} \lambda_i \qquad [14]$$

Here N is the total number of sub-rockets with the subscript i referring to the individual sub-rockets.

Consider for instance a two stage rocket:

- *Payload ratio second sub-rocket:* $\lambda_2 = M_U/(M_o)_2$
- *Payload ratio first sub-rocket* $\lambda_1 = (M_U)_1/(M_o)_1$

As $(M_U)_1 = (M_o)_2$ *we find for the overall payload ratio:*

$$\lambda_{total} = M_U/(M_o)_1 = M_U/(M_o)_2 \; x \; (M_o)_2/(M_o)_1 = M_U/(M_o)_2 \; x \; (M_U)_1/(M_o)_1$$

$$\lambda_{total} = \lambda_1 \; x \; \lambda_2$$

Notice that in the above example, the second sub-rocket is considered to be the payload of the first sub-rocket.

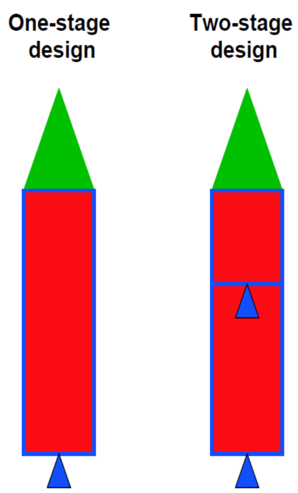
For an N-stage vehicle the total (ideal) velocity increment follows from the sum of the velocity increments achieved by the N sub-rockets:

$$(\Delta v)_{total} = \sum_{n=1}^{N} (V_e)_n \cdot \ln(M_o / M_e)_n$$
$$(\Delta v)_{total} = \sum_{n=1}^{N} I_{sp,n} \cdot g_o \cdot \ln[(1+\sigma_n)/(\lambda_n + \sigma_n)] \qquad [15]$$

Hereafter, we will first provide an example to illustrate that staging does allow for increasing the payload ratio (for identical mass at lift-off and identical mission characteristic velocity), we will assume equal structural efficiency and effective exhaust velocity for both stages. It will be left to the reader to find out for himself to what extend the mission characteristic velocity can be increased by staging in case payload mass is kept constant. Next, two examples will be given of how to determine structural coefficient for an existing rocket. The first example is for a serial staged rocket, whereas the second example is for a parallel staged rocket.

Example: Effect of staging on payload ratio

To illustrate that staging allows for increasing the payload ratio, we determine the payload ratio of a single-stage and a two-stage launcher, see figure.



Given are:

- *Mission velocity requirement:* $\Delta V = 9200$ *m/s*
- *Launch mass:* $M_o = 100$ *Mg (100 ton)*
- *Stage characteristics:*
    - *(Average) effective exhaust velocity of engines:* $V_e = 4400$*m/s*
    - *Stage propellant mass fraction:* $\mu = 0.9$

1) Single-stage design:

First we determine propellant mass using the rocket equation:

$$M_p = \left(1 - e^{-\Delta v / V_e}\right) M_o = 87643 \; kg$$

$$M_s = M_p \left(\frac{1}{\mu} - 1\right) = 9738 \; kg$$

$$M_u = M_o - M_p - M_s = 2619 \; kg$$

$$\lambda = \frac{M_u}{M_o} = \frac{2619 \; kg}{100000 \; kg} = 0.026 \; or \; 2.6\%$$

2) *Two-stage design:*

For the design of the serial staged two-stage rocket, we assume that the total mission delta-V is divided equally over the two stages (sub-rockets) and that both stages have a propellant mass fraction of 0.9. It follows:

For the propellant mass of the first sub-rocket:

$$M_{p,1} = \left(1 - e^{-\Delta v_1 / V_{e,1}}\right) M_{o,1} = 64847\ kg$$

The structural mass of stage 1 is:

$$M_{s,1} = M_{p,1}\left(\frac{1}{\mu_1} - 1\right) = 72015\ kg$$

This leaves for the useful load of the first sub-rocket:

$$M_{u,1} = M_{o,2} = M_{o,1} - M_{p,1} - M_{s,1} = 27948\ kg$$

For the propellant mass of the second sub-rocket follows:

$$M_{p,2} = \left(1 - e^{-\Delta v_2 / V_{e,2}}\right) M_{o,2} = 18123\ kg$$

The structural mass is:

$$M_{s,2} = M_{p,2}\left(\frac{1}{\mu_2} - 1\right) = 2013\ kg$$

The useful load of the second sub-rocket:

$$M_{u,2} = M_{o,2} - M_{p,2} - M_{s,2} = 7811\ kg$$

It now follows for the payload mass ratio of the complete rocket:

$$\lambda = \frac{M_u}{M_o} = \frac{7811\ kg}{100000\ kg} = 0.078\ or\ 7.8\%$$

Using staging, the total payload mass into orbit can be significantly increased. The example presented above shows that for a single stage vehicle to attain a delta-V of 9200 m/s a payload mass ratio of 2.6% results, whereas for a two-stage design this is 7.8%. See here the benefit of staging.

Example calculation of mass fractions of serial-staged rocket
The Russian Dnepr rocket, see figure, is a three stage rocket launcher used to reach Low Earth Orbit. Typical data of the three stages are given in the table below [LVC].

STAGE	1	2	3
Designation	R-36M2-1	R-36M2-2	R-36M2-3
Manufacturer	Yuzhnoye	Yuzhnoye	-
Length (m)	22.3	5.7	1.0
Diameter (m)	3.0	3.0	3.0
Dry mass (t)	15	8.6	2.35
Propellant:			
➢ Type	Liquid	Liquid	Liquid
➢ Fuel	UDMH	UDMH	UDMH
➢ Oxidizer	N_2O_4	N_2O_4	N_2O_4
Propellant mass (kg)			
➢ Fuel	-	-	-
➢ Oxidizer	-	-	-
TOTAL	147	36.7	1.91
Tank pressure (bar)	-	-	-
Total lift-off mass (t)	162	45.3	4.26

To this we should add the payload and the fairing. Data provided suggest a maximum payload mass (including mass of fairing and S/C adapter) into LEO of 3600 kg.

Using the above data, we can determine the following mass fractions for each of the sub-rockets:

For the first sub-rocket, it follows:
Mass at lift off is 162 + 45.3 + 4.26 + 3.6 = 215.2 ton.
Propellant mass is 147 ton (propellant mass carried by first stage)
Empty mass is 68.2 ton
Structural mass is 15 ton
Payload mass is 45.3 + 4.26 + 3.6 = 53.16 ton.

This gives:
- *Payload mass fraction: 53.16/215.2=0.247*
- *Propellant mass fraction: 147/215.2 = 0.683*
- *Structural mass fraction: 15/215.2 = 0.070*

Verify that the sum of the three mass fractions equals 1.

For the second sub-rocket it follows[22]*:*
Mass at start of burn is 45.3 + 4.26 + 3.6 = 53.2 ton (is payload mass of first sub-rocket).
Propellant mass is 36.7 ton
Empty mass is 16.5 ton
Structural mass is 8.6 ton
Payload mass is 4.26 + 3.6 = 7.86 ton.

[22] At some point in flight the fairing is jettisoned. This reduces the payload mass to be carried into orbit slightly. Here this effect is neglected.

This gives:

- *Payload mass fraction: 7.86/53.2=0.148*
- *Propellant mass fraction: 36.7/53.2 = 0.690*
- *Structural mass fraction: 8.6/53.2 = 0.162*

Check whether the sum of the three mass fractions is equal to 1. If not, some error has been made.

For the third sub-rocket it follows:
Mass at start of burn is 4.26 + 3.6 = 7.86 ton (is payload mass of second sub-rocket).
Propellant mass is 1.91 ton
Empty mass is 2.35 + 3.6 = 5.95 ton
Structural mass is 2.35 ton
Payload mass is 3.6 ton.

It follows:

- *Payload mass fraction: 3.6/7.86=0.458*
- *Propellant mass fraction: 1.91/7.86 = 0.243*
- *Structural mass fraction: 2.35/7.86 = 0.299*

Check if sum of the three calculated mass fractions equals 1.

For the payload ratio of the complete rocket follows:

$$\lambda_1 \times \lambda_2 \times \lambda_3 = 0.247 \times 0.148 \times 0.458 = 0.0167$$

This value should be identical to the value that follows by dividing the actual payload mass by the mass at lift off: 3.6/215.2 = 0.0167.

From the data given, we can also determine the structural coefficient (μ) of the three stages:
Structural coefficient 1st stage = 15/147 = 0.102
Structural coefficient 2nd stage = 8.6/36.7 = 0.234
Structural coefficient 3rd stage = 2.35/1.91 = 1.23

The value of the structural coefficient is best (lowest) for the first stage. The relatively high value of the structural coefficient of the third stage is attributed to that most of the vehicle controls and the avionics are located in this stage and contribute heavily to the mass.

In an identical way, we can also determine the propellant mass fraction (μ=1/(1+σ) for the three stages:
Propellant mass fraction 1st stage = 147/162 = 0.907
Propellant mass fraction 2nd stage = 36.7/45.3 = 0.810
Propellant mass fraction 3rd stage = 1.91/4.26 = 0.448

From the above data, we find that the first stage has the highest propellant mass fraction.

Exercise: Obtaining data on mass fractions
Use Launch Vehicle Catalogue (LVC) to obtain typical values for stage propellant mass fraction and structural coefficient of rocket stages and compare the values with the values earlier reported in Figure 22, Figure 29, Table 5 and Table 6. See also the problems included later in this work.

5.4 Optimum staging

In case a certain delta-V is to be delivered by a multiple stage vehicle, the question is how much delta-V each stage should deliver or what is the optimum delta-V distribution over the various sub-rockets to obtain maximum payload ratio. It turns out that the optimum distribution depends on the structural coefficient of the individual stages and the specific impulse of the propulsive units used.

For a two-stage vehicle, the payload mass fraction l of the rocket with respect to a given mission ΔV can be obtained from the following equation (verify):

$$\lambda = \lambda_1 \cdot \lambda_2 = \left(\frac{1}{\mu_1}\left(\frac{1}{e^{\Delta v_1/V_{e,1}}} - 1\right) + 1\right) \cdot \left(\frac{1}{\mu_2}\left(\frac{1}{e^{\Delta v_2/V_{e,2}}} - 1\right) + 1\right) \qquad [16]$$

Problems:

1) Demonstrate that the payload ratio (λ) of a sub-rocket can be related to the stage propellant mass fraction, the mission characteristic velocity and the exhaust velocity according to:

$$\lambda = \frac{1}{\mu}\left(\frac{1}{e^{\Delta v/w}} - 1\right) + 1 \qquad [17]$$

2) Verify also:

$$\lambda = \frac{1}{e^{\Delta v/w}}(1 + \sigma) - \sigma \qquad [18]$$

Next figure shows effect of stage propellant mass ratio and propulsion unit effective exhaust velocity on optimum distribution of delta V over the stages for a two-stage to orbit rocket.

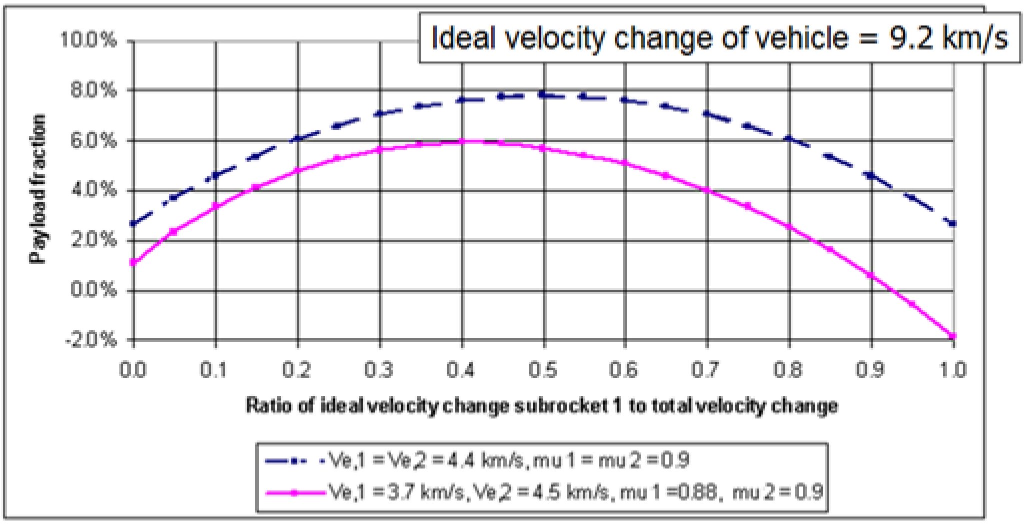


Figure 26: Optimum staging of two stage rocket for different propellant mass fractions and (effective) exhaust velocity

Above figure shows that for a rocket consisting of two stages with each same specific impulse and propellant mass fraction total performance is optimum (maximum λ) in case $\Delta V_1 = \Delta V_2$. In case the effective exhaust velocity and the stage propellant mass fractions differ, clearly the optimum delta-V ratio differs from 0.5.

Question: How would above equation change in case we have a 3-stage rocket?

In the foregoing, we have used the rocket equation, which essentially holds in gravity free space and in vacuum. In reality, the theoretical optimum is influenced by the ascent trajectory due to:

- Gravity and drag losses
- Change in engine performance (V_e depends on ambient pressure)

For further study: Consider how you would aim to find the optimum in case you have a three stage rocket and how this optimum is affected in case the stages have different structural efficiency, specific impulse and so on.

5.5 Limiting gravity loss and drag

Equation of motion in presence of gravity field (vertical flight) and drag is given by:

$$Resulting\ Force = Thrust - Weight - Drag = M\ x\ a \qquad [19]$$

To limit gravity loss, we know from an earlier course (AE1110-II) that the higher the thrust, the shorter the burn time and the lower the gravity loss. So this would mean we need to opt for high thrust. However, thrust level is limited by maximum allowed acceleration level. For instance, the Space Shuttle has a thrust to weight (T/W) ratio of 1.5 at lift-off. The T/W ratio of course increases during flight as the rocket gets lighter (constant thrust). At a T/W ratio of about 3, the controls kick in to limit the acceleration loads the astronauts have to endure. Next table shows initial thrust to weight (T/W) ratio for a range of vehicles.

Table 4: Lift-off thrust to weight ratio and length-to-diameter (L/D) of some rockets [Francis]

Vehicle	Vehicle Mass (kg)	Thrust to Weight Ratio	L/D
Energia	2,524,600	1.48	7.54
Ariane V	737,000	1.73	10.00
Proton D-1	712,460	1.27	7.97
Ariane IV (44L)	470,000	1.17	15.37
Soyuz	297,400	1.38	11.50
Delta II (7925)	230,000	1.56	15.88
Titan II	150,530	1.28	11.61
Kosmos	107,500	1.40	10.96
Rokot	97,170	1.63	8.80
Conestoga	87,407	1.99	12.67
Taurus	73,030	1.80	11.25
Athena	64,820	2.02	6.25
Pegasus XL	24,000	2.07	13.54
Shavit	23,400	1.80	11.54

Clearly can be seen that at lift-off all rockets have a T/W ratio in excess of 1. For sub-rockets working at higher altitudes, the thrust can be much less than the weight of the SLV. For instance, the Ariane 5 2^{nd} sub-rocket consists of 2^{nd} stage, VEB + payload. Maximum total mass (18 ton payload) is 30.3 ton (from [LVC]. This represents a weight at sea level of roughly 300 kN. Motor thrust is 29.1 kN, which indicates a T/W ratio of just 0.1.

When selecting a thrust level, it is important to realize that, during flight, in case of a constant thrust, the T/W ratio increases because of propellant being expelled and hence the rocket becomes lighter. This may lead to very high accelerations during some stages of the ascent flight.

Example: Acceleration level
Consider a rocket of total mass 10 tons of which 9 tons are propellant. Maximum acceleration allowed during flight is 6g. In addition, thrust of the rocket is constant.

Option 1: Thrust is identical to thrust at end of flight (maximum acceleration occurs at end of flight). This gives a thrust level at end of flight of 1000 kg x 6 x 9.81 m/s^2 = 58,860 N. Comparison with the weight of the rocket at lift-off shows that the thrust is too low to ensure lift-off of the rocket (rocket weight initially is 98100 N)

Option 2: Thrust is constant and is set equal to realize 1.2g acceleration at lift-off. This gives a thrust of 117,720 N. However, at end of flight this gives an acceleration of 12g.

Note that in the above, we have neglected the effect of weight on the acceleration. This is important when considering flight where there is a gravity component in the direction of flight, like in vertical flight for launch from Earth surface into space. For space applications, most of the time we aim to have the thrust force in a direction perpendicular to gravity and hence it has no effect.

To limit drag loss, we need to limit the drag experienced by the SLV. This is done by designing slender vehicles with a high length-to-diameter ratio as in that case a low value of the drag coefficient can be attained; see later in this course for more detailed info. From Table 4, we learn that typical L/D values are in range 6 to 16. See later in this course for more on vehicle aerodynamics.

6 Rocket design details (some detailed analysis)

In the foregoing, we have discussed launch vehicle design using highly generic methods that do not include a lot of detail. Hence the results might also not be too accurate. To allow for more accuracy, we need to increase the level of detail. Hereafter we will go into more detail concerning rocket configuration, aerodynamics, stability, avionics, separation etc. to find out how these elements determine the design of the launch vehicle. To this end, we will first perform a breakdown of a launch vehicle in its main constituents/elements. In the present course, we will focus on expendable rockets only. Later in the study, the analysis may be extended to also include air-breathing SLV, winged vehicles and so on. This though is considered beyond the scope of this course.

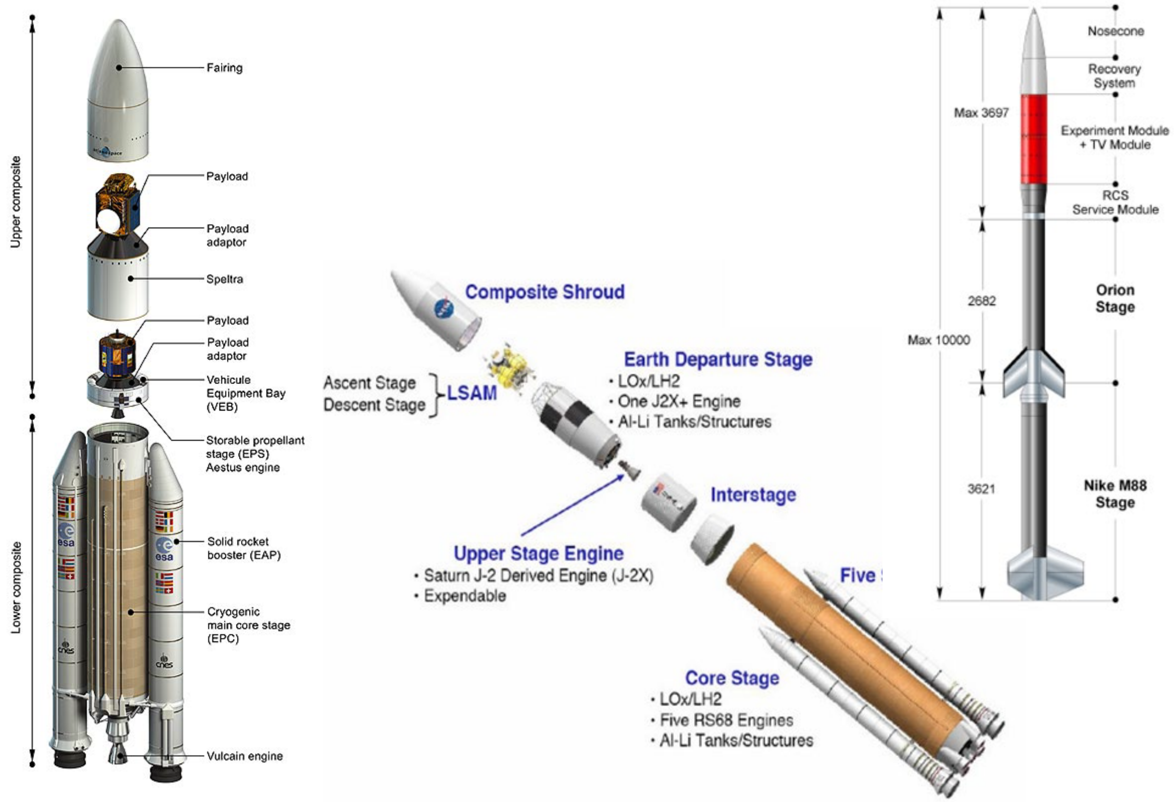


Figure 27: Expendable Launch Vehicle (Ariane 5 left, Ares 1 middle) and MiniTexus sounding rocket right) breakdown (courtesy ESA, NASA)

From Figure 27 we learn that the main building blocks of a launch vehicle or sounding rocket are one or more propulsive stages or rocket stages. A rocket stage is a separable element of a rocket containing its own propulsion unit. By staging, no longer useful mass is discarded, thereby allowing the remainder of the rocket to lift an increased payload mass in to orbit and/or to reduce the initial mass of the rocket at lift-off. Besides the various rocket stages, a rocket also consists of interstages connecting the rocket stages, a fairing, an avionics system, etc.

6.1 Rocket stages

Different types of rocket stages exist based on the type of propulsion system used. From historical data, we find that there are essentially two types of rocket stages, being stages that use liquid propellants and those that use solid propellants. Figure 28 shows a schematic of 3 rocket stages, being the Ariane 5ME liquid rocket upper stage, Saturn V S1C liquid rocket stage and Space Shuttle Solid Rocket Booster.

The Ariane 5 ME and Saturn V S1C are both of an integral tank design, where part of the tank wall also act as part of the stage structure. This allows for a reduced mass design; see later entry on stage structures. For the Ariane 5 ME upper stage only the tank wall of the larger (liquid hydrogen, LH2) tank is part of the stage structure. The smaller liquid oxygen (LOX) tank is not part of the stage structure (does not carry launch loads) and is partly installed in the hydrogen tank to ensure a reduced height of the stage and hence a low stage mass. For the Saturn V S1C stage the cylindrical part of both oxidizer and fuel tank form part of the stage structure. For both stages also skirt structures are present that allow for mating with other stages, fairings (payload and aft), payload or that allow for attaching aerodynamic fins. The Space Shuttle SRB, shown in the figure essentially is SRM + nose assembly, forward/aft skirt with hold down points, separation system, TVC systems, avionics, parachutes (in case SRB is of a recoverable design), and electrical system.

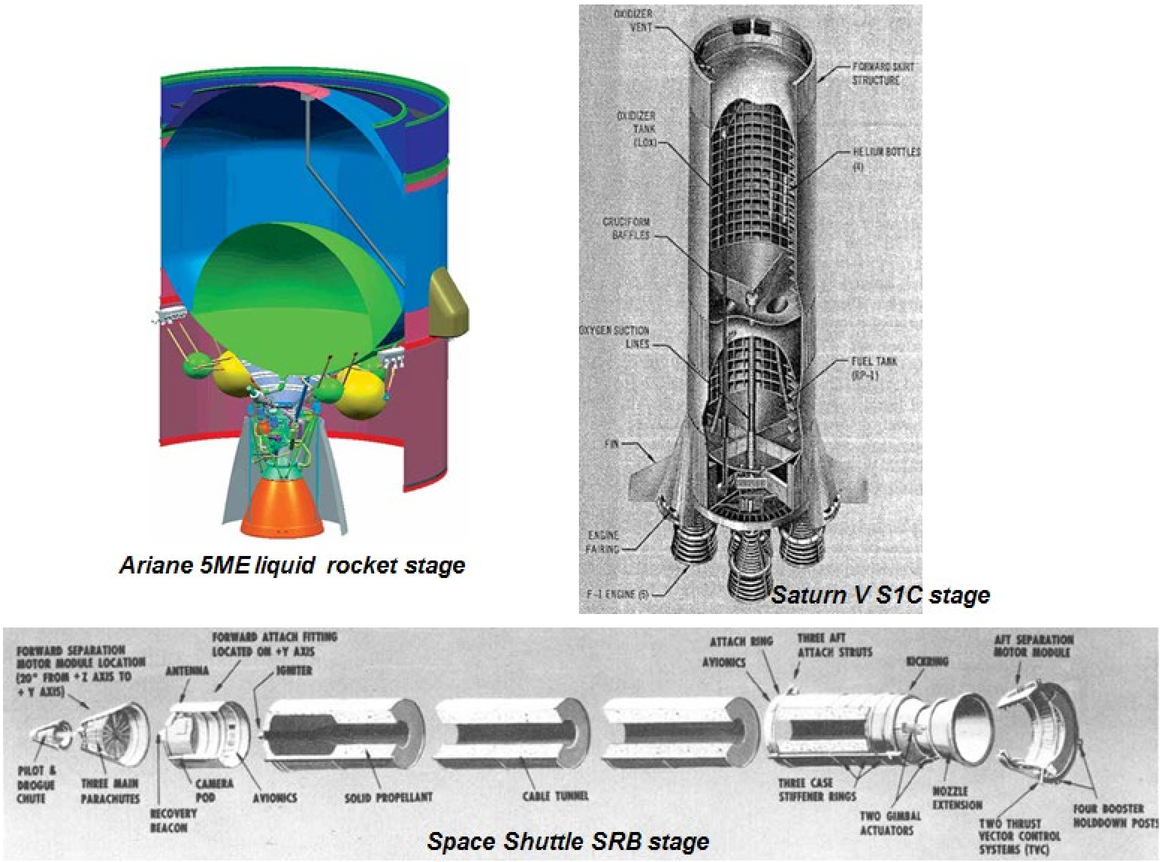


Figure 28: Schematic of Ariane 5 ECB upper stage (left upper figure), Saturn V S1C first stage (right upper figure) and Space Shuttle SRB stage (courtesy ESA, NASA)

Summarizing, the main stage components are:

- Propulsion unit
- Front and aft skirt for mating with other stages, payload fairing, payload adapter, aft fairing
- Reaction control system (mostly only on upper stages)
- Avionics section
- TVC and ignition control and energy supply system
- Recovery system
- Separation system
- Safety system

Important for the design of a rocket stage is the thrust it should deliver. Determining the thrust level is usually done in a launch vehicle study wherein different values are investigated to find the value that leads to the best result. This is generally done while working closely together with experts that can model the ascent (and descent) of a rocket. Still some general rules Some general design rules have been discussed in the section on limiting gravity loss and drag. Some simple relationships that allow for a first estimation (a starter) can be found in appendix E.

Also important for the design of a rocket stage is the stage propellant mass fraction (or the stage structural coefficient) as these affect the overall payload ratio of the rocket. Table 5 and Table 6 show typical propellant mass fractions for a range of liquid and solid rocket stages. Propellant mass fraction here is taken as propellant load carried on board of a stage divided by the stage total structural mass. For a liquid rocket stage the structural mass includes front and aft skirts, tanks, engine, piping and other structural elements as well as some non-structural elements, like cables, batteries (if present), etc.. For solid rockets, this does include, front and aft skirt, separation systems, and nose cone and recovery system (if present). Typical values for liquid propellant stages are in range 0.768 to 0.943. Lowest values are usually found for the upper stage which in general also carries most of the avionics equipment. It is not clear from the data whether the propellants carried on board have an effect on propellant mass fraction. In general, it is expected though that cryogenic stages carry more insulation to keep the propellants cool (and hence liquid). For solid rocket stages, we find a propellant mass fraction in range 0.705 to 0.928 (structural coefficient in range 0.08 to 0.17). For the Shuttle IUS stage the value is much higher as this stage also includes the avionics system and the equipment needed to host the payload.

Table 5: Propellant mass fraction of specific liquid propellant rocket stages [Francis]

Vehicle	Vehicle Mass (kg)	Propellant Mass Fractions (ζ)			
		Stg. 1	Stg. 2	Stg. 3	Stg. 4
Energia	2,524,600	0.901	0.906		
Ariane V	737,000	0.874	0.912	0.768	
Proton D-1	712,460	0.931	0.930	0.918	0.858
Ariane IV (44L)	470,000	0.897	0.928	0.902	0.872
Soyuz	297,400	0.917	0.936	0.906	
Delta II (7925)	230,000	0.896	0.942	0.883	0.883
Titan II	150,530	0.943	0.917		
Kosmos	107,500	0.939	0.931		
Rokot	97,170	0.926	0.878		
Conestoga	87,407	0.882	0.877	0.877	0.892
Taurus	73,030	0.913	0.867	0.898	0.794
Athena	64,820	0.921	0.895		
Pegasus XL	24,000	0.839	0.904	0.794	
Shavit	23,400	0.892	0.839	0.917	

Table 6: Characteristic mass data of specific solid rocket stages

Launcher	Stage designation	Dry mass	Propellant mass	Total mass	Propellant mass fraction	Structural coefficient / efficiency
		[ton]	[ton]	[ton]	[-]	[-]
Vega	P80	7.8	88	95.8	0.919	0.089
Vega	Zefiro 23	1.85	23.9	25.75	0.928	0.077
Vega	Zefiro 9	0.85	10.1	10.95	0.922	0.084
Ariane 5	Ariane 5 SRB	39	238	277	0.859	0.164
Space Shuttle	Shuttle SRB	86976	499000	590000	0.846	0.174
PAM -D	PAM-D	208	2011	2270	0.886	0.103
Shuttle	IUS-1	1160	9707	10867	0.893	0.120
Shuttle	IUS-2	1140	2722	3862	0.705	0.419

* Space Shuttle SRB, PAM-D, Shuttle IUS values are all in [kg].

Figure 29 shows a plot of propellant mass fractions for liquid rocket stages. The plot shows that propellant mass fraction increases with increasing propellant load. This shows it is favorable to design large rockets (able to carry heavy payloads). An interesting question is whether the same behavior can be noticed for solid rocket stages? This question we leave for the reader to investigate for him/herself. An equally interesting question, which we also leave for the reader to investigate, is whether solid rocket stages tend to be lighter or heavier than liquid rocket stages? A first answer may be found in appendix D and/or E.

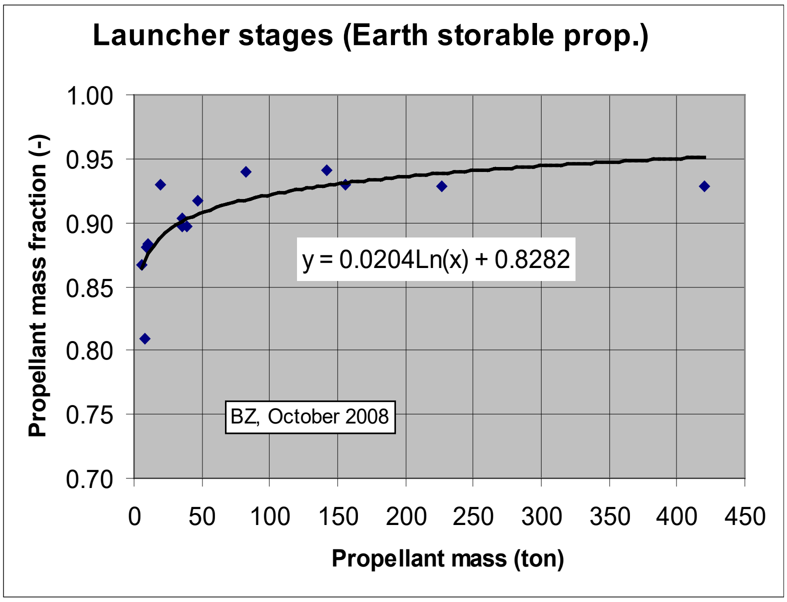


Figure 29: Liquid rocket stage propellant mass fraction in relation to propellant mass

6.2 Propulsion units

The function of the propulsion unit of a rocket stage (or vehicle) is to produce thrust. When producing thrust, it should do so at the least possible expense. Its design starts with some propulsion unit (or system) requirements. The most important one is the thrust to be attained, and the thrust duration or the propellant mass to be carried onboard. Next to that many other requirements flow down from the vehicle (or stage) level to the propulsion level. These requirements may relate to mass, size, reliability, cost, etc. Below, liquid and solid propellant propulsion units, i.e. the units that drive liquid and solid propellant rocket stages, are described in some detail so that the reader develops an understanding of the various options and is able to reason/quantify how these options affect the design.

6.2.1 Liquid propulsion units

A liquid rocket unit consists of one or more rocket engines that generate the necessary thrust, propellant tanks that store the propellants, and a propellant feed system that feeds the propellants to the engine(s). Figure 30 shows a schematic of Ariane 5 ECB upper stage propulsion unit (repeated from Figure 28) next to a schematic of the Automated Transfer Vehicle (ATV) propulsion unit.

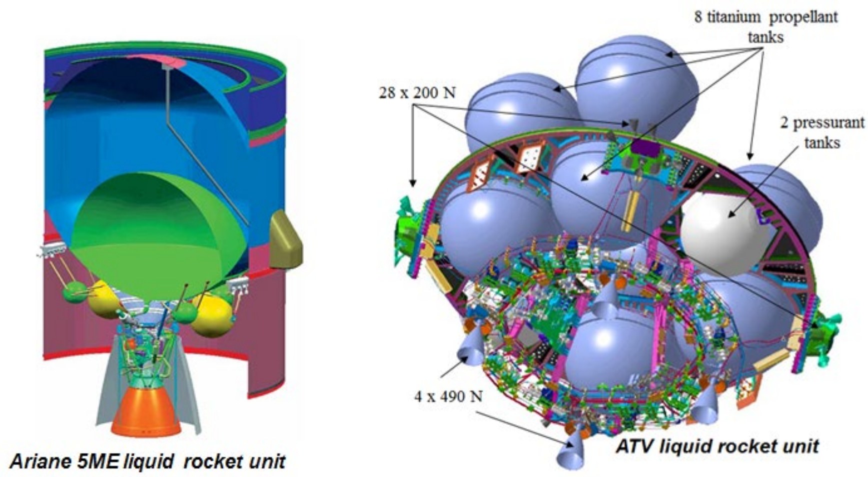


Figure 30: Schematic of Ariane 5ME liquid propellant rocket unit (left) and ATV liquid rocket unit (courtesy ESA)

The two designs are both for upper stages, be it that the Ariane 5ME liquid rocket unit is of an integral tank design, whereas for the ATV the tanks only have to carry the internal pressure loads. The Ariane 5 ME liquid unit is based on the use of two tanks fit to store a total of 24 ton of LOX/LH_2 propellant, a 155 kN Snecma Moteurs Vinci engine and two turbo-pumps (not really visible in figure) that feed the propellant to the engine. The largest tank carries the liquid hydrogen and the smaller one the liquid oxygen. Both propellants are stored at cryogenic temperatures to ensure they remain liquid. The ATV propulsion system comprises 4 x 490 N main engines and 28 x 200 N attitude control thrusters. Eight titanium propellant tanks (build by MT Aerospace) hold up to seven tonnes of Earth-storable (non-cryogenic) liquid propellants (MMH – Monomethylhydrazine, and N_2O_4 - nitrogen tetroxide), used for attitude and orbit control of ATV itself as well as of the ISS. The propellants are pressurised by helium stored in two high-pressure wound carbon fibre tanks.

Rocket propellants

The performance of liquid (and solid) rocket units largely depends on the propellants selected. The next table provides data obtained from selected liquid rockets taken from http://www.braeunig.us/space/systems.htm. Some further data are given in appendix C.

Table 7: Selected rocket and their propellants

Rocket	Stage	Engines	Propellant	Specific Impulse
Atlas/Centaur (1962)	0	Rocketdyne YLR89-NA7 (x2)	LOX/RP-1	259s sl / 292s vac
	1	Rocketdyne YLR105-NA7	LOX/RP-1	220s sl / 309s vac
	2	P&W RL-10A-3-3 (x2)	LOX/LH2	444s vacuum
Titan II (1964)	1	Aerojet LR-87-AJ-5 (x2)	NTO/Aerozine 50	259s sl / 285s vac
	2	Aerojet LR-91-AJ-5	NTO/Aerozine 50	312s vacuum
Saturn V (1967)	1	Rocketdyne F-1 (x5)	LOX/RP-1	265s sl / 304s vac
	2	Rocketdyne J-2 (x5)	LOX/LH2	424s vacuum
	3	Rocketdyne J-2	LOX/LH2	424s vacuum
Space Shuttle (1981)	1	Rocketdyne SSME (x3)	LOX/LH2	363s sl / 453s vac
	OMS	Aerojet OMS (x2)	NTO/MMH	313s vacuum
	RCS	Kaiser Marquardt R-40 & R-1E	NTO/MMH	280s vacuum
Delta II (1989)	1	Rocketdyne RS-27	LOX/RP-1	264s sl / 295s vac
	2	Aerojet AJ10-118K	NTO/Aerozine 50	320s vacuum

The table shows that specific impulse varies from about 220-360 s at sea level and 285-444 s in vacuum. The higher performance in vacuum is associated with the reduced atmospheric pressure. Note that all propellants used consist of a separate fuel and oxidizer. Such propellants are referred to as bipropellants. Differences for identical propellants are a.o. associated with different engine designs.

Table 8 shows some more general data next to typical propellant densities that can be accomplished; the higher the mass density, the smaller the tank volume needed (for the same propellant mass).

Table 8: Sea level specific impulse and mean density of various propellant combinations [SSE]

Fuel	Oxidizer	Ideal specific impulse (s)	Mean density kg/m^3
H_2 (hydrogen)	O_2 (oxygen)	390	280
	F_2 (fluorine)	410	460
Kerosine	O_2	301	1020
	F_2	320	1230
	RFNA (red fuming nitric acid)	268	1355
	N_2O_4 (nitrogen tetroxide)	276	1260
	H_2O_2 (hydrogen peroxide)	278	1362

Note: All quoted values are for $p_c = 7$ MPa with an ideal expansion to $p_e = 0.1$ MPa.

Exercise:
Use [LVC] to identify typical liquid propellants in use as well as the attainable specific impulse range for these propellants both in vacuum and at sea level.

Rocket engines
The most important element of each propulsion unit, next to the propellant itself, is the engine system that provides the necessary thrust. In most units, the engine system consists of one or more engines/thrust chambers see for instance Figure 31. Main elements of a rocket engine are the combustion chamber where the combustion of the chemical propellants occur and a convergent-divergent (Con-Di) nozzle wherein the flow is accelerated from a low subsonic velocity to a high supersonic velocity.

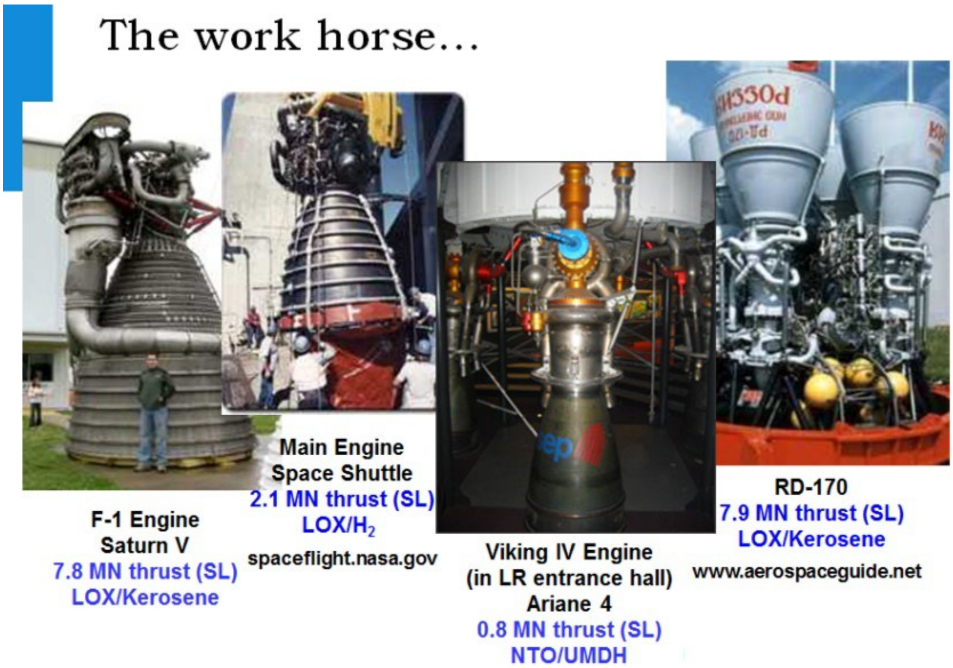


Figure 31: Typical large rocket engines for first stage propulsion units

Engine thrust is determined once the propellant is selected and the specific impulse known. For this we need only information on allowable T/W or acceleration levels and the number of engines. Maximum thrust levels generally follow from the requirements. "Actual" thrust levels are generally taken from trajectory analysis studies with as aim to optimize payload mass. This process can be highly iterative as for the trajectory analysis some information on the vehicle needs to be known.

Once the thrust is set, we can determine the mass flow rate using the following relation, see also spacecraft design:

$$F = m\,w = m\,I_{sp}\,g_o \qquad [20]$$

From mass flow rate and propellant mass than the thrust time can be determined. For a constant thrust (and hence constant mass flow rate) follows:

$$t = M_p/m \qquad [21]$$

Total impulse (at constant thrust) can be determined using:

$$Ft = m\,t\,w = M_p\,w \qquad [22]$$

When considering existing engines, a distinction must be made between sea level and vacuum thrust and total impulse. Like with the specific impulse also the thrust and total impulse at sea level and the thrust and total impulse in vacuum tend to differ.

In the early stages of design, engine mass and size are mostly determined using simple models based on historical data. In appendix G typical such models are given. These models are based on the assumption that to generate high thrust the rocket engine must be large and hence heavy.

Propellant tankage

Once the propellant has been selected and the value of specific impulse determined, the propellant mass can be determined using known values of initial mass and mission characteristic velocity (delta-V). The latter usually follow from the requirements. Typically a margin of 1-3% is added to the calculated propellant mass to make up for propellant that cannot be expelled from the tanks.

Once propellant mass is determined, we can determine the volume required to store the propellants. For this it is important to realize that oxidizer and fuel are to be burned in a certain ratio. Too much fuel or oxidizer will cause a significant reduction in specific impulse and may even lead to no combustion at all. Mixture ratio of the oxidizer and fuel, usually denoted as O/F, is usually expressed in terms of mass (flow rate) with oxidizer mass serving as the numerator and fuel mass serving as the denominator. For instance, in case we burn every second 6 kg of oxygen with 1 kg of hydrogen, this gives a (mass) mixture ratio of 6/1 = 6. The next table taken from [Whitehead] gives representative mixture ratios (both in terms of mass and volume[23]) for various propellant combinations. From this table we find that the optimum mixture ratio of oxygen with RP-1 (Rocket Propellant 1, i.e. a certain type of kerosene) is 2.77, meaning that for every kg of RP-1, we need 2.77 kg of oxygen to be combusted. Hence it also means that in case we have 1000 kg of O_2 - RP-1 propellant, fuel mass = 1/3.77 x 1000 kg = 265 kg and oxygen mass is 1000 – 265 = 735 kg.

Table 9: Characteristics of candidate propellants [Whitehead]

propellant combination	O/F ratios mass	O/F ratios vol	specific gravity ox	specific gravity fuel	specific gravity bulk
O_2—H_2	6.00	0.37	1.14	.071	.363
O_2—CH_4	3.45	1.25	1.14	.415	.821
O_2—RP-1	2.77	1.96	1.14	.810	1.03
98% H_2O_2—JP-5	7.00	4.01	1.43	.820	1.31

For data on a number of other propellants, see appendix C, table 2. Note that not all propellants are used often. From the above table, mostly the first and the third are used often. The second is getting a lot of attention lately due to SpaceX indicating that Methane (CH4) can be easily produced on Mars, whereas this is almost impossible for kerosene or RP-1 (also no stock of kerosene has been detected on Mars.

More general, it follows for the fuel and oxidizer mass in relation to propellant mass and mass mixture ratio (O/F):

$$M_{fuel} = \frac{1}{1+O/F} \cdot M_{propellant} \qquad [23]$$

And:

$$M_{ox} = \frac{O/F}{1+O/F} \cdot M_{propellant} \qquad [24]$$

[23] A volume mixture ratio of 0.36 means that for every litre of fuel we need 0.36 litre of oxidizer.

Once oxidizer and fuel mass are known, we can determine the fuel and oxidizer volume using information on the mass density of the respective fuel and oxidizer. Typical values are also included in the foregoing table. For instance the density of oxygen when stored in liquid state is roughly 1140 kg/m^3 and of hydrogen about 70 kg/m^3, depending on the storage temperature.

Example:
Consider a LOX- LH2 propellant mass of 1000 kg (including margin). In case this is burned at an O/F ratio of 4, this means we have 800 kg of LOX and 200 kg of LH_2. Using the above given mass densities, it follows a LOX volume of 0.70 m^3 and an LH_2 volume of 2.63 m^3. Total propellant volume hence is 3.33 m^3, which leads to a bulk density of 1000/3.33 = 300 kg/m^3.

The in the example calculated value for the mean density compares reasonably well with the value for the bulk density in Table 9. The difference is because of a difference in O/F ratio. Additional data on mean density can be obtained from Table 8.

Tank volume usually is taken about 10% larger than the calculated propellant volume. This is to compensate for differences in expansion of propellant and tank, which may lead to excessive stresses in a tank. This excess volume is referred to as ullage volume.

The relevance of a high mass density is that tank volume can be kept small and hence tank mass low. Also frontal area can be limited and hence the drag on the vehicle. On the other hand, however, high density may not lead to high performance in terms of propellant mass, compare e.g. the combination hydrogen-oxygen versus kerosene-oxygen, and hence may require more propellant mass and hence larger and heavier tanks (more on tank mass and size in a later section). Hence a careful design should be made weighing the different options. As an example, we refer to the Ariane 5 rocket and the Space Shuttle that both use cryogenic propellants for the core stage, meaning high specific impulse, but low mass density, but also uses high density and less performing solid rocket boosters to allow reducing drag during the first stages of the ascent.

What propellant we select is of influence on the specific impulse and on the mean propellant density and hence the tankage volume. However, when selecting some propellant also other aspects should be considered, like (see also annex D):

- Storability of liquids (liquids are to be stored in a closed container prior to launch
    - *Storable* propellants include a.o. kerosene (RP-1), hydrazine, nitrogen tetroxide (N_2O_4) and, mono-methyl-hydrazine (MMH)
    - *Cryogenic* propellants include Liquid Oxygen or LOX (90K) and Liquid Hydrogen or LH_2 (14K).
- Ignitability
    - Liquid propellants that require an ignition stimulus
    - Hypergolic (self-ignition) propellants: no igniter needed -- propellants react on contact in engine.
- Safety related parameters (toxicity, fire hazardousness, explosiveness, etc.)

Once tank volume is known, we can decide upon the number of tanks and the placement in the rocket stage to limit e.g. CG travel. Some typical tank configurations are shown in Figure 32.

Once a tank configuration is selected, also tank mass can be determined for this configuration. Typical tank mass estimation relationships can be found in appendix E. These relationships all indicate that tank mass is roughly proportional to propellant mass. A distinction is made after type of propellant. Not included is effect of tank pressure, tank material, tank shape, etc. How the latter parameters affect tank mass is treated in some detail in a later section (see structures section).

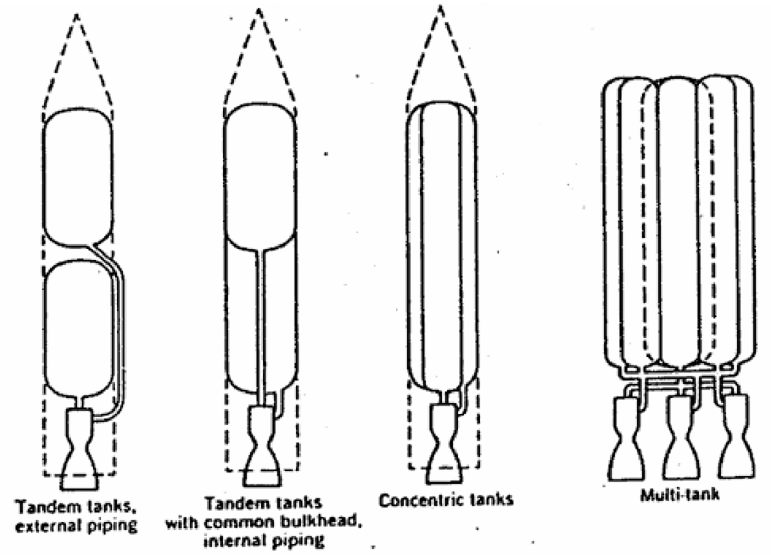


Figure 32: Tank configurations [Sutton]

Propellant feed system

To feed the propellants from the respective storage tanks to the engine(s), a propellant feed system is needed. Two types of propellant feed system exist, being pump-fed systems and gas pressure (or pressurant)-fed systems.

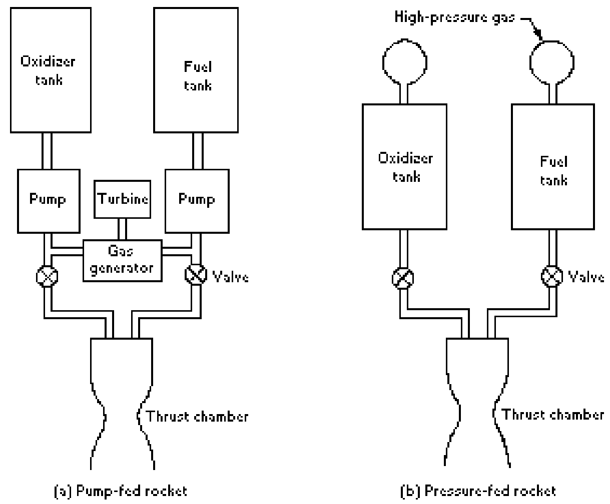
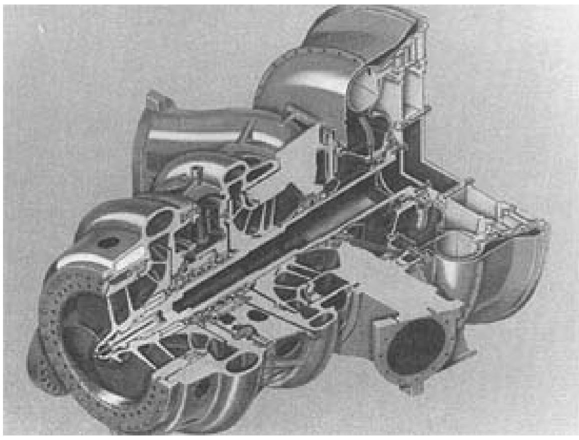


Figure 33: Schematic of liquid propellant feed systems (courtesy NASA)

Pump fed systems use oxidizer and fuel pumps to feed the propellants from the tanks to the engine. Most pumps are driven by a gas turbine (like a compressor in an air-breathing jet engine is driven by a turbine). Next figure shows a F-1 engine turbo-pump, consisting of a fuel- and an oxidizer pump both driven by one and the same turbine. Total input power is 41 MW, which allows for pumping 975 l/s of RP-1 and 1565 l/s of liquid oxygen from the low pressure tanks to a high pressure sufficient for the fluids to overcome the high pressure in the combustion chamber. It is reported that the turbo-pump absorbed more design effort and time for fabrication than any other component of the engine.

Figure 34: Mark 10 turbo-pump for F1 engine (courtesy NASA)



Gas pressure-fed systems use a high pressure gas to force the propellant to flow from the tank to the tanks. Essentially two types of gas pressure-fed systems are available: regulated and blow down.

In pressure regulated systems, see Figure 35, a gas (i.e. the pressurant) is stored separately from the propellant under high pressure. A pressure regulator regulates the flow of pressurant into the propellant tank, thereby maintaining a constant, but much lower pressure. Propellant tank pressure is typically 15-25 bar. Pressurant tank pressure is in the range 150-300 bar.

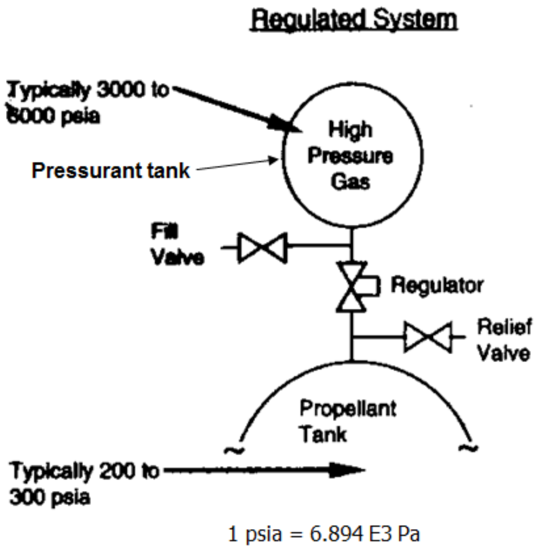


Figure 35: Regulated pressurant-fed feed system

In blow down systems, the pressurant (typically an inert gas, like helium or nitrogen, as to prevent any reaction from the pressurant with the propellant) is stored in the propellant tank itself. This of course adds to the required tank volume and may easily require a doubling of the tank volume as opposed to the regulated case. Typical (initial) tank pressure is 15-25 bar. As during operation the tank empties, the available volume for the pressurant increases and hence tank pressure decreases. The feed pressure in the blow down case must remain large enough to allow good operation to the thrust generator (thruster). Proper operation of the thruster is generally ensured provided that the combustion chamber pressure is in excess of about 2-5 bar.

For rocket applications wherein the total impulse delivered by the stage is in excess of 20 MNs, pump fed systems are used. This is mainly, because pumps allow for low tank pressures (typically 1-3 bar) and hence light weight tanks. For gas pressure-fed systems tanks need to be designed for much higher pressures of up to 20-30 bar, which adds of course mass. Another reason for using pumps is that much higher pressures can be reached in the engine of up to about 200 bar as compared to 20-30 bar for regulated systems.

Below 10-15 MNs, gas pressure-fed systems and more in particular regulated feed systems are used. Blow down systems are almost exclusively used on spacecraft only. The reason for using gas pressure fed feed systems at these low total impulse levels is that propellant mass is relatively low and hence also tank mass. The increase in tank mass in that case is limited as compared to when using pumps. As pumps are high cost items, bring a lot of vibrations and also reduce system reliability, for low total impulse missions pressurant-fed systems are preferred.

For total impulse levels in between 10-15 MNs and 20 MNs a careful trade needs to be made, thereby taking into account not only system mass, but also system reliability, vibration levels and cost.

For turbo-pump systems, pump mass is already included in engine mass, see earlier. For gas pressure-fed systems we do need to take into account the mass of the pressurant as well as the mass of the high pressure gas tank (regulated system) or the additional mass of the enlarged propellant tank (blow down system).

Example:
In this example, we will consider a single stage rocket vehicle that is to provide for a delta-V of 3 km/s. Initial mass of the rocket is limited to 100 ton and initial T/W ratio is 1.45. For this vehicle, we will determine estimates of propellant mass, propellant volume, tank volume, rocket thrust and we will select the type of feed system needed.

For this example, we will select hydrogen-oxygen as the propellant combination at a mass ratio of 6. We furthermore, assume that this allows for providing a sea level specific impulse of 390 s. Using the rocket equation, it follows a mass ratio of 2.19, which gives an empty mass of 45.7 ton and a propellant mass of 54.3 ton.

Using the given O/F ratio it follows a hydrogen mass of 7.8 ton and an oxygen mass of 46.5 ton. In case we store both propellants in a liquid state (under cryogenic conditions), it follows a hydrogen volume of 7.8 ton/71 kg/m^3 = 109.9 m^3 and an oxygen volume of 46.5 ton/ 1140 kg/m^3 = 40.8 m^3. Note that for now we have neglected taking a propellant margin for this mission. Also note that the decision to store both propellants under cryogenic conditions requires a storage temperature below about 20K for hydrogen and 50K for oxygen.

Sea level thrust to be generated is roughly 1.43 MN. Given the sea level specific impulse of 390 s, leads to a mass flow rate of 371.8 kg/s and a burn time of 45.7 ton/371.8 kg/s = 123 s. Notice that for now we have assumed a constant thrust level, but in reality some means of reducing the thrust level may be necessary to limit acceleration levels.

Total impulse delivered by the system (constant thrust) is 1.43 MN x 123 s = 176 MNs. This is more than 20 MNs and hence a pump-fed feed system is selected. The fuel pump needs to deliver 893 liter/s as compared to 332 liter/s for the oxidizer pump.

Suppose that instead of selecting a pump feed system, we would have selected a regulated pressure system providing a constant pressure of 20 bar and using nitrogen as a pressurant. At end of operation, the pressurant fully fills up the propellant tank, meaning that we have for the fuel tank 109.9 m^3 of nitrogen stored in the tank at a pressure of 20 bar. At room temperature (293K), this indicates a mass density of about 23 kg/m3. This gives a total pressurant mass in the fuel tank of 2526 kg. To this we should also add the mass of the high pressure gas storage tank; this is left for you to explore for yourself. In any case, when storing this gas at a pressure of 200 bar in this storage tank, this tank needs to have a volume of roughly 11 m^3.

6.2.2 Solid rocket units/motors

Solid rocket motors for SLVs are used as part of:

- Solid rocket boosters that work in part parallel to a longer burning sustainer stage, like for the Titan IV rocket, Space Shuttle and Ariane 5; Booster stages are mounted parallel to the core stage and provides for the majority of thrust when active. As they are mounted in parallel, they are equipped with a nose cone to provide for low drag.
- Strap on stages that are used as an addition to augment the payload or range capability of rockets; Strap on stages on an individual basis generally have modest thrust as compared to the central core stage
- Launcher stages, like for Vega, Taurus, and Pegasus; Launcher stages are characterized by the absence of a nose section. They do allow for interstages and or an aft skirt to be connected for serial staging.
- Stages for missiles and sounding rockets; such stages generally have high T/W ratio; otherwise they are of the same basic lay-out as launcher stages.

Figure 36 shows a breakdown of a typical Solid Rocket Motor. It essentially consists of a solid block of propellant stored in a casing and a nozzle that accelerates the combustion gases to a high exhaust velocity.

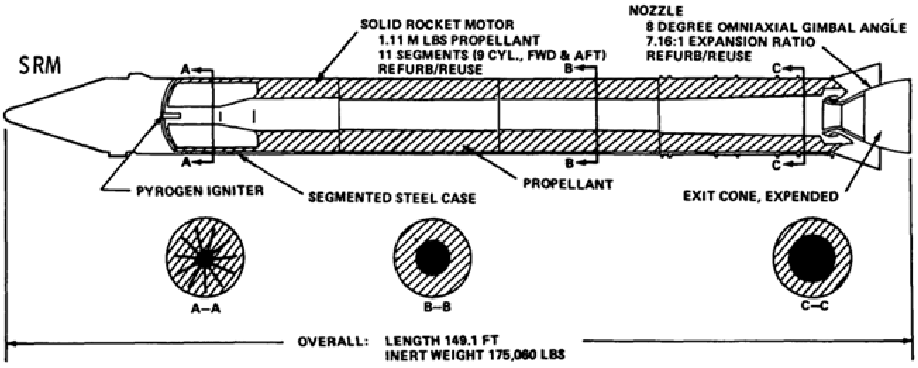


Figure 36: Schematic of Space Shuttle Solid Rocket Booster (courtesy NASA)

Solid propellants for SLV rocket stages generally consist of a solid fuel and oxidizer contained in a solid binder (some type of rubber). Typical specific impulse levels attained for solid rocket motors are in the range of 170-265 s at sea level, depending on the propellant composition, see appendix D. In vacuum the values can be up to about 285-290 s. Typical solid propellant mass densities are in range 1500-1900 kg/m^3, which allows for realizing compact rockets (rocket stages) with a very small cross-section (low drag area), but also keeps the mass of the casing holding the propellants low.

The thrust generation of a solid (propellant) rocket motor is similar to that of a liquid (propellant) rocket engine. Like for the LRE, the propellant combination chosen largely determines the rocket exhaust velocity or the specific impulse. Also like for the LRE, the mass flow rate that is expelled then determines the thrust. For estimating thrust and burn time, see relations in previous section.

For a given propellant mass, the solid propellant volume can be determined using known solid propellant mass density. Total motor volume follows by taking into account the volumetric loading factor, i.e. the propellant to total motor volume ratio:

$$V_{motor} = \frac{V_{propellant}}{volumetric\ loading\ factor} \qquad [25]$$

Typical values for this factor are in range 0.79 to 0.95, see Figure 38. As said earlier, the high propellant mass density allows for a relatively (compared to liquid rocket systems) compact design.

For a more extensive description of the characteristics of solid rockets, see appendix D, where solids are compared with liquids and hybrid propellant rockets. The latter are rockets that use a solid fuel and liquid oxidizer (not discussed in this work).

Next figure shows some details of a solid rocket motor that forms part of a rocket stage. It mainly consists of a casing holding the propellant and a rocket nozzle that accelerates the propellant to a high exhaust velocity and produces thrust. The solid propellant charge or grain burns like ordinary firework from the inside out, which allows for a large mass flow rate and hence high thrust (of up to several tens of MN). Like fireworks, solid rocket motors are of a single burn design -- no restart capability.

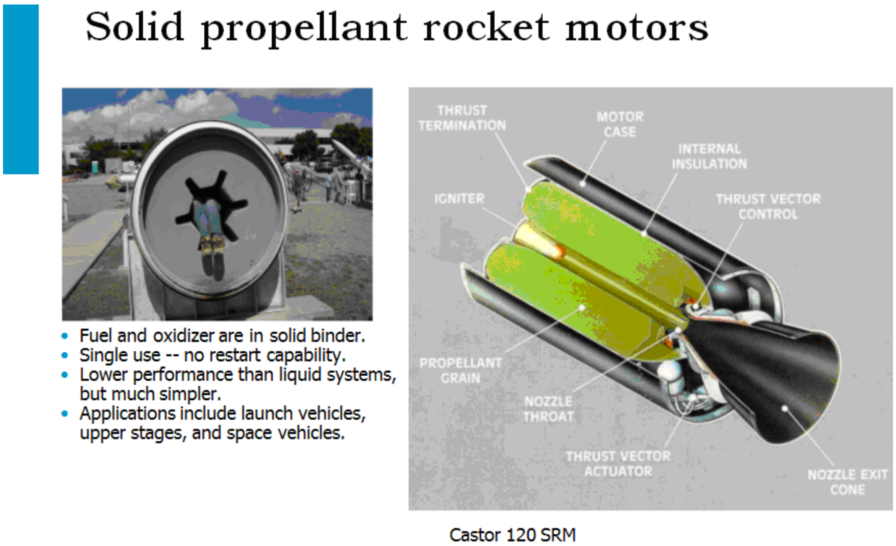


Figure 37: Castor 120 SRM (courtesy ATK)

By shaping the grain a special thrust profile (thrust versus time) can be obtained. This shaping is referred to as thrust programming. Figure below shows some conventional grain shapes (cross-sections) and the accompanying thrust profile. It also shows example missions (mission phases) where they can be applied.

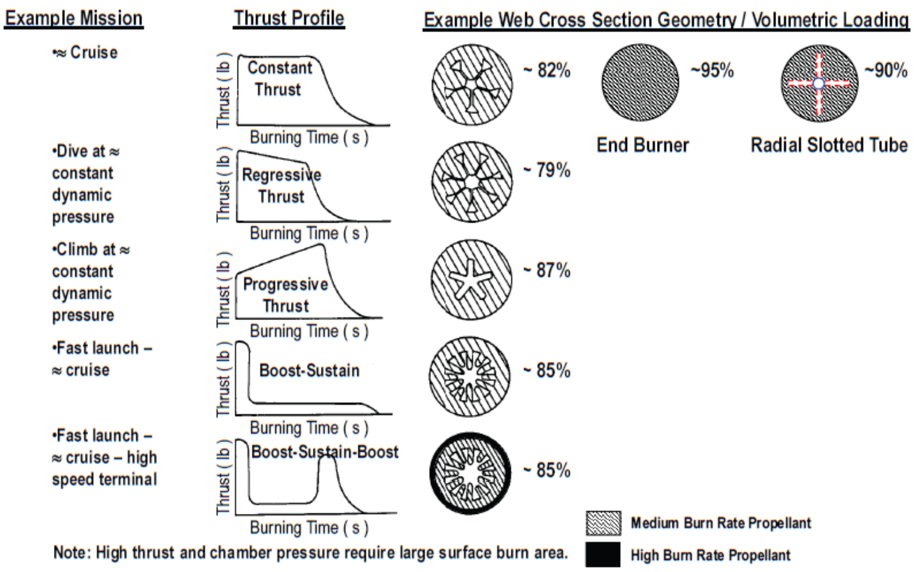


Figure 38: Solid rocket motor thrust programs [Fleeman]

More info on the effect of grain shape on the thrust produced can be obtained from a later course on propulsion and power.

Solid rocket motors can come in a great number of sizes with diameters ranging from several meters down to a few centimeters. Most motors are of a more or less cylindrical design, but may range from short and stubby to long and slender, see next figure.

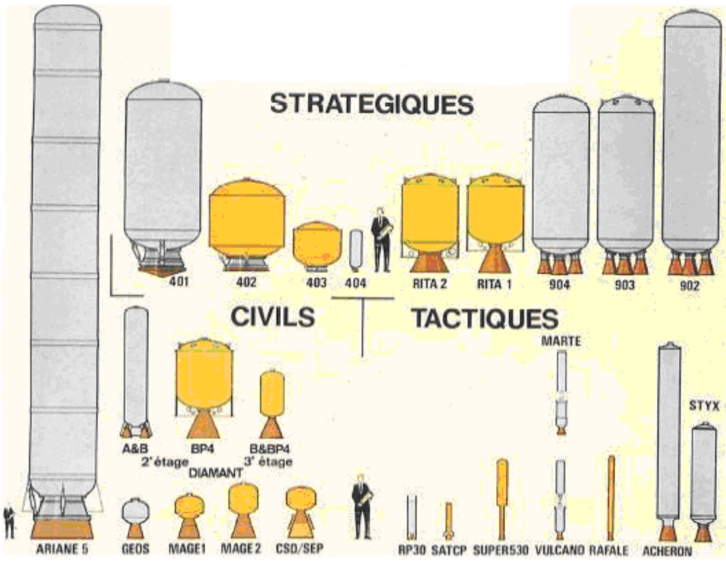


Figure 39: Solid rocket stages (courtesy Safran)

Notice that also solid rocket motors can be equipped with multiple nozzles, see motors designated 902 to 904 in above figure. This allows for limiting the length of the stage.

Typical specific impulse levels attained by SRMs are in range 170-265 s at sea level, depending on the type of propellant used, see appendix D. In vacuum the values can be maximum about 285-290 s. Size of SRMs depend largely on the mass density of the solid propellant. Solid propellant mass density typically is in range 1500-1900 kg/m^3, which allows for realizing compact motors (and hence rockets) with a very small cross-section (low drag area), but also keeps the mass of the casing holding the propellants low. A more extensive overview of existing solid rocket motors and their performances can be obtained from [ATK] or [Boury], see also appendix C, table on large SRMs. Typical failure rate can be obtained from appendix D, whereas Appendix E provides for an SRM mass estimation relationship. Further estimation relationships, including SRM cost, mass and size relationships can be obtained from [Frank].

6.2.3 Solid-liquid comparison

Specific impulse of solid rocket motors is usually below that of most bipropellant liquid rocket engines. So for the same performance in terms of delta-V they need more propellant. An advantage of solid propellants though is that they have a much higher mass density than most liquid propellant combinations, which allows building smaller and hence lighter rockets. In addition, solid propellants allow for easier storage over longer periods of time. A further advantage of solid propellants is that they lead to much simpler rockets that are more reliable and offer low cost (up to a factor 3 per unit of total impulse delivered) than their liquid counter parts. The higher reliability is because solid stages fail roughly 5-7 times less than liquid stages. For further information, see appendix D.

6.2.4 Thrust control

To limit the acceleration levels produced during flight some means of thrust magnitude control may need to be added. Thrust Magnitude Control (TMC) is the capability to control/change the thrust of an individual rocket motor. For liquid rocket motors thrust magnitude control is expressed as a

'percentage' (%) of nominal thrust; For example, a throttling capability of 50% means that the thrust can be reduced to 50% of its nominal value. Such a capability is vital for launcher stages to allow reducing acceleration loads towards the end of the flight, when the propellant tanks are almost empty. Thrust magnitude control is also applied on such vehicles as lunar landers to allow for soft landing.
Thrust magnitude control can be accomplished by:

- Switching off (some, not all) engines; this may require multiple engines that can be switched off at will.
- Reducing the thrust produced
    - Liquid rocket engines: Throttling
    - Solid rocket motors: Thrust programming, see Figure 38.

TMC is mostly based on reducing the mass flow rate of propellants. In normal operation the mass flow rate is 100%, but when throttling, the mass flow rate is greatly reduced. For instance, for a soft lunar landing, a throttle range down to 10% of the nominal (100%) thrust may be needed.

Adding a means of TMC for pump-fed liquid rocket systems may be quite costly, as it also makes the pump design more complicated.

For steering the rocket also a means of Thrust Vector Control may be necessary. This is dealt with to some extent in the next section.

6.3 Aerodynamics, stability and control

<u>Aerodynamics</u>
When flying through the atmosphere, aerodynamic forces and moments will act on the rocket. For a simplified control analysis, it suffices to consider only the lift (L) and drag (D) force, see Figure 40 and the pitching moment (M) about the pitch axis. Knowledge of the forces and moments is needed to allow for selecting the proper thrust level and for the design of the vehicle controls.

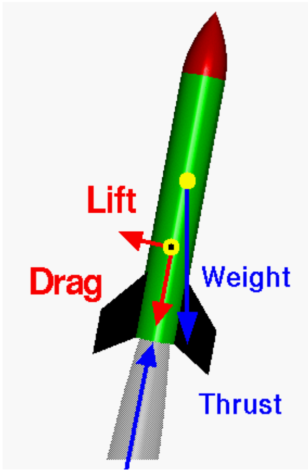


Figure 40: Aerodynamic forces on rocket

The forces and moment can be determined using:

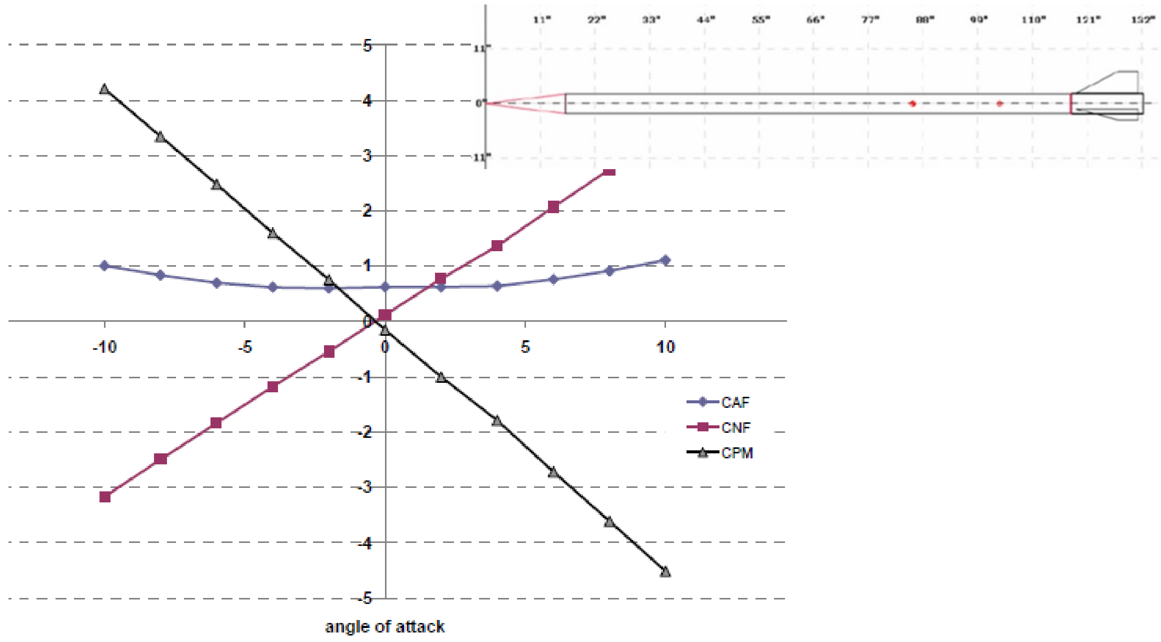
$$D = C_D \cdot \mathrm{q} \cdot \mathrm{A} \qquad [26]$$

$$L = C_L \cdot \mathrm{q} \cdot \mathrm{A} \qquad [27]$$

$$M = C_M \cdot \mathrm{q} \cdot \mathrm{A} \cdot \mathrm{l} \qquad [28]$$

Here C is aerodynamic coefficient, q is dynamic pressure, A is reference area of rocket and l is reference length of rocket. At small angles of attack: $C_D = C_A$ (axial force coefficient) and $C_L = C_N$ (normal force coefficient).

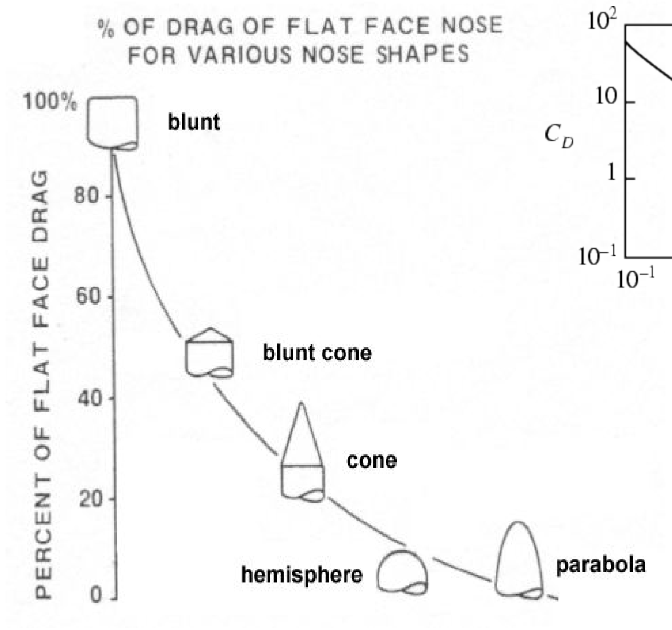
Aerodynamic coefficients come from aerodynamic analysis (see lectures on aerodynamics). Dynamic pressure can be determined based on known density of the atmosphere and the flight velocity. Reference area and reference length are usually defined in accordance with the aerodynamic coefficients. Figure 41 shows force coefficients and moment coefficient for a typical small SLV. At zero angle of attack, the normal force and pitching moment coefficient are both zero. The axial force coefficient differs from zero.

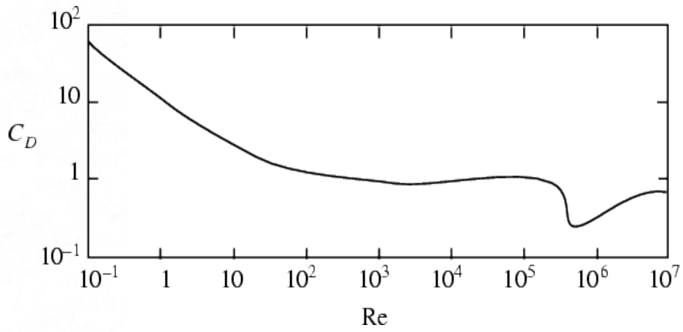


Values of C_{AF}, C_{NF} and C_{PM} versus angle of attack.

Figure 41: Aerodynamic coefficients rocket vehicle; CA$_F$ and CN$_F$ are axial and normal force coefficient, CP$_M$ is pitching moment coefficient. Reference area and length are body cross-sectional area and body diameter, respectively

When travelling through the atmosphere, it is important that the drag is as low as possible. This can be accomplished by selecting an appropriate body shape.





Drag coefficient of an infinite circular cylinder of diameter *d* (based on frontal area) versus Reynolds (Re) number

Figure 42: Effect of nose shape on vehicle drag + effect of Re on drag coefficient

Figure 42 (r) shows typical values of drag coefficient of an infinite circular cylinder for various Reynolds numbers. It follows that with increasing Mach number and or by reducing frontal area the value of this coefficient can be reduced. Only at very high Reynolds number the coefficient increases again. Using the figure on the left, the effect of different shapes can be determined.

Lift becomes important in case the vehicle flies at an angle of attack. Such conditions may occur when changing the flight path angle and/or under conditions of strong winds. Next to lift, when flying at some angle of attack also we should reckon with aerodynamic moment about the pitch axis.

For some modern launchers, it is considered to equip them with wings giving them airplane like characteristics, meaning they design the launcher to develop lift, thereby allowing for the vehicle to fly like a "sailplane". A first such example is the Space Shuttle. Figure 43 shows Space Shuttle maximum Lift to Drag (L/D) values in comparison with that of a new design (Bargouzin launcher, see insert); the higher the L/D value, the larger the cross and down range capability.

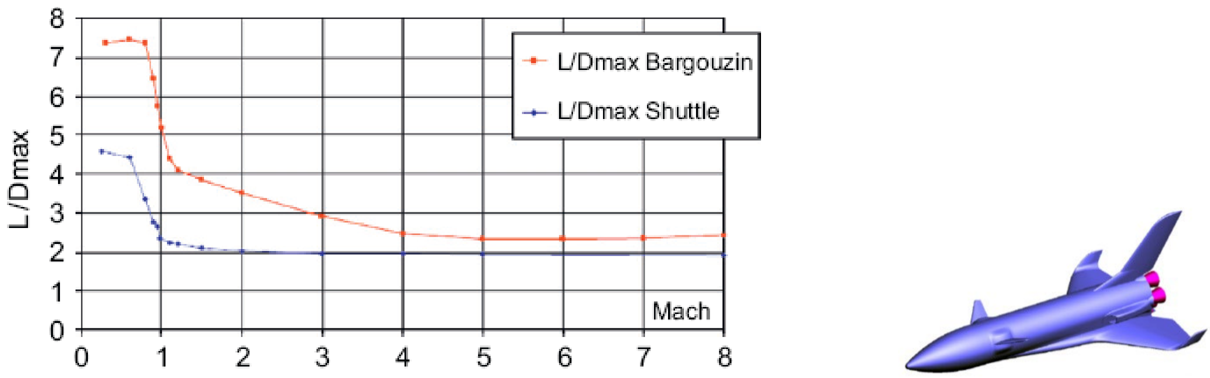


Figure 43: Aerodynamic characteristics of advanced airplane like space launcher

A first approach to determine the aerodynamic coefficients of a rocket can be by using existing data from more or less comparable vehicles as collected in the Missile DatCom (Data Compendium), which is ITAR-controlled. Another way is to use ESDU sheets [ESDU] or to use computerized prediction tools like the RASAero software (see http://www.rasaero.com/), which is suited for predicting aerodynamic coefficients of orbital rockets, sounding rockets, amateur rockets, and so on.

Example: Pitching moment on rocket

In this example, we consider the Ariane 44P rocket. More in particular, we consider this rocket at one particular instant during its ascent. At this instant, the rocket reaches a vertical flight velocity of 600 m/s at 15 km altitude (=> air density = 0.15 kg/m^3). At that altitude, it experiences a horizontal wind gust of 60 m/s (note this velocity is much higher than actually can be expected at that altitude, see Figure 44). The angle of attack of the vehicle is:

$$\tan^{-1}(\alpha) = \frac{60m/s}{600m/s} \Rightarrow \alpha = 5.7\,\text{deg}.$$

Suppose that from aerodynamic analysis follows an aerodynamic pitching moment under these conditions of C_M = 0.8.

It follows for the aerodynamic pitching moment experienced by the launcher:

$$0.8*0.5*0.15*(600^2+60^2)*\pi/4(3.8)^2*3.8 = 940\ kNm$$

Here the rocket diameter has been taken equal to 3.8 m (boosters have been discarded already) and booster diameter has been taken as reference length.

Note that the above calculated moment has to be compensated for by the launcher controls.

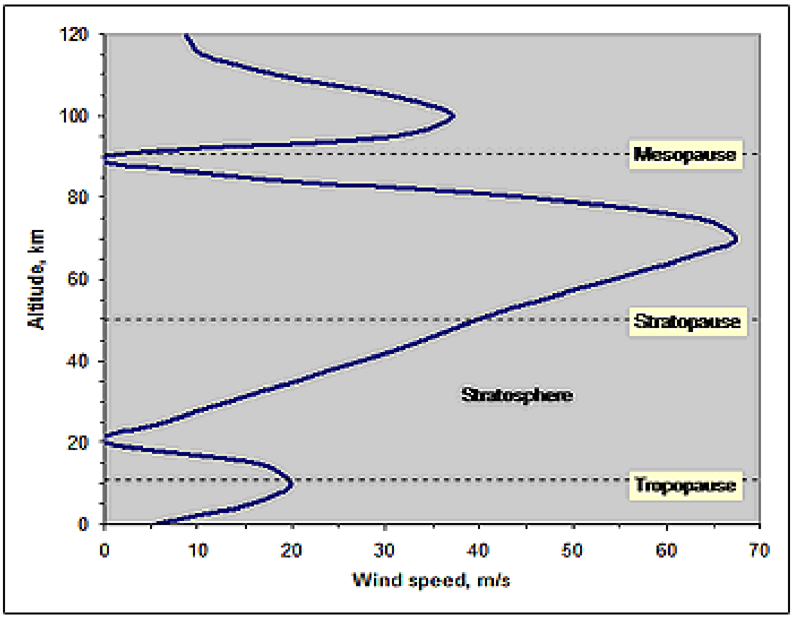
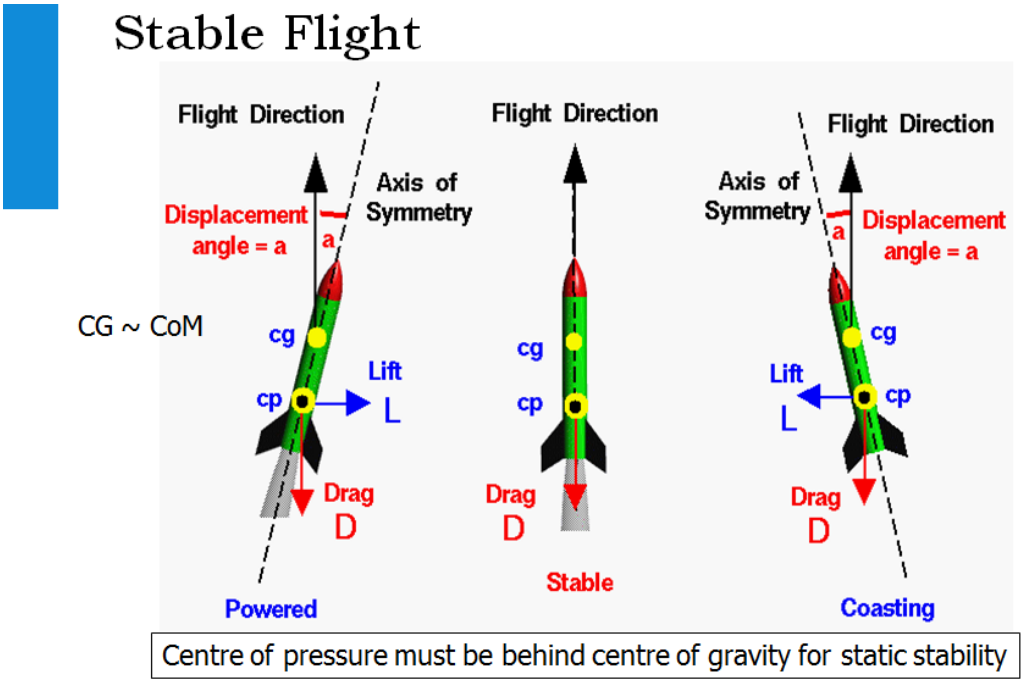


Figure 44: Wind speed versus altitude (from: http://www.intercomms.net/AUG03/content/struzak1.php).

Stability

A rocket is stable in case center of pressure is behind center of gravity. With respect to the motion about the pitch axis, this would mean that the derivative of the pitching moment coefficient with angle of attack is negative. During ascent flight propellant mass reduces and at certain instants in time stages are discarded. As a consequence both the aerodynamic center and cg may shift. For instance, the aerodynamic center of a rocket generally moves forward as the rocket reaches higher Mach numbers.



So to be able determine whether a rocket is inherently stable or not we need to know the location of the CG and the CoP. How CG is determined has been discussed earlier, so here we will only present an example calculation based on Ariane 44P, see figure.

Given is that we consider all stages as solid cylinders with following data:

- *SRB: 12.6 ton each and 11.5 m high*
- *Stage 1: 255 ton and 25.4 m high*
- *Stage 2: 38.5 ton and 11.6 m high*
- *Stage 3: 12.4 ton and 10.2 m high*

To solve for the location of the CG, we assume that:
1) CG is CoM
2) CoM of individual cylinders is in geometric centre
3) CoM of total vehicle is situated a distance x above CoM of first stage

It follows:
$255 * x + 4 * 12.6 * (25.4/2 + x - 11.5/2) =$
$38.5 * (25.4 + 11.6/2 - (25.4/2 + x)) + 12.4 * (25.4 + 11.6 + 10.2/2 - (25.4/2 + x))$

Solving for the unknown distance x gives a value of 2 m. So the CoM of the rocket is 12.7 + x = 14.7 m above ground level.

It should be clear that during flight the propellant tanks will empty and hence we need to make sure whether the CG shifts and if so, how much. This is left for you to explore when making your own design.

Now we turn to the calculation of the CoP. For this we will use a simplified method based on projected area (plane perpendicular to the flow direction) and location of CoP. Of course, in case detailed information on how the pitching moment changes with a change in angle of attack, the below method can be neglected.

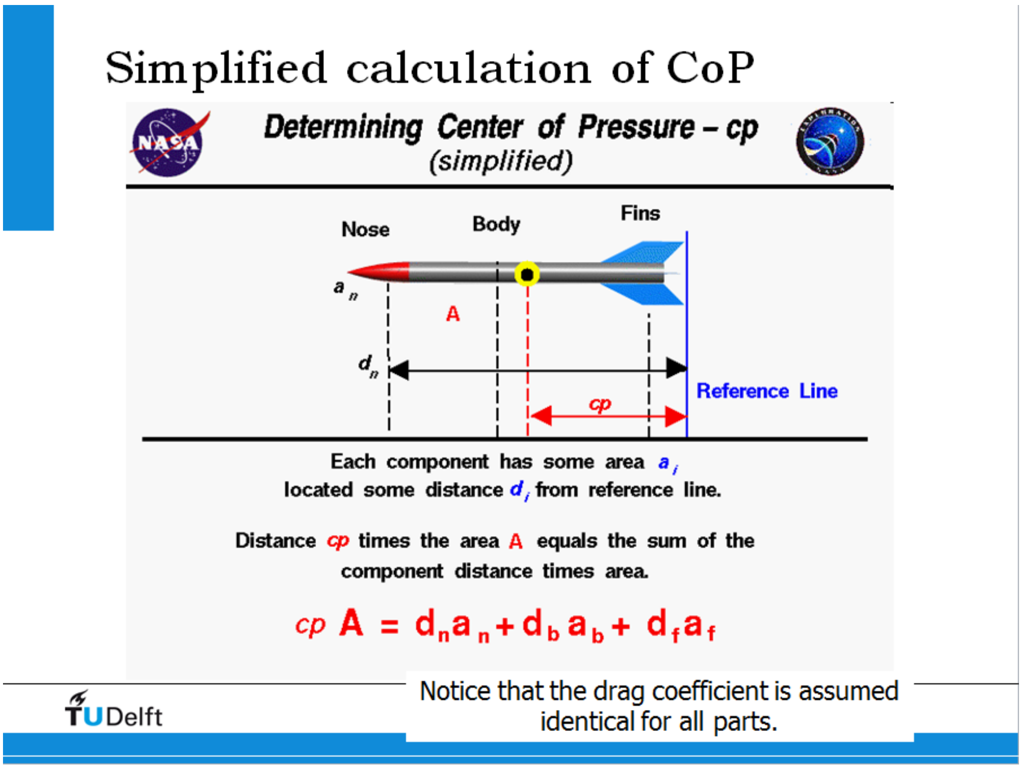


Figure shows a simplified version of the calculation procedure that you can use to calculate the CoP of a rocket. We assume that we already know the projected area and location, relative to some reference location, of each of the major parts of the rocket: the nose, body tube, and fins. The projected area **A** of the rocket is the sum of the projected area **a** of the components.

$$A = \mathrm{a(nose)} + \mathrm{a(tube)} + \mathrm{a(fins)} = \sum_{i=1}^{n} a_i \qquad [29]$$

Since the center of pressure is an average location of the projected area, we can say that the area of the whole rocket times the location of the center of pressure CoP is equal to the sum of the projected area of each component times the distance **d** of that component from the reference location.

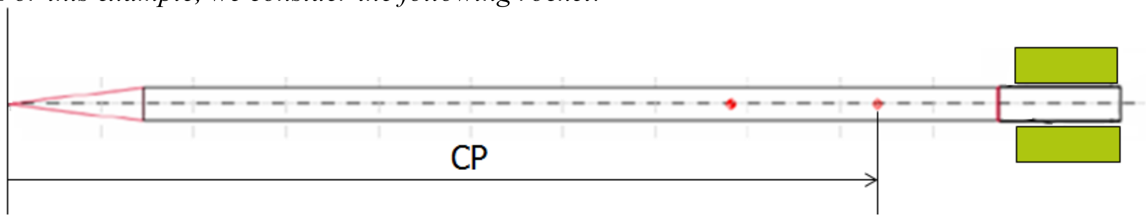
$$A * CoP = [a * d](nose) + [a * d](tube) + [a * d](fins) \qquad [30]$$

The "location" of each component is the distance of each component's center of pressure from the reference line. So you must calculate or determine the center of pressure of each of the components.

To determine the location of CoP on the vehicle axis (measured from the nose) we have to consider the rocket in cross flow. In that case we consider the projected area of the various elements in a direction perpendicular to the vehicle axis. For instance, the projected area of the body tube is a rectangle. Its CoP is on the axis, half way between the end planes. The same we do for the fins and the nose, etc.

Simplified calculation of CoP: Example

For this example, we consider the following rocket:



For this rocket are given:

- *Conical nose of height 20cm with base of 5cm*
- *Cylindrical body of height 150cm and diameter of 5cm*
- *2 fins of square plan form with length of 20 cm and width of 5 cm.*

Now we are looking at the location of the centre of pressure (CP) along the longitudinal axis.

The total projected area A (= perpendicular to the flow):

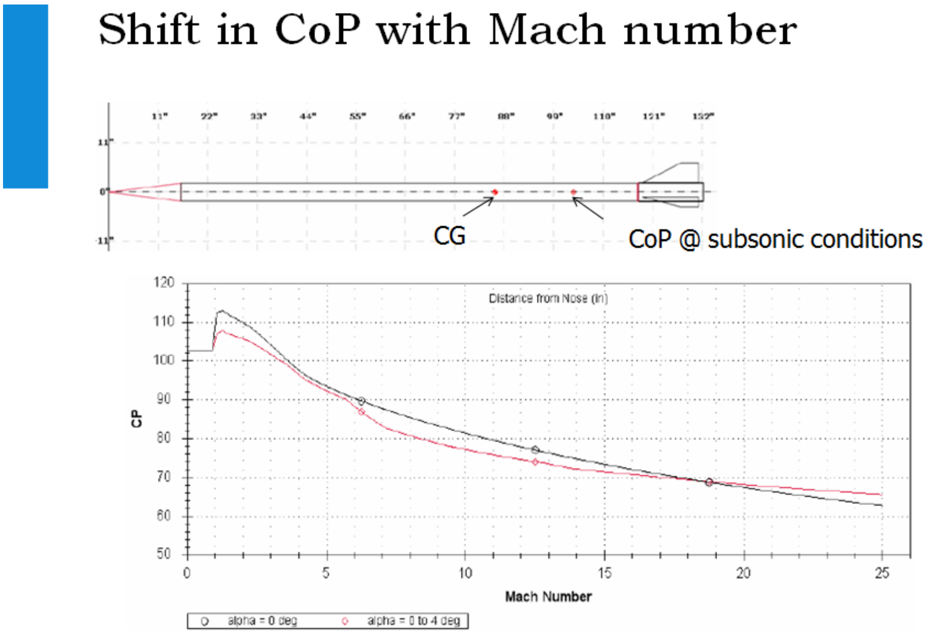
- $A = A_{nose} + A_{body} + A_{fins} = 0.5(20)(5)+(150)(5)+2*(20*5)$
- $A = 1000\ cm^2$

It follows for location of CP along the longitudinal direction:

- $A * CP = A_{nose} * d_n + A_{body} * d_b + A_{fins} * d_f$
- $1000 * CP = 50 * 2/3 * 20 + 750 * (20+75) + 200 * (20+150-10)$
- $CP = 103.92\ cm$

The location of the CP is quite forward (total length of vehicle is 170 cm). Consider increasing the size of the fins to move the CoP further back!!

The simple method discussed in the foregoing allows for making a quick check on stability. However, it is only a very limited method and does not allow for e.g. taking into account the shift in CoP with varying Mach number, see below. For this more advanced aerodynamic methods are needed that allow for determining the change in aerodynamic coefficients over the full flight regime.



Controls
A launch vehicle usually is controlled (steered) by the guidance system, see also later, that provides inputs to the control actuators, like thrusters, aerodynamic rudders, etc.. The next figure schematically shows 4 different methods of how SLVs are steered.

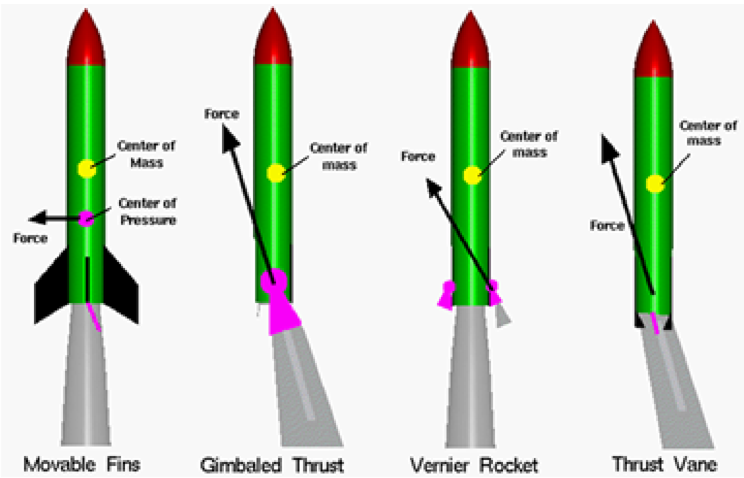


Figure 45: Examples of controls [Wikipedia]

The following text has been taken in part from [Wikipedia].

Early rockets and current air-to-air missiles typically use **movable fins** at the rear of the rocket. The movable fin adjusts the amount of the aerodynamic force on the rocket. The aerodynamic force acts through the centre of pressure which is normally not located at the centre of gravity. The difference in location generates the torque about the centre of gravity, or centre of mass. On the figure, the trailing edge of the fin facing us has been coloured magenta and has been deflected to the right. The resulting aerodynamic force would move the nose of the rocket to the right. This method allows for control about all three vehicle axis, but is limited in that it requires an atmosphere to work. So at high altitude, this system the system is not effective.

Most modern rockets mechanically rotate, or gimbal the engine or nozzle to produce a control torque. In a gimballed thrust system, the exhaust nozzle of the rocket can be swivelled from side to side. As the nozzle is moved, the direction of the thrust is changed relative to the centre of gravity of the rocket. This technique allows for pitch and yaw control during powered flight (engines active) and is used on most of today's modern large liquid propellant rocket engines, like the Space Shuttle Main Engine, the RS-68 engine (successor to the RS-27) used on the Delta launcher, and Ariane 5's HM 60 engine. A somewhat complicating factor is that all propellant leads must be flexible.

Some rockets, like the Atlas missile and the Soyuz rocket use small additional rocket engines at the bottom of the main rocket to generate the control torque. The small control rockets are called **vernier rockets**. In Figure 45, the right vernier rocket engine is coloured magenta and has been fired to cause the nose of the larger rocket to move to the right.

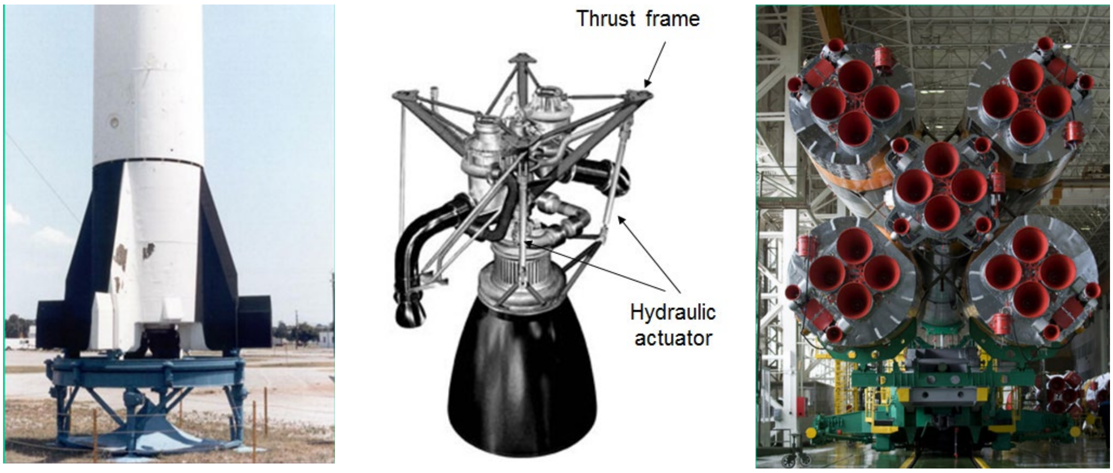


Figure 46: Redstone rocket showing movable fins at base (figure left) and Russian Soyuz rocket showing Vernier engines at base (4 on the cores stage and 2 each for the boosters). Middle picture shows RS-68 rocket engine with vectorable nozzle.

On some early rockets, like the V2 and Redstone rocket, small **thrust vanes** were placed in the exhaust stream of the main rocket to deflect the thrust and produce a control torque. In the figure, a thrust vane has been coloured magenta and is deflected to the right. This causes the exhaust stream to be deflected and the nose of the rocket would move the right.

A fifth method that can be applied in case multiple engines are operating at the same time is by changing the thrust of one or more of the engines as to produce a torque on the vehicle. This method is becoming more and more popular as it allows for omitting the sometimes heavy TVC provisions.

A sixth method is by using a reaction control system (RCS) consisting of small clusters of engines mounted at the circumference of the vehicle as is used on the European Ariane 5 and Vega rocket. By firing the right combination of these small rockets, the vehicle can be turned in any direction. Figure 47 below show the basics of this method. It essentially uses two clusters of 3 engines placed at the circumference of the launcher. The figure on the left shows a single cluster, whereas the figure in the middle shows the two clusters located diametrically opposite of each other on the launch vehicle circumference. Picture on right shows how the Reaction and Attitude Control System is integrated in the stage. Of the total of 6 thrusters, we use two for pitch control, 2 + 2 for yaw control and 2 + 2 for roll control. For yaw and pitch control pure torques can be generated with two engines operating at the same time, whereas for pitch only one engine is operative at any one time and also a translation results.

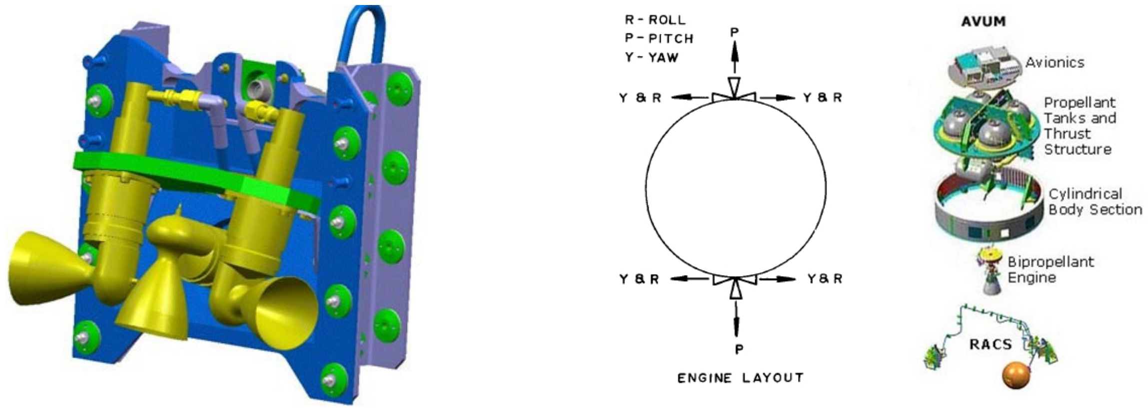


Figure 47: Vega rocket reaction control system [LVC]

The Vega RACS performs SLV roll control during second, third and fourth stage as well as for attitude control during coast phases where the main engine is not operative. The latter provides for pitch/yaw control during powered flight. The six RACS thruster are cold gas thrusters using nitrogen as propellant. Thrust level is 50 N/engine.

From [Haeselaer] we learn that the Ariane 5 reaction control system is capable of delivering a total impulse of 150 kNs. This is accomplished using a hydrazine monopropellant RCS. This system uses about 75 kg propellant stored in two small tanks pressurized with nitrogen. Besides the two tanks, the system consists of six 400 N thrusters, various control valves, filters, sensors, tubing and structural elements. The system dry mass is about 59 kg and the total wet mass is 135 kg. Each tank has a volume of 58 litres and all together the system volume comprises about 155 litres.

Next figure shows the Space Shuttle RCS lay-out. It consists of a forward (figure on left) and an aft RCS system (figure on right) and provides for the thrust needed for attitude (rotational) maneuvers (pitch, yaw and roll) and for small velocity changes along the orbiter axis (translation maneuvers). The forward RCS has 14 primary engines of 3.87 kN thrust each, and two vernier engines of ~107 N thrust each. Each aft pod has 12 primary and two vernier engines. The aft RCS system is integrated in the Orbital Maneuvering System (OMS), see figure. Each RCS consists of high-pressure gaseous helium storage tanks, pressure regulation and relief systems, a fuel and oxidizer tank, a system that distributes propellant to its engines, and thermal control systems (electrical heaters). The oxidizer and fuel are supplied under gaseous helium pressure to the RCS engines.

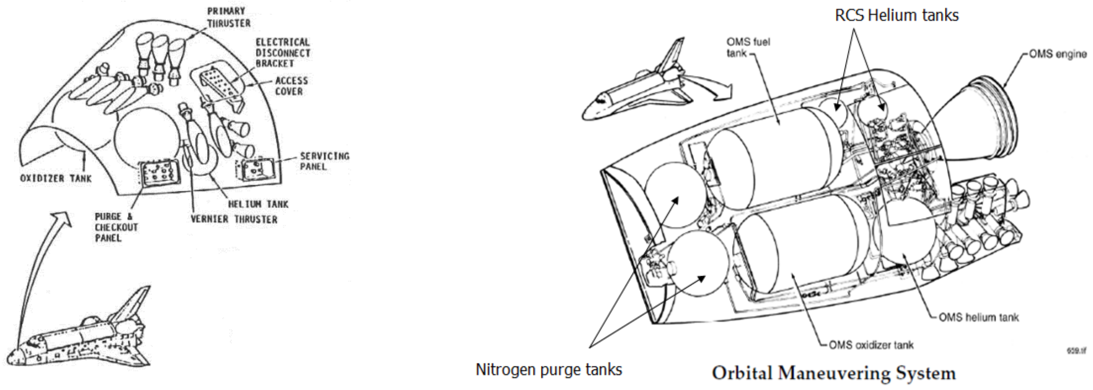


Figure 48: Space Shuttle RCS schematics (courtesy NASA)

In general, the torque (T) needed to rotate a vehicle depends on the vehicle MMOI and the required (rotational) accelerations. Considering motion in a single plane only, we find:

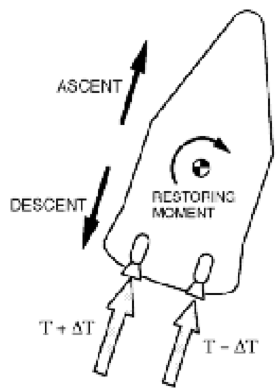
$$T = I\omega \qquad [31]$$

To allow for generating large torques with only modest thrust level (low propellant consumption), it is advantageous to put the thrusters as far away as possible from the centre of gravity (CG) of the vehicle. A torque may also need to be generated to compensate for a disturbing torque. In that case it is required that the torque generated can compensate for the disturbing torque.

Example control problem

In this example we are considering the control (motion about its CG) of an axis-symmetric rocket performing vertical descent (engines first), see figure on right. The following data are known:

- *Rocket mass: 5000 kg*
- *Nominal T/W ratio: 1.1*
- *Gravitational acceleration is 3 m/s^2*
- *4 identical rocket engines placed 1.3 m (= L_{eng}) apart symmetrically about CoM (CoM = CG).*
- *Maximum disturbance torque about pitch axis: T_m = 1572 Nm*
- *Stabilization is achieved by changing the thrust (T) of two engines (1 up and 1 down, equal %). Other two engines provide nominal thrust*



The percentage change in thrust needed for the two stabilizing engines to achieve stable landing can be calculated as follows:

- *We calculate the rocket weight: 3m/s^2 x 5000kg = 15kN*
- *Next we determine the nominal thrust level: 1.1 x 15kN = 16.5kN => 4125N per engine*
- *Torque produced by 2 engines of which one is throttled down and one up is given by: T_m = $((T + \Delta T)-(T - \Delta T)) * L_{eng}/2$*

  *It follows 2 ΔT * 0.65m = 1572Nm*
  => ΔT = 1.21 kN => $\Delta T/T$ = 1.21/4.125 = 29.3% (this would mean that the engines should have a throttle range of roughly 30% to allow for disturbance torque compensation during descent.

Some problems for exercising upon can be found in the problems section in this work and or on the course Brightspace pages.

6.4 Structure

A launcher typically is made up of a number of structural elements, like the individual stages, the inter-stages (sometimes inter-stages are referred to as skirt) connecting the various stages, and the nose and aft fairing, see Figure 49.

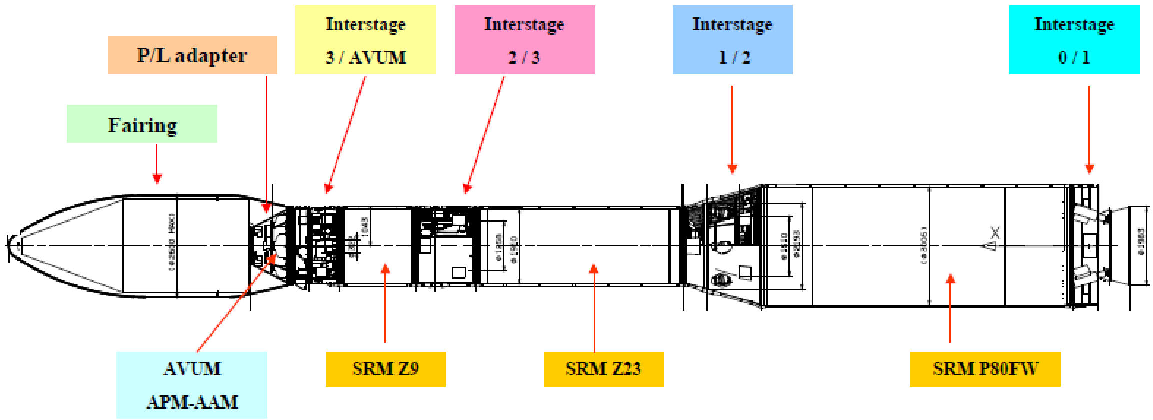


Figure 49: Main structural elements of the European Vega rocket (courtesy ESA)

As indicated earlier, critical for the design of any rocket stage is that we construct lightly to allow minimizing the structural coefficient. To allow for a light construction, most launcher designs have a semi-monocoque structure, i.e. it consists of a load bearing skin stiffened by internal components (longerond and circular stiffeners).

Figure 50 shows a typical lay-out of a liquid propellant rocket stage taken from [Huzel]. It consists of an outer cylindrical shell that holds the propellant and pressurant tanks and provides for attachment points for front/aft skirt or inter-stage. In addition, it provides for mounting of the rocket thrust structure to which the rocket engine is attached and strong points for attaching the propellant and pressurant feed lines, etc.

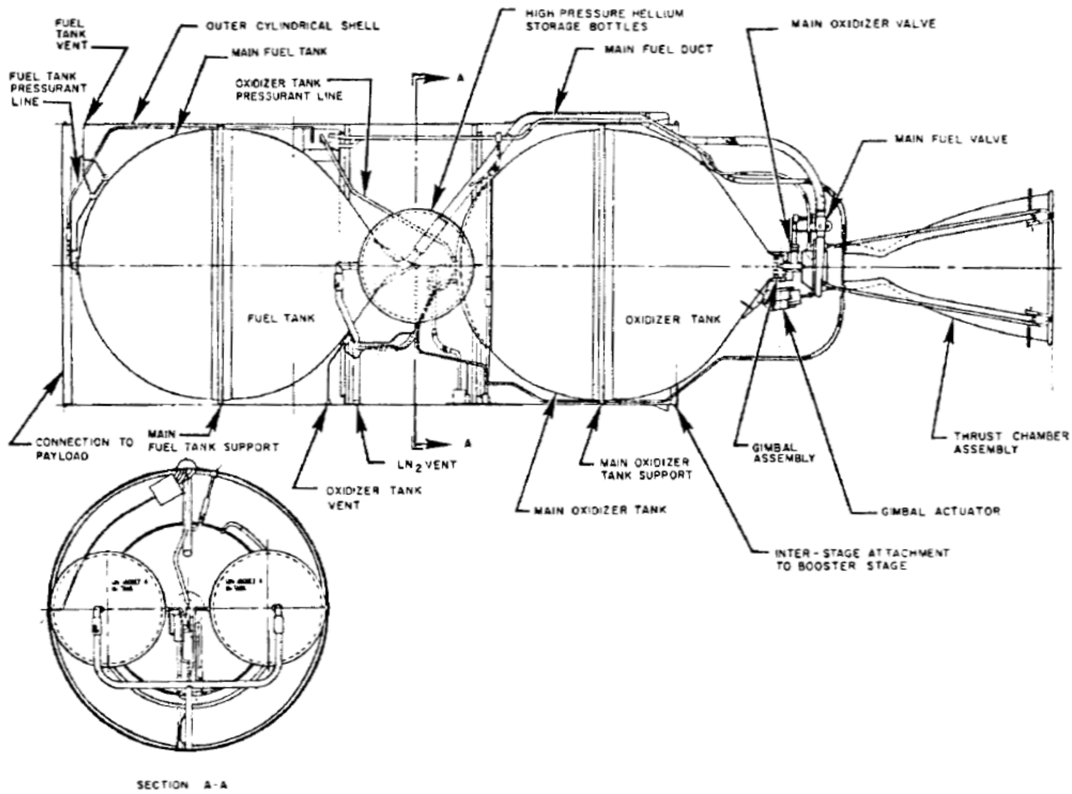


Figure 50: Liquid propellant rocket stage structure (non-integral tanks) [Huzel]

Advantage of the design shown is that the tanks only need to be designed for internal pressure loading while the stage walls are designed to carry the launch loads. This makes for a simple and safe, but also relatively heavy design.

Hence, nowadays, most launch vehicle rocket stages have an integrated tank design, meaning that the tanks are built within the normal contours of the rocket vehicle and using the skin of the vehicle as a wall of the tank, see for instance Figure 51. This allows for the tanks walls to support the skin of the launch vehicle to carry the launch loads

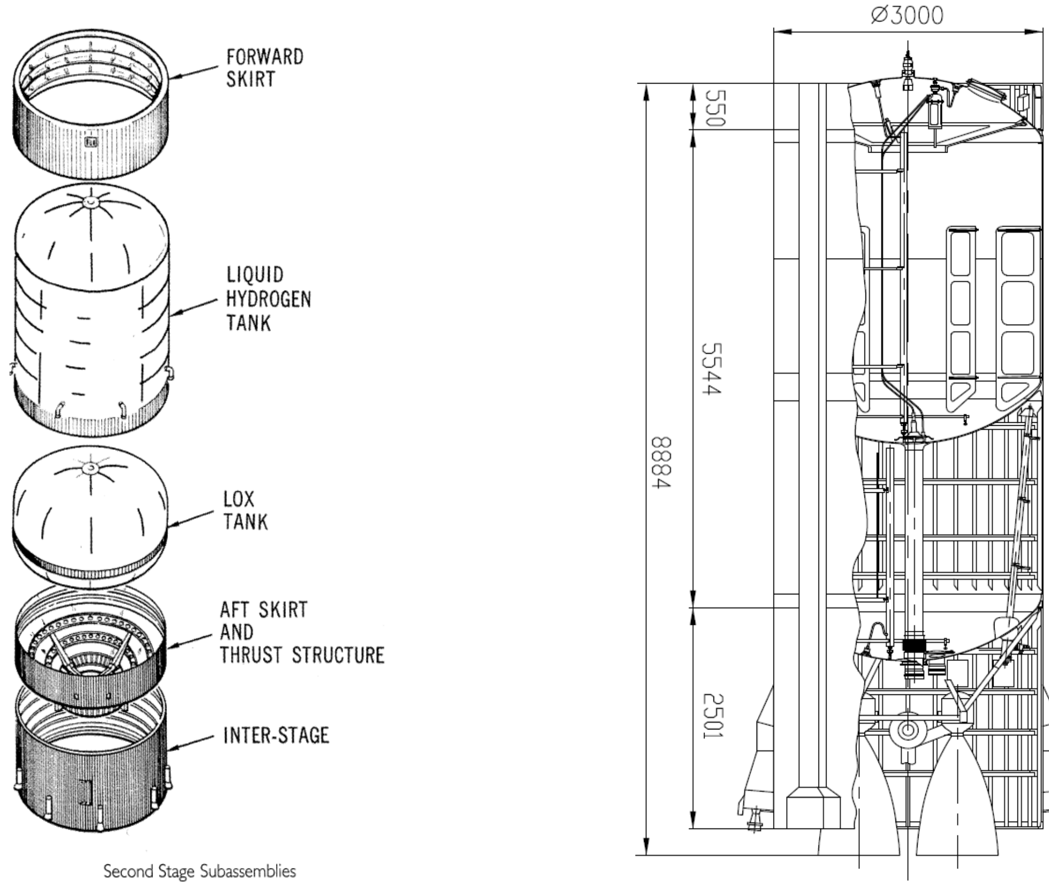


Figure 51: Structural elements of Saturn V, S-II stage (engine not included) [Lawrie] and stage with integral tank on right clearly showing the longerons and circular stiffeners that support the skin.

As indicated earlier tank size depends on the propellant load. In case of a bipropellant, at least two tanks are needed (one for the fuel and one for the oxidizer). For sizing purposes it is important to know the volume of fuel and oxidizer that needs to be stored on board. Once the propellant volume is known, the tank volume can be determined using:

$$V_{tank} = K_v \cdot V_{propellant} \qquad [32]$$

Here K_v is a factor ($K_v > 1$) that allow for taking into account that for practical reasons actual tank volume is taken about 5-10% larger. The excess volume that results is referred to as ullage and is needed to allow for thermal expansion of the propellants/tank with a change in temperature. In case also a pressurant gas is stored in the tank (consider a pressurant-fed feed system), of course the tank volume needs to be even larger ($k_v \cong 1.5$).

Tank mass

Most tanks used in rocket launchers are either of a spherical or a cylindrical shape. Also the tank wall thickness is much less than the diameter ($t \ll 0.1$ D). Because of this, a tank can be considered a thin-walled shell structure. To determine mass for such a structure, we only need information on the thickness (t) of the shell, the mass density (ρ) of the material and the surface area (S) of the shell:

$$M_{tank} = K_M \cdot \rho \cdot S \cdot t \qquad [33]$$

Besides the shell, a tank may include provisions for mounting and propellant expulsion, which also contributes to tank mass. For this reason, we use the tank constant K_M. Typical values for K_M for large tanks as used for launchers are in the range 1.1-1.3. For small tanks, this factor can be much higher up to a factor 3.

So to determine tank mass, we need to know the thickness of the tank walls. To allow determining tank thickness using simple relationship, we will assume for the present work that all tanks are of a simple spherical, cylindrical or cylinder-spherical shape, see figure, with a thin wall. In all three cases the hoop stress (see earlier lectures dealing with structures) is the limiting factor.

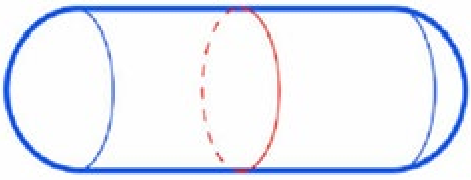


Figure 52: Simple cylindrical tank with spherical head ends

Stresses in the wall of the tank are related to internal pressure, size of the tank and the thickness of the tank wall. It follows:

- For the cylinder

$$\sigma_{hoop} = p \cdot R / t$$
$$\sigma_{axial} = p \cdot R / 2t$$
$$\sigma_{radial} \approx \text{negligible}$$

- For the spherical end cap

$$\sigma_{hoop} = \sigma_{axial} = p \cdot R / 2t$$
$$\sigma_{radial} \approx \text{negligible} \qquad [34]$$

R: radius
t: wall thickness
p: pressure
σ: stress

(Yield) strength of some commonly used structural materials is given in the next table. Actually the thickness shall be calculated both for ultimate and yield strength using appropriate safety factors. The largest thickness shall be used for mass calculations. Note that thickness of the cylindrical part may be different from that of the spherical part.

Material Properties (Wikipedia)	Young's Modulus, GPa	Elastic Limit, MPa	Density, g/cm^3
Aluminum Alloy	69	400	2.7
Carbon-Fiber Composite	530	-	1.8
Fiber-Glass Composite	125-150	-	2.5
Magnesium	45	100	1.7
Steel	200	250-700	7.8
Titanium	105-120	830	4.5

The next table (for metals) provides for minimum factors of safety in use for pressurized tanks of expendable launchers. Notice the different safety factors for yield and ultimate load. Further information and FOS for other structures (including non-metallic ones) can be obtained from [ECSS E30].

Table 10: Minimum Factors Of Safety for metallic tanks of expendable launchers [ECSS-30]

Structure type / sizing case	FOSY	FOSU
Pressurized tanks for liquid propellants and solid propellant stage boosters	1.1	1.25
Pressurized tanks for small solid propellant boosters: ▪ Pressure ▪ Other loads	 1.5 1.1	 2.0 1.25

Figure 53 shows the effect of a different tank shape on tank radius and tank weight. All tanks have same volume and same maximum shell stresses due to internal pressure. Other loads are neglected.

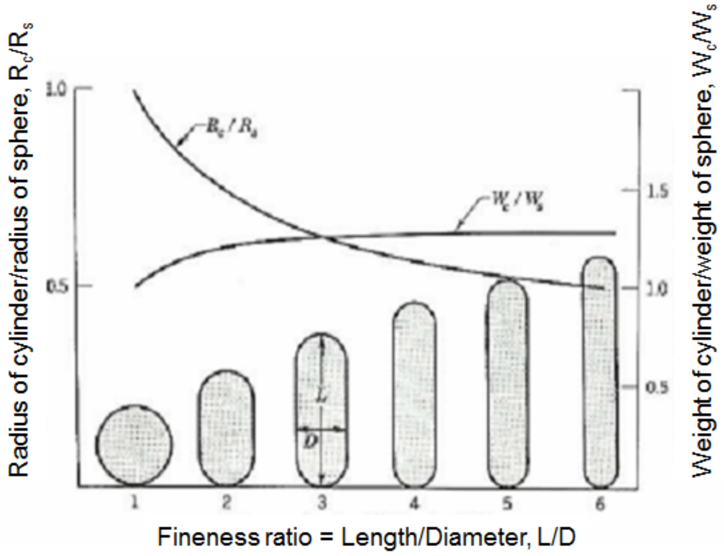


Figure 53: Effect of tank shape on tank mass [Sechler]

The figure shows that with increasing fineness ratio the cylinder diameter reduces and the weight increases. This then demonstrates that from a mass perspective the sphere is optimum. However, from the perspective of launcher frontal area and hence vehicle drag, it certainly is worth considering other tank shapes, be it that the gain in delta-V from a drag reduction may be (partly) offset by the increase in structure mass. On the other hand, cylindrical tanks can better support the skin of the SLV to carry the launch loads. This will be discussed in the next section.

Example: Tank mass
Consider the hydrogen tank of a Hydrolox (liquid oxygen – liquid hydrogen) rocket of a shape as depicted in Figure 52. *This tank is of a cylindrical shape with spherical head ends with a diameter of 5 m and a total height of 25 m (fineness ratio of 5).*

Given are:

- *Qualification load factor = 1.2*
- $FoS_y = 1.1$
- $FoS_u = 1.25$

Determine for this tank the tank mass that results based on internal pressure loading only given that you select aluminum (yield strength is 400 MPa, ultimate strength is 503 MPa and mass density is 2700 kg/m^3) as the tank material and that the tank acts as a simple pressure vessel at a pressure of maximum 3 bar. It is required that the tank is able to withstand the yield load.

Solution:
Using the relations valid for stresses in pressurized, thin-walled cylindrical tanks and taking into account the appropriate safety factor, we find for the thickness of the cylindrical part (Hoop stress is dimensioning load):

$t = MEOP\ FoS\ D/(2\ \sigma) = \sim 2.06\ mm$

Thickness of spherical part is:

$t = \sim 1.03\ mm$ *(half the thickness of the cylindrical part).*

And for ultimate strength:

$t = MEOP\ FoS\ D/(2\ \sigma) = \sim 1.86\ mm$

Thickness of spherical part is:

$t = \sim 0.93\ mm$ *(half the thickness of the cylindrical part)*

So yield strength is more critical than ultimate strength. For now we select a thickness of 2.10 mm for the cylindrical part and 1.05 mm for the spherical end-caps.

Surface area of cylindrical part: $\pi D L = \pi\ 5 \times 20 = 314.2\ m^2$
Surface area of spherical end caps: $4\ \pi r^2 = 4\ \pi\ 2.5^2 = 78.5\ m^2$

It follows a total tank mass based on shell thickness: $(314.2 \times 0.0021 + 78.5 \times 0.00105) \times 2700\ kg/m^3 = 2.0$ *ton.*

Actual tank mass (including propellant management devices and local strengthening for welding and mounting) is estimated at 1.2 x 2.0 t = 2.4 t, wherein a tank constant K = 1.2 is assumed.

Stage mass

Like tanks, also stages can be considered as thin-walled structures. This means that mass can be determined based on known mass density, surface area and wall thickness. However, for most stages, the determining loads are not internal pressure loads, but the launch loads.

How to calculate maximum stresses due to compression and tension due to launch loads for simple cylindrical structures has been discussed in the section on spacecraft design and will not be repeated here. Still students should prepare for sizing the side walls of the tanks also for tension loads due to longitudinal and transversal accelerations and due to bending.
Here we will only consider the critical buckling load of a cylindrical structure with attention to the pressure stiffening aspect that results in case of an integral tank design. Aim is to show that by having a pressurized cylinder supporting the launcher wall (pressure stiffening) a considerable increase of the critical buckling stresses and hence a lighter structure might be possible.

We start by considering the case of no internal pressure. As we have seen earlier, buckling of thin walled cylinders (no internal pressure), like for an interstage or a stage with non-integral tanks, is governed by the following relation for the critical stress σ_c [Sechler]:

$$\frac{\sigma_c}{E} = 9\left(\frac{t}{R}\right)^{1.6} + 0.16\left(\frac{t}{L}\right)^{1.3} \quad (no\ internal\ pressure)$$

[35]

Here t is wall thickness, R is radius of cylinder, L is length (or height) of cylinder, E is Young's modulus. The relation shows that critical stress can be increased by:

- Increasing E-modulus (or Young's modulus)
- Increasing wall thickness (solid wall versus honeycomb wall)
- Reducing the size of the plate fields by adding rings.

Figure 54 shows effect of radius and length of cylinder on critical stress. It can clearly be seen that for large L/R ratio, critical stress is mainly affected by the radius of the cylinder. Only for short cylinders, the length L is of importance in that a short cylinder greatly increases critical stress. For this reason circular rings are added.

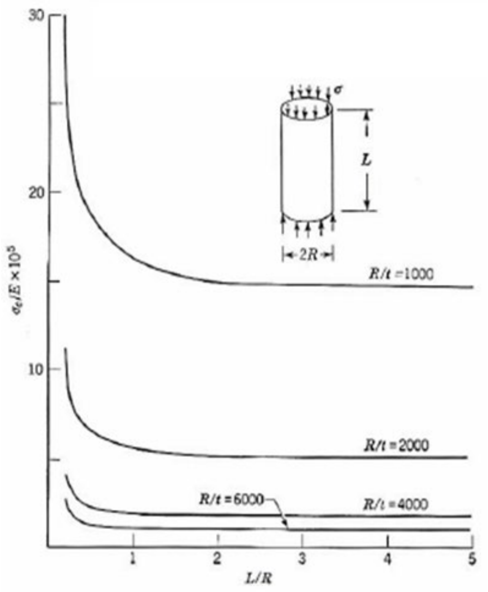


Figure 54: Effect of radius and length on critical buckling stress [Sechler]

In case the actual stresses occurring are in excess of the critical stress, the design should be adapted such that the critical stress is in excess of the actually occurring stress in the cylinder wall. One way is by an integral tank design, where the internal pressure in the tank supports the skin in preventing buckling. This is referred to as pressure stiffening of the structure. For instance, the Atlas Mercury launch vehicle was designed with balloon propellant tanks. The vehicle would collapse without internal pressurization. To prevent collapsing, the tank was filled with nitrogen gas when not fuelled.

From [Sechler], we learn that in case of internal pressure:

$$\sigma_c = (K_c + K_p)\left(\frac{Et}{R}\right) \quad [36]$$

Where:

$$K_c = 9\left(\frac{t}{R}\right)^{0.6} + 0.16\left(\frac{R}{L}\right)^{1.3}\left(\frac{t}{R}\right)^{0.3} \quad [37]$$

$$K_p = 0.191\left(\frac{p}{E}\right)\left(\frac{R}{t}\right)^2 \quad [38]$$

Example: Stage primary structure mass
The tank calculated in the foregoing example is mounted into a thin-walled cylindrical aluminium stage of diameter 5m and height 25 m. This stage supports various upper stages with a total mass of 50 ton (concentrated as a point mass on top of the stage. Maximum flight load in axial direction is 6g (both compression and tension). Determine the stage structure mass given the following data:

- *E-modulus: 71 GPa*
- *Mass density of the aluminium material of the stage: 2800 kg/m^3*
- *Ultimate tensile strength 504 MPa*
- *Same FoS as given in foregoing example*

Solution:
It follows for the axial (flight limit) load (L) of the stage under consideration:

$$L = M \text{ x } 6g = 50{,}000 \text{ x } 6 \text{ x } 9.81 = 2{,}943{,}000 \text{ N}$$

Minimum thickness (t) to provide sufficient strength (tension) is:

$$t = FoS \text{ x } L/(\sigma \text{ x } \pi \text{ x } D) = 1.2 \text{ x } 1.25 \text{ x } 2{,}934{,}000/(503E6 \text{ x } \pi \text{ x } 5) = 0.55 \text{ mm}$$

It follows for the stress occurring in the stage:

$$\sigma = P/A = 2{,}943{,}000 \text{ N}/ \pi 5*0.55E\text{-}3 = 340.6 \text{ MPa}$$

The critical axial stress in the thin-walled cylindrical stage is:

$$\sigma_c = (9*(t/R)^{1.6} + 0.16 *(t/L)^{1.3})*E$$
$$\sigma_c = (9*(0.55E\text{-}3/2.5)^{1.6} + 0.16 *(0.55E\text{-}3/25)^{1.3}*71E9)$$
$$\sigma_c = (1.265E\text{-}5 + 1.41 E\text{-}7) *71E9$$
$$\sigma_c = 0.908 \text{ MPa}$$

It turns out that the thickness of 0.55 mm is insufficient to prevent buckling. For this a thickness of about 5 mm (verify) is needed. This thickness helps reducing the axial stress in the cylindrical stage down to about 37.5 MPa and increases the critical buckling strength to about 31 MPa, which shows that actual thickness should be chosen still somewhat higher, but for now, we leave it at 5 mm.

The mass of the stage structure (M) in that case becomes:

$$M = \pi \text{ x } D \text{ x } H \text{ x } t \text{ x } \rho$$
$$M = \pi \text{ x } 5 \text{ x } 20 \text{ x } 0.005 \text{ x } 2800$$
$$M = 5498 \text{ kg}$$

To this we still must add the mass of the tank(s). This then becomes a quite heavy design, see also next example.

Example: Integral tank wall design
As a next step in our computations, we could consider an integral design with the internal pressure in the tank increasing the critical strength of the stage. Here we use a wall thickness of 2.1 mm (same as tank wall thickness earlier calculated). It follows:

$$K_o = 9*(t/R)^{.6} + 0.16 *(R/L)^{1.3}(t/R)^{0.3}$$
$$K_o = 9*(2.1E\text{-}3/2.5)^{.6} + 0.16 *(2.5/25)^{1.3} * (2.1E\text{-}3/2.5)^{0.3}$$
$$K_o = 0.128$$

$$K_p = 0.191(p/E)(R/t)^2$$
$$K_p = 0.191(3E5/71E9)(2.5/2.1E\text{-}3)^2$$
$$K_p = 1.144$$

$$\sigma_c = (K_o + K_p) * E*t/R$$
σ_c *= 75.8 MPa (slightly below the stress experienced (~89 MPa) occurring.*

Structure mass now reduces to (with integrated tank) 2309 kg. To this we should add only the additional structures also required for the separate tank design, like baffles, mounting provisions, etc. Again taking a tank constant of 1.2 gives a final mass of 2.77 t (1.2 x 2309 kg).

Note that the tanks should also be sized to withstand the weight of the propellant during launch. This will notably add to the thickness of the tank shell. This is left for later in your study.

Part of the stage structure is also formed by the thrust structure, i.e. the structure that allows for mating the engines with the tanks. Next figure shows examples of thrust structures as used in practice. The mass of this structure can contribute heavily to the overall stage mass. The analysis of such structures is considered out of scope for the present course. In appendix E an estimation relationship for the thrust structure is given.

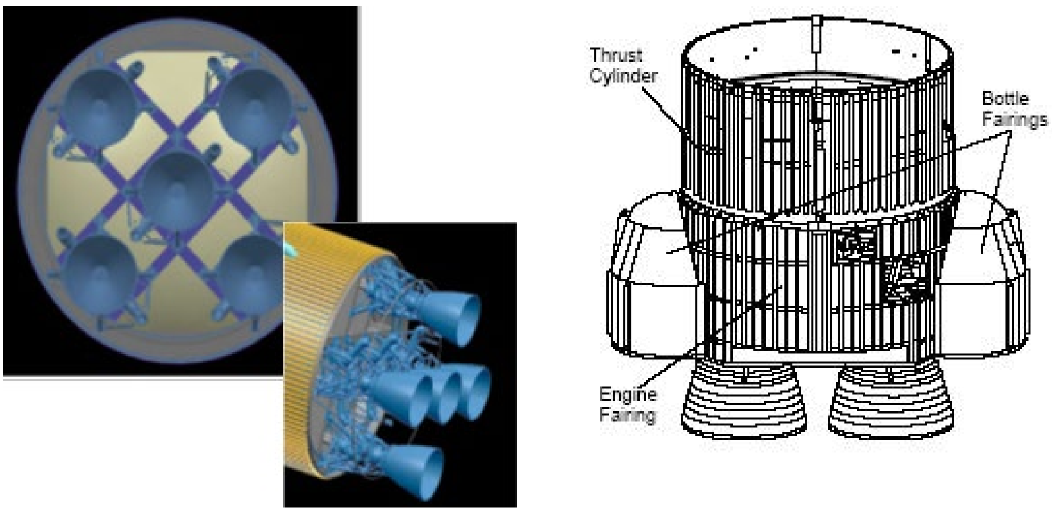


Figure 55: Thrust structure of ARES V first stage and Atlas II first stage (courtesy NASA)

Mass of inter-stages
The mass of inter-stages can be dealt with in the same way as for the mass of stages except that inter-stages rarely are defined with pressure stiffening in mind.

Effect of thermal stresses
Thermal stresses are related to excessive temperatures and/or from restricting thermal expansion. Thermal stresses can be expected when storing cryogenic propellants, for instance liquid hydrogen is stored at 20 K and liquid oxygen at about 50 K, or because of aerodynamic heating and/or heat accumulated in the wall of the combustion chamber and nozzle of a rocket engine. The latter may also induce a large heat flow to other parts of the rocket. The following aspects should be considered:

- Direct weakening of material by high temperature, see figure.
- Embrittlement of materials (metals and rubbers and plastics becoming brittle) at low temperature.
- Internal stress caused by differential temperature, e.g. on common bulkhead between hydrogen and oxygen tanks

When designing structures taking into account thermal stresses we need to make sure the material is strong enough over the full temperature range encountered. In case of large thermal gradients, thermal stresses need to be considered for their effect on total stress in the structure.

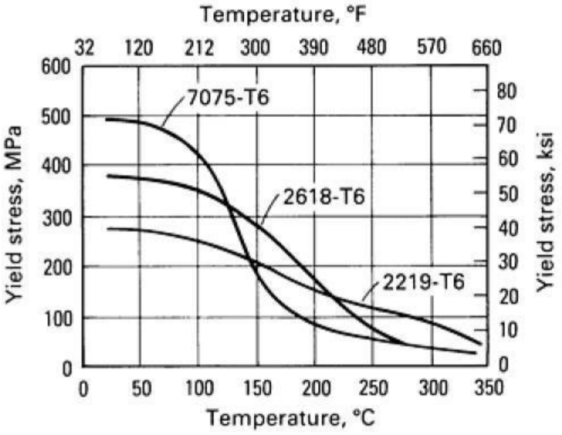


Figure 56: Yield stress versus temperature for various Aluminum alloys [Wikipedia]

6.5 Payload fairing

Payload fairing essentially is an auxiliary (meaning it does not carry the primary loads) structure that encapsulates the payload. It serves to:

- reduce vehicle drag (see discussion on launcher drag earlier);
- protect payload during ascent against impact of atmosphere, i.e. aerodynamic pressure and heating;
- maintain a controlled environment for precision instruments on board of the spacecraft carried.

Payload fairings mostly are of a cylindrical design with an aerodynamically shaped nose. Standard fairing is a cone-cylinder combination, see Figure 57. Due to aerodynamic considerations, however specialized nose designs (parabolic/elliptic) are in use as well.

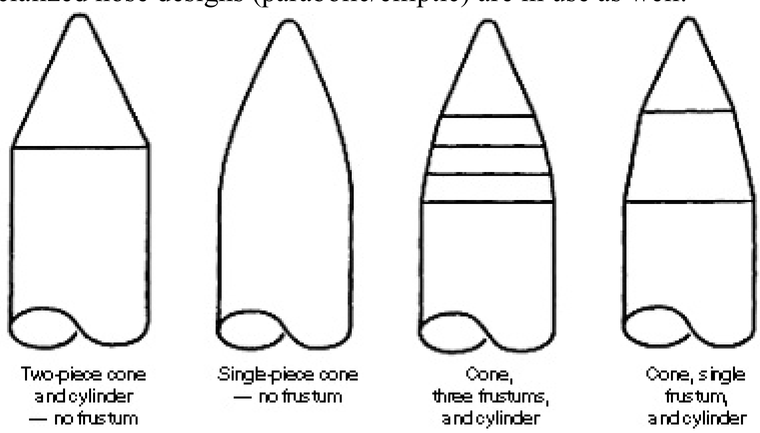


Figure 57: Typical cone-cylinder payload faring shapes (with or without use of frustum[24]) (courtesy NASA)

[24] With **frustum** is meant here the portion of a cone or that lies between two parallel planes cutting it.

Data taken from currently available designs shows that the cylindrical part of the fairing has at least an L/D in range 1-3, see Figure 58. This is to allow accommodating the payload. Overall length of the fairing typically ranges between 2.5 to 4 times fairing diameter.

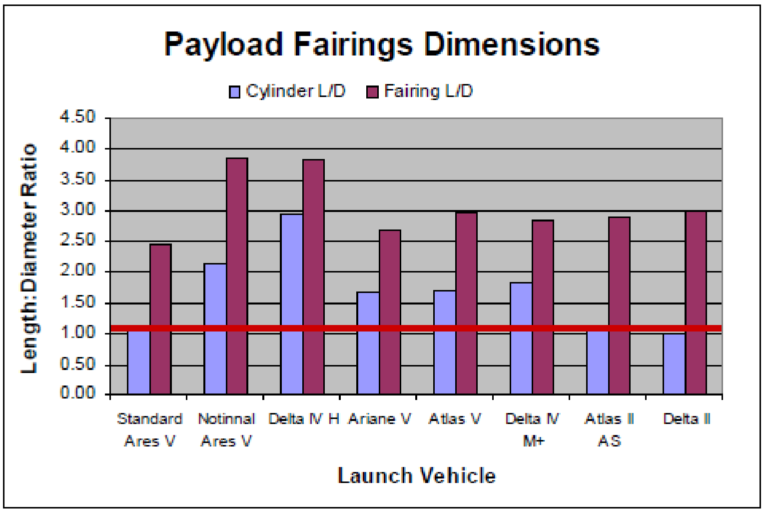


Figure 58: Typical fairing dimensions for various launchers

Some further characteristic data collected from literature are provided below.

- **H2**: The standard 4.1 m diameter, 12.0 m high 1,400 kg aluminum honeycomb core/skin 4S fairing provides a 3.7 m diameter envelope for single payloads. Separation of the clamshell halves is effected by a mild detonating fuse and springs when aerodynamic heating has fallen to 1,135 W/m^2 during stage 1 burn.
- **Pegasus**: 127 kg 2-piece composite fairing, which is jettisoned during stage 2 burn. The payload may remain attached to stage 3 or separated by clamp, following pointing to +/-2^o accuracy or spin-up. On the ground, air-conditioning maintains 21 +/-5^{o}C. Payload volume: 1.17 m diameter, 2.13 m long for 3-stage
- **Titan 4**: Construction: iso-grid aluminum 6061. Length: 17.08 m NUS (No Upper Stage), 26.23 m Centaur, 17.06 m IUS (15.25, 20.1, 23.2 m versions also available). Diameter: 5.09 m. Mass: 4,033 kg NUS, 6,073 kg Centaur, 4,026 kg IUS. Separation: low-explosive detonating fuse along seams and 12 explosive bolts divides fairing into three sections in 0.2 s.
- **Ariane 4**: Three basic lengths of Contraves' 25 mm-thick aluminum alloy honeycomb two-piece fairings are available: 8.6 m (type 01, 740 kg, 60 m^3), 9.6 m (type 02, 800 kg, 70 m^3) and 11.1 m (type 03, 86 m^3, available on special request). The fairing halves are separated by pyrotechnic cord and piston when the heating flux has reduced to 1,135 W/m^2, at about 285 s/115.3 km during stage 2's burn. The base clampband is released first. Several standard payload adapters are available. Acoustic load: 142 dB integration at launch + transonic. Thermal load: max 500 W/m^2 radiated by fairing/Spelda.
- **Ariane 5**: Contraves provides two standard 5.40 m diameter fairings to accommodate 4.57 m diameter payloads: 12.70m/2.3 t short and 17 m long units. The fairing jettisons at about 191 s/106 km during stage 1 burn when aerodynamic heating has reduced to 1,135 W/m^2. Acoustic load: 142 dB at launch/transonic. Thermal load: max 1,000 W/m^2 radiated by fairing + VEB

From these data, we learn that:

- Typical heat flow from atmosphere to environment is 1000-1200 W/m^2. Maximum temperature experienced by fairing may be up to 600 degrees Celsius.
- Fairing consists of at least two parts to ease separation/jettisoning; after separation, the parts of the fairing should drift away without endangering the payload(s).
- Fairing is typically jettisoned at about 100 km altitude ("outside the atmosphere"). At this moment, mechanical shocks and a spike in acceleration may be observed.

- Separation mechanism, see also later entry, consists of clampband, explosive bolts or pyrotechnic cord that ensures proper release and a separation system (springs, piston or other) that ensures separation.

For large payloads, hammerhead fairing are often used, i.e. the maximum diameter of the fairing exceeds the diameter of the launch rocket and is subject to severe buffeting[25] loads as it traverses transonic speeds.

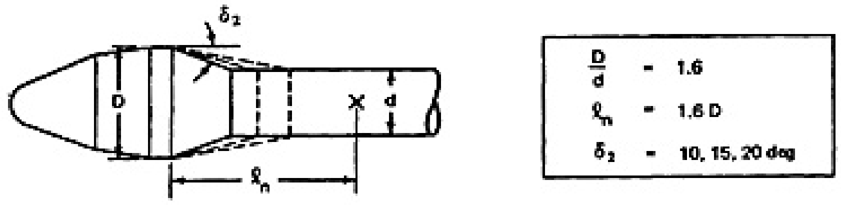


Figure 59: Typical hammerhead style fairing

Fairing volume, surface and mass
Fairing volume determines the size of the spacecraft it can contain, whereas fairing surface is important for the fairing mass. Next to fairing surface, the mass is also determined by the specific mass of the fairing construction. The next few relations are valid for a fairing with a cylindrical part of length $L_{cylindrical}$ and diameter $D_{fairing}$ and a conical nose with length L_{nose}:

$$V_{fairing} = \frac{\pi}{3}\left(\frac{D_{fairing}}{2}\right)^2 L_{nose} + \pi\left(\frac{D_{fairing}}{2}\right)^2 L_{cylindrical} \qquad [39]$$

$$S_{fairing} = \pi\left(\frac{D_{fairing}}{2}\right)\sqrt{\left(\frac{D_{fairing}}{2}\right)^2 + (L_{nose})^2} + 2\pi\left(\frac{D_{fairing}}{2}\right) L_{cylindrical} \qquad [40]$$

$$M_{fairing} = a \cdot S_{fairing} \qquad [41]$$

Typical nose cone volumes of various shapes are provided in appendix H. It should be realized though that very few spacecraft allow for usage of the full volume under a cone. Typical specific mass values of fairings (symbol “a” in above relation) are given in Appendix E.

Example problem:
A rocket of diameter 5 m requires a fairing capable of hosting a S/C of diameter 4 m and height 6 m. You select a cylindrical shape for the fairing with a conical tip of height 2 m. As fairing material you select materials and a construction that allows for a specific mass $M_{specific, fairing}$ = 10.3 kg/m^2 with SSD = 2.3 kg/m^2. Calculate:
- *Fairing mass (MLE or Most Likely Estimate)*
- *Mass of fairing in case we are required to guarantee that the probability of exceeding the calculated mass figure is below 2.5%*

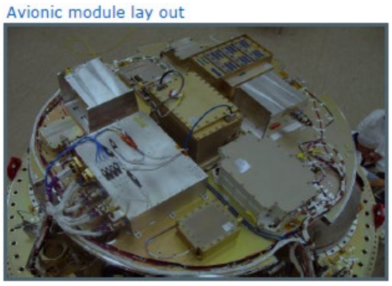
Solution:
- *Surface of cylindrical part of fairing: π x D x H = 94.2 m^2*
- *Surface of conical part of fairing: π x D x H / 2 = 15.3 m^2*
- *MLE of mass of fairing =109.5 m^2 x 10.3 kg/m^2 = 1128 kg*
- *Mass of fairing in case we should guarantee that the probability of exceeding the mass figure is less than 2.5%:*
  109.5 m^2 x (10.3 + 2 x 2.3)kg/m^2 = 1632 kg

[25] Buffeting is high-frequency instability, caused by air-flow separation or shock waves oscillations from one object striking another.

Homework problems:
1. Consult the Launch Vehicle Catalogue [LVC] and determine in what altitude range the fairing typically is ejected.
2. Determine estimation relationships for surface area of parabolic and elliptic fairings and conical frustrum.

6.6 Avionics subsystem

The avionics subsystem essentially encompasses all the electronic systems, equipment and other devices controlling an aerospace vehicle. Figure 60 shows the avionics system of the Vega and the Ares 1 rocket. The Vega avionics system consists of three subsystems:
- **GNC** - Guidance, Navigation and Control Subsystem consisting of the on-board computer for data processing and organization and an inertial reference system providing navigation data (accelerations, etc.), a command processing network and a power processing network
- **SAS** - Safeguard Subsystem including a flight termination system
- **TMS** - Telemetry subsystem for transmission of flight measurement data



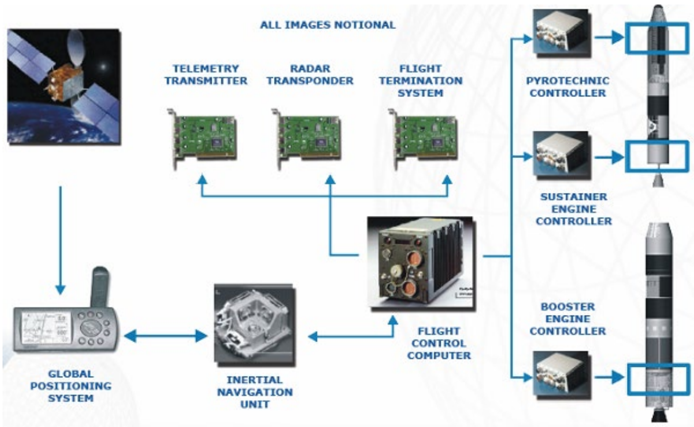


Figure 60: Vega avionics module lay-out (left, courtesy ESA) and Ares 1 Rocket Flight Controls (courtesy NASA)

The Ares 1 avionics system consists of
- **Flight control computers** that provide electrical signals for controlling the rocket stage systems and hydraulic actuators.
- **Instrumentation** for monitoring (like an inertial navigation unit) and controlling the rocket (stage) systems
- **Telemetry, and radio frequency subsystems** for transmission of flight measurement data

Most of the avionics are integrated on the most upper stage or are integrated in a separate vehicle bay on top of the rocket. Some parts, consider e.g. Ares 1, can be found on lower stages. The reason for putting most avionics on top is that it limits duplication.

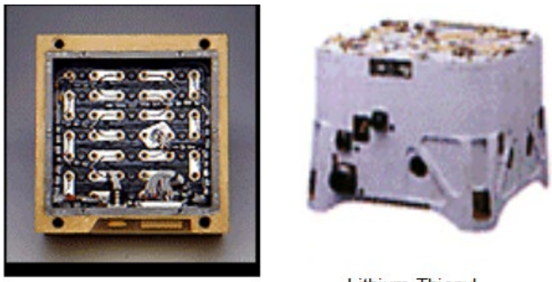
For reliability reasons, most times the avionics system is designed as a duplex system, meaning that most of the system is available in two fold.

6.7 Electrical power and distribution subsystem

Most if not all SLV require an electrical power and distribution system to provide electrical energy to the avionics, to initiate separation, to power actuators, to power pumps, etc. Some SLV even provide for electrical power to the payload carried on board. The electrical power and distribution system essentially consists of an energy source, a distribution system (the electrical leads), a conditioning system, and a power control system (the switches and fuses). The discussion below mainly focusses on the energy source used.

Most SLV use primary batteries as the energy source. Some SLV though use other means, like fuel cells. This is for instance the case for the Space Shuttle. Here we will discuss primary (i.e. non-rechargeable) batteries only. Fuel cells and secondary batteries have been discussed earlier when discussing spacecraft design.

To start with, we define a **primary battery a**s a battery that is designed to be used once after which it is discarded. It is not recharged with electricity and reused like a secondary cell (rechargeable battery) as used on board of most spacecraft. Primary batteries have as advantage that they have a very low self-discharge rate and are more energy efficient than rechargeable batteries, meaning that primary batteries can have lower mass and volume than rechargeable batteries. Like secondary batteries also primary batteries consists of a number of cells put in series. The cells are generally put in a case and sealed with a cover. Some typical batteries for SLV applications are shown in Figure 61.



Silver-Zinc

Lithium Thionyl Chloride

Figure 61: Batteries for launch vehicle applications (courtesy SAFT)

The performance of a battery is usually expressed in the amount of energy it can store and in the amount of power it can deliver. For dimensioning, the amount of energy is most important.

The total energy (usually expressed in Watt-hour, Wh) to be delivered by a primary battery follows from:

$$E_{BAT} = P_{BAT} \cdot t_{discharge} \quad [42]$$

Mass and size (volume) of the battery system can then be determined using two earlier introduced relations (see spacecraft design):

$$M_{BAT} = \frac{E_{BAT}}{(E_{sp})_{BAT}} \quad [43]$$

$$V_{BAT} = \frac{E_{BAT}}{(E_{\delta})_{BAT}} \quad [44]$$

Here $(E_{sp})_{BAT}$ is specific power of battery (energy/unit of battery mass) and $(E_{\delta})_{BAT}$ is power density of battery (expressed in units of power per unit of battery volume). Spacecraft batteries must have acceptable volumetric (Wh/l) and specific energy (Wh/kg) at a useable depth of discharge (DOD) and also good cycle life. Typical values for specific energy and energy density of space grade primary batteries as well as the values applying to a single cell are shown in the following table.

Table 11: Primary batteries; Sample battery characteristics and performance [Surampudi]

PRIMARY BATTERIES

Sample Battery Characteristics and Performance

Type	Cell Parameters and Battery Parameters by Mission Application	Nominal Voltage	Specific Energy, Wh/kg	Energy Density, Wh/l	Specific Power, W/kg	Operating Temp. Range, °C	Capacity Loss % Per Year	Mission Life (yrs)
Ag-Zn	Cell	1.61	200	550	1100	0 to +55	60	1
	Typical Launch Vehicle	28	119	280	120	5 to +40	60	1
$Li\text{-}SO_2$	Cell	2.9	238	375	680	-40 to +70	<1	
	Galileo Probe Battery	38	91	145	260	-15 to +60	<1	9
	Genesis Battery	24	142	125	400	-20 to +30	<1	6
	MER	30	136	**390**	390	0 to +60	<1	4
	Stardust	20	192	182	519	-26 to +50	<1	10
$Li\text{-}SOCl_2$	Cell	3.2	390	875	140	-30 to- 60	<2.5	
	Sojourner	9	245	515	100	-20 to 30	<2.5	4
	Deep Impact	33	221	380	105	-20 to +30	<2.5	4
	DS-2	14	128	340	65	-80 to +30	<2.5	4
	Centaur Launch batteries	30	200	515	85	-20 to +30	<2.5	6
Li- BCX	Cell	3.4	414	930	150	-40 to +70	<2	
	Astronaut Equipment	6	185	210	115	-40 to +72	<2	3
$Li\text{-}CF_x$	Cell	2.6	614	1050	15	-20 to 60	<1	
	Range Safety battery	39	167	150	15	-20 to 60	<1	

Ag Zn=Silver Zinc, $Li\text{-}SO_2$=Lithium Sulfur Dioxide, $Li\text{-}SOCl_2$=Lithium Thionyl Chloride, Li-BCX=Lithium Bromide Complex, $Li\text{-}CF_x$=Lithium Carbon Monofluoride

From: R Surampudi, R Bugga, MC Smart, SR Narayanan HA Frank and G Halpert, Overview of Energy Storage Technologies for Space Applications, Jet Propulsion Laboratory, Pasadena, CA 91109

From the table we learn about typical values for specific energy and energy density for both battery and cell. Typically the values for the individual cells are in excess of those of the full battery. This is because the latter not only consists of the individual cells, but also includes a battery case, connecting leads, etc. Some information is provided on mission life. For launch vehicles, mission life is usually limited to minutes up to maybe 8-12 hr rather than years. More details on the functioning of batteries is provided in a later (2nd yr course) course entitled "Power and Propulsion".

6.8 Other subsystems

Separation systems

Parts of a launcher are separated during flight to jettison amongst other stages and the payload fairing. The criticality of such systems can be illustrated by that a large number of failures are caused by the separation system, see annex B.

A typical separation mechanism consists of a hold down and release system and the separation system itself. The hold down system ensures proper attachment to the rocket, whereas the release system ensures that the elements to be jettisoned are released for jettisoning. Finally a separation system separates the various parts.

As an example, the Delta II rocket stage 1/2 separation system, which is integrated with the inter-stage between stages 1 and 2 initiates separation by explosive bolt detonation 8 s after stage 1 burnout; six spring-driven separation rods at the forward end then separate the stages and stage 2 ignition occurs 5 s later. Such a separation system is schematically drawn in the next figure.

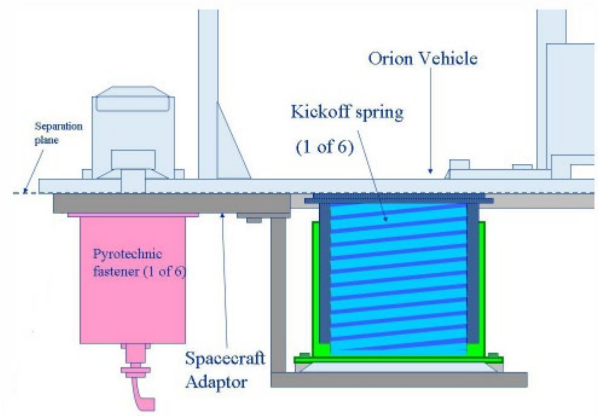


Figure 62: Separation system schematic (courtesy NASA)

Figure 63 shows the separation mechanisms as found on the Saturn I rocket. Here use is made of a linear shaped charge placed around the circumference, to initiate separation. Next retrorockets are used to separate the stages.

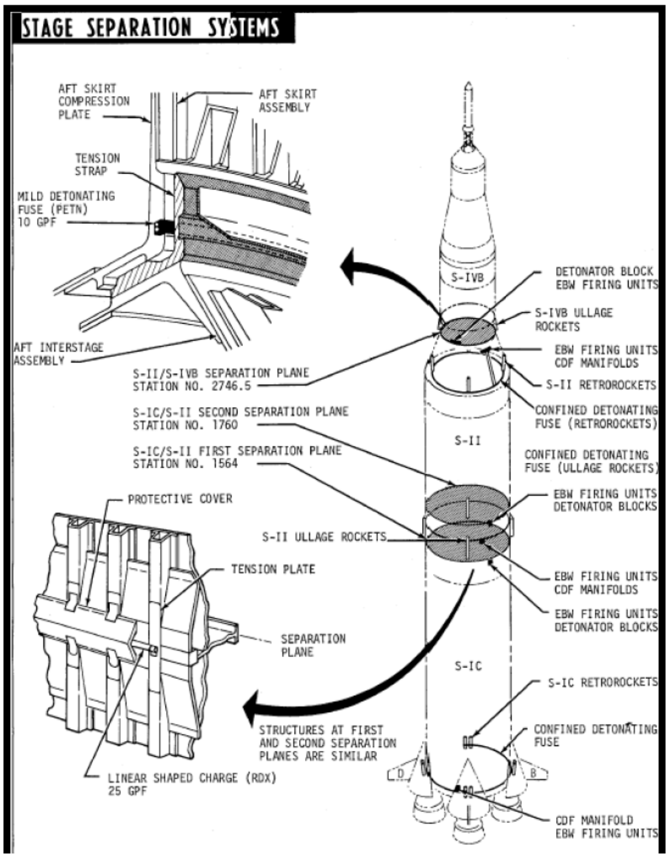


Figure 63: Saturn I rocket separation systems (courtesy NASA)

Separation systems are considered quite determining for the success of an SLV, see e.g. appendix B and Figure 64. The latter summarizes the record of several launch vehicle programs in overcoming the risks specific to staging. All of these vehicles have suffered mission failures due to separation system problems at some point in their history. Of all launch vehicle failures in the U.S. between 1983 and 1998, 5 out of 22 failures are attributable to separation systems. The Ariane launch vehicle has the lowest staging-related failure rate of all those shown in the Figure. Ariane uses an explosive cord severance device, retro-rockets on the lower stage and acceleration rockets on upper stage.

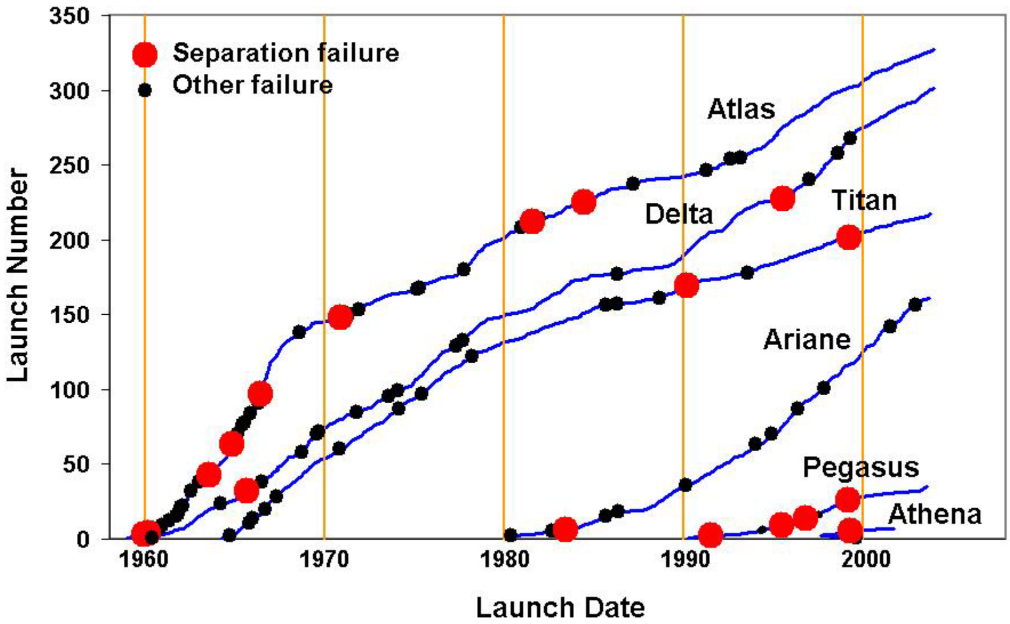


Figure 64: Overview of stage separation failures [Isakowitz]

Recovery system

Some rocket stages are recovered for re-use. This is particularly the case for the Space Shuttle SRB., see Figure 28. Figure shows SRB with at the top parachutes that allow for a soft landing after which the boosters are retrieved. Also for the Ariane 5 SRB some studies have been performed to investigate recovery (and reusability). However, production of such systems remains limited.

Landing system

Current rockets are almost all expendable systems. However, in case one or more stages can be reused, this potentially offers reduced launch cost. A first example of an SLV with a landing system is the Space Shuttle Transportation System of which the orbiter vehicle is equipped with wings and aerodynamic controls to provide it with the necessary cross- and down-range to reach a potential landing site, control software, a landing gear that allows for a soft landing and propellant to allow for control in the higher layers of the atmosphere. More recently (February 11, 2015), SpaceX has made a successful attempt to land a rocket stage vertically on a floating barge after two unsuccessful earlier attempts. Vertical landing, see also figure, is considered an old technology as it was already demonstrated successfully for the Apollo lunar module and it has been flight tested on a number of flight test vehicles. A typical mass distribution of a piloted test vehicle is shown in Table 12.

Figure 65: Example of vertical landing test vehicle [Inatani]

Table 12: Physical characteristics of piloted vertical landing test vehicle [Bellman]

Weights, kilograms	
Primary structure including engine gimbal ring	227
Landing gear (plus 17.2 kg when casters replace the pads)	64
Manual controls	20
Control avionics and wiring	73
Jet-engine system including hydraulics	363
Rocket system	168
Flight instruments, radar sensors, wiring, and console	52
Electrical system	52
Ejection-seat parachute and breathing oxygen	61
Drogue parachute and attachments	9
Research instrumentation and telemetry system	48
Communications	2
Normal empty weight	1139
Useful load at takeoff:	
Pilot	84
JP4 fuel (195 kg less 6 min of idle at 3.6 kg/min)	173
Hydrogen peroxide (305 kg less 27.2 kg for preflight checks)	285
Helium gas	2
Engine oil	4
Total useful load	548
Maximum takeoff weight	1687

Table shows that the landing system makes up roughly 5% of the Maximum Take-off Weight (read mass) or about 8% of the normal Empty Weight (mass). The latter is considered more representative for the design of the landing gear as when landing, the vehicle mass will be closer to the empty mass than to the takeoff mass.

7 Analyzing for mass, reliability, etc.

In the foregoing sections, we have focused on discussing the various subsystems, their main functions and on how well they perform. Next to a functional design, we also need to be able to design the vehicle so that it will fulfill all constraints with respect to mass, cost, reliability, size, etc. In this chapter, we will introduce some simple models that allow for estimating mass, size, reliability, and cost of multi-staged vehicles. Some thoughts are also presented on estimating still other parameters of importance for the operations of an SLV, see Figure 18, as well as the use of more complex/detailed models that generally allow for a higher estimation accuracy.

7.1 Simple launch vehicle mass estimation model

For rocket sizing purposes, it is important to have some methods that allow for early dry mass estimation of the rocket. One approach is to use information on propellant mass and structural efficiency (ratio of structural mass-to-propellant mass ratio) or propellant mass and stage dry mass to develop analogous or parametric mass estimation models for rocket stages. To this, we then only need to add the mass of the fairing as well as the mass of the Vehicle Equipment Bay (VEB), i.e. the section that carries most of the SLV avionics, electrical power system, and RCS. The mass model presented below is a first such model.

A Simple Launch vehicle Mass Model

Inputs for this model include:

1. Mass of propellant ($M_{Propellant}$)
2. Vehicle dry mass (M_{Dry})
3. Surface area of fairing

Solid Rocket stage (N_{data} = 13):

$M_{stage\ dry}$ (kg) = 0.1554 $M_{propellant}$ (kg)
Valid in propellant mass range 2 – 500 ton
R^2 = 0.9857
RSE = 22.1%

Cryogenic (liquid oxygen and liquid hydrogen) rocket stage dry mass (N_{data} = 29)

$M_{stage\ dry}$ [ton] = 0.1011 $M_{propellant}$ [ton] + 1.201
Valid in propellant mass range 8 – 985 ton
R^2 = 0.9914
RSE = 26%

Vehicle Equipment Bay (VEB; it is essentially a bay that holds the SLV avionics, electrical power and RCS)

M_{VEB} [kg] = 0.345 (M_{dry} [kg])$^{0.703}$
R^2 = 0.9381
RSE = 72%

Fairing mass

$$M_{fairing}[kg] = a \cdot S_{Fairing}[m^2]$$

a = 10.3 kg/m², SSD = 2.3 kg/m²

The above model can be found in appendix E, next to some slightly more detailed models. A key issue here is that all these models are accurate up to some degree. Key is to develop simple, but reasonably accurate methods by using data from similar vehicles. Additionally, also a key factor is for the designer to determine the accuracy of the model outputs by validating the model outputs through comparing these outputs with actual results. Once the accuracy is known/established, the designer shall select proper margins that will allow the design to develop in a stable way in future phases of the project.

7.2 Simple rocket stage sizing model

To size rocket stages in a simple way, we will use data available from historic launch vehicles to determine the size of the "to be designed rocket stage". In first principle, it is like we size a rocket stage using data from a comparable vehicle. In a more practical way, we use data of a range of vehicles to come up with estimation relationships that allow us to determine the stage size. The approach followed is a two-step-approach, wherein we use the loaded mass of the stage as input and assume that all stages are of a cylindrical shape characterized by the height and diameter of a cylindrical envelope, i.e. tightest fit of a cylinder, of the stage, see Figure 66.

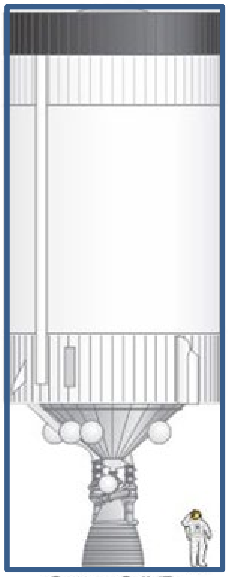


Figure 66: Rocket stage envelope (see cylindrical box encompassing the stage)

The two steps are:

1. Based on the given loaded mass, the model provides us with a stage mass density (ρ) from which the stage volume V is calculated using:

$$V = M_{loaded}/\rho \qquad [45]$$

2. Again using the given loaded mass, the model provides us with a stage height-to-diameter (slenderness) ratio (λ_{sl}), which in conjunction with the volume allows us to estimate both stage diameter and height using $V = \pi/4 \ D^2 x \ H$:

$$D = \sqrt{\frac{4}{\pi}\frac{V}{H}} = \sqrt{\frac{4}{\pi}\frac{V}{\lambda_{sl} \ x \ D}} \qquad [46]$$

and:

$$H = \lambda_{sl} \ x \ D \qquad [47]$$

Above model requires that we are able to estimate stage mass density (ρ) and height-to-diameter ratio (λ_{sl}). These parameters (in a first approach) can be estimated from historical data, be it that the accuracy may be somewhat limited. Typical historical data are given in appendix F.

Based on the data collected in appendix F, average values of mass density and slenderness ratio and their sample standard deviation about the average can be determined. As an example, next table gives average stage mass density and (sample) standard deviation data for various different stage types (based on propellant used) for vertical launch, expandable, rocket launch vehicles.

Table 13: Stage mass density values [kg/m^3] for varous stage types (based on propellant type used)

	# of points	Mean	SSD	Min value	Max value
	[-]	[kg/m^3]	[kg/m^3]	[kg/m^3]	[kg/m^3]
Hydrolox (all)	17	221.9	50.3	108.4	326.4
Hydrolox lower stages	5	251.2	12.0	242.1	268.4
Hydrolox upper stages	12	203.5	44.4	108.4	247.6
Kerolox (all)	15	646.9	184.6	262.7	834.1
Kerolox lower stages	11	708.9	136.4	394.2	834.1
Kerolox upper stages	4	476.4	210.3	262.7	665.1
Storable (all)	18	608.2	236.6	207.2	958.2
Storable lower stages	13	700.3	204.7	249.0	958.2
Storable upper stages	5	381.7	181.6	207.2	643.4
Solid	18	1074.6	182.2	631.5	1352.8
Solid lower	14	1139.1	133.1	931.1	1352.8
Solid upper	4	849.0	155.5	631.5	1001.0

The table shows average (or mean) mass density, standard deviation (SSD) on average mass density as well as the maximum and minimum mass density value found for the historical stages which have been used in the analysis. The table also provides the number of data points used. As indicated earlier, a distinction is made based upon the propellant combination. This is because the propellant determines the mass density greatly. The table also distinguishes between lower and upper stages, where the term upper stage refers to the stage that burns out last. All other stages are referred to as lower stages. This distinction is made because the upper stage usually caries most avionics and power equipment and the RCS whereas all other stages only carry part of the avionics.

Similarly also average values (and SSD) for the slenderness ratio of the stages can be determined. Data clearly shows that lower stages are usually more slender than upper stages. This is because for upper stages the diameter is largely determined by the diameter of the lower stages be it that sometimes we do see a reduction in stage diameter for higher stages.

Next to averaging also regression analysis can be applied to determine mass density and stage slenderness values in relation to stage loaded mass. This is left for the reader to study for him/herself.

7.3 Simple launch vehicle reliability estimation model

In chapter 5, we have shown that by staging, the payload mass into orbit can be increased. However, also the vehicle reliability decreases whereas cost increases. Hence, we should not blindly strive for multiple staging, without considering amongst others vehicle reliability. Hereafter, we will briefly discuss the reliability of a multi-stage launch vehicle considering only catastrophic[26] failures to occur. For a discussion on the basics of reliability, see the appropriate section(s) in the Spacecraft Design and Sizing reader.

[26] Catastrophic failure is any failure that leads to a total loss of mission.

The reliability (R) of a multi-staged rocket where any failure is considered a catastrophic failure can simply be determined from the product of the reliability of its individual elements/stages (R_i):

$$R_{LV} = \prod_{i=1}^{n} R_i \qquad [48]$$

Here i refers to the various elements that constitute an SLV, n to the total number of elements and F is the failure probability. As elements we can consider the individual propulsive stages, but also the navigation system, the stage separation system, the payload separation system, the electrical system and so on, see appendix B. Rewriting R in terms of failure rate (λ), we obtain:

$$R_{LV} = e^{-\lambda_{LV}N} = \prod_{i=1}^{n} e^{-\lambda_i N} \qquad [49]$$

Here λ_{LV} is SLV failure rate in failures per launch, λ_i is failure rate in failures per launch of subsystem i and N is number of launches. For a non-reusable SLV N = 1. Appendix B gives typical failure rates of non-reusable SLVs and their elements. In case failure rates are much smaller than 1, above relation can be written for expendable vehicles as:

$$R_{LV} = 1 - \lambda_{LV} = 1 - \sum_{i=1}^{n} \lambda_i \qquad ; \qquad (\lambda_i \ll 1) \qquad [50]$$

Here terms of second order small are neglected.

It should be stressed that the above model is a very simple model, where it is assumed that all failures are catastrophic failures (complete failures) and that all propulsive stages have to work successfully to accomplish the launcher mission. Still it allows for attaining insight in how the different elements of a rocket determine its reliability.

Example 1: Ariane 5 reliability estimation from stage reliability data

The European Ariane 5 rocket consists of 2 segmented solid booster stages, a cryogenic stage and a non-cryogenic upper stage. Using the data from Appendix B, Table 3 (SPIAG data), and neglecting the contributions of the non-propulsive stages, the following reliability can be estimated for the vehicle:

$$R = R_{booster1} * R_{booster2} * R_{cryogenic\ stage} * R_{upper\ stage}$$

$$R = 0.9925^2 * 0.9664 * 0.9826 = 0.935 \text{ (or 93.5\%)}$$

The results indicate that when launching a large number of Ariane 5 (1000 or more) roughly 6.5% will fail due to a propulsion failure.

In the foregoing example, the reliability of Ariane 5 is estimated based on publicly available reliability data for US systems. The result indicates a fairly low reliability for Ariane 5. Taking into account also the non-propulsive elements, as well as separation events, the predicted reliability of Ariane 5 will be even less. However, US data is not necessarily also valid for the European Ariane 5 as in Europe different technologies are used. Also the data reported may be outdated as of the introduction of new technologies, different production methods etc. Compare for instance the difference in failure rate data of solid rocket stages as follows from SPIAG and Futron (appendix B).

Example 2: Rocket LV reliability determination
Consider a rocket with a first stage consisting of 3 solid rocket stages in parallel, a liquid storable propellant rocket second stage and a liquid rocket cryogenic rocket upper stage. Given are:

- *Solid rocket stage reliability = 0.9946*
- *Liquid storable propellant rocket 1st and 2nd stage reliability = 0.9803 (each)*
- *Cryogenic upper stages fail two times more often than storable propellant stages*

What is the reliability of this rocket?

First we determine the reliability of the first stage. Since the first stage consists of 3 identical stages in parallel of which none is allowed to fail, it follows for the reliability of the first stage (three solids): $R_1 = (0.9946)^3 = 0.9839$

Reliability of second stage: $R_2 = 0.9803$

Reliability of third stage (R_3) *is determined using:*

- *Failure probability of liquid storable propellant rocket stage is 0.0197 (F = 1-R)*
- *Failure probability* (F_3) *of liquid cryogenic stage is 0.0197 *2 = 0.0394*

This leads to a reliability of the upper stage: $R_3 = 1\text{-}F_3 = 1\text{-}0.0394 = 0.9606$

It now follows for the total reliability of this rocket:

$$R = R_1 \times R_2 \times R_3 = 0.9839 \times 0.9803 \times 0.9606 = 0.9265$$

7.4 Simple launch vehicle cost model

Earlier we have discussed the importance of cost for a good design. This would require a cost estimation model to allow for determining the main factors contributing to cost and to provide engineers with the insight in how to control SLV cost. A first cost model could be based on cost data as provided for in appendix A. These data can be used to determine specific launch vehicle cost, here defined as cost per unit of SLV dry mass, in relation to vehicle dry mass. Of course, also more detailed cost models can be used, see e.g. the work of [Koelle] and/or [Drenthe]. This is left for you to explore for your own in your future career.

7.5 Still other models

Models may also be generated to allow for estimation of availability, operability, etc.. This though is left for you to explore for your own in the future.

7.6 More complex models

To estimate the performances as well as a range of other parameters of importance to judge the operations of SLVs also more complex/detailed models can be used. This can be models that allow for a detailed structural model to determine the mass of the vehicle elements, or models that allow for detailed flight trajectory simulations to determine the required delta v, or an extensive aerodynamics simulation to get accurate values of the vehicle's lift and drag, etc., or an extensive cost/reliability analysis. However, early in the design the decisions to be made allow one to use simple models, whereas the more detailed models just waste time on details that can also be worked out later in the design. What to analyse and to what extent is the essential question for any designer and may make a difference between a successful design or not. Starting with simple models allows the (junior) designer to build up the experience needed to distinguish between important and not so important matters (related to the phase the design project is in).

8 Compare, make choice, evaluate

Once the various concepts have been analysed, the results can be compared and traded leading to the selection of a preferred design. Also the results can be evaluated and the work for the next phase in the design can be prepared (including a down flow of the requirements) and prepared.

To allow comparing the design options, we need to characterize the various concepts investigated. This in general means that we need to link the vehicle concept definition, i.e. how it looks like, what important technologies used, and its important dimensions (size), to how well the SLV performs with respect to the functional and operational performances and how well it fulfils the constraints with respect to for instance:

- Vehicle mass
- Vehicle size
- Development cost
- Operations cost
- Vehicle reliability
- Safety measures (to prevent loss of crew and/or damage to facilities, etc.)
- Etc.

Only then, we can perform a trade and select the concept that is going to be worked out in more detail in an ensuing phase. Below, some examples will be discussed.

As a first example we compare in Figure 67 expandable and reusable launch vehicles on a cost basis as a function of payload mass. As a representative figure for the cost, we use the number of man-year (MY) needed to build the rocket. According to the results shown in figure, fully reusable SLVs offer a significant reduction in launch cost. The figure furthermore shows that specific cost reduces with increasing payload mass and increases with increasing distance. The former is due to the economies of scale that a larger vehicle provides (consider e.g. a small ship versus a large ship). The latter is because the further a rocket travels, the lower the payload mass that can be carried onboard.

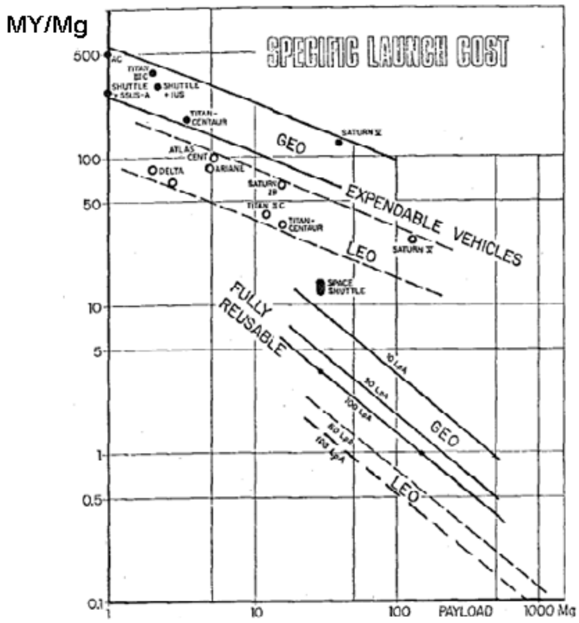


Figure 67: Specific launch cost for expendable vehicles and expected range for fully reusable systems [Koelle]; MY is Man-Year; Mg is 1 ton mass.

From the figure, we might conclude that we need to develop fully reusable SLVs. Still, the Space Shuttle did not quite live up to expectations, hence demonstrating that models used in the design should be validated, meaning that the designer should check or prove the accuracy of the models used. In the end, this means you need to compare the outcomes of the design (the design performances and the design characteristics) with the actual results.

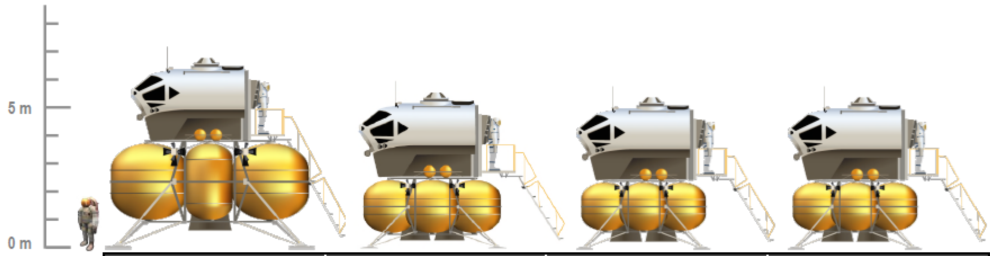
As a second example we provide Table 14, wherein some important characteristics of three expendable SLVs are summarized. They all are capable of launching a payload mass of 7000-8600 kg in to a low Earth orbit, be it that orbital inclination and altitude varies slightly. Now ask yourself the question which SLV you would select in case a payload mass of 6900 kg has to be orbited in a 185 km Low Earth Orbit (you may neglect any difference in orbital inclination)?

Table 14: Comparison of vehicle performance for a 185 km LEO mission [adapted from Francis]

Vehicle	Country	Length (m)	GLOW (kg)	Payload (kg)	Launch cost	Cost per kg ($/kg)
Atlas IIAS	USA	47.5	234,000	8610	105	12,195
Ariane IV (44L)	Europe	58.5	470,000	7700	100	12,987
Soyuz	Russia	50.6	310,000	7000	35	5,000

Maybe you would select the Soyuz SLV as it is by far cheapest. However, reasons for selecting are not always purely cost based. It may also be based on political reasons (we buy American good only, national security; we do not want to rely on foreigners). Other reasons may be w.r.t. for instance reliability and availability or the possibility to launch two spacecraft (with total mass of 7000 kg) at the same time as is possible when using (Ariane IV).

A third example is about the effect of changing the propellants on the mass and size for a lunar lander design carrying a Space Exploration Vehicle (SEV) on top (constant mission delta-v). SEV mass is 4.0 t. The results show that lowest total mass is obtained for the vehicle using LOX/LH_2 propellant. This is associated with the higher specific impulse that can be obtained for this propellant combination as compared to the others. Even though total mass is lowest, still, the vehicle using the LOX/LH_2 combination is largest. This is because of the much lower overall mass density of the propellant.



	LOX/LH2	LOX/CH4	LOX/RP	Non-toxic Storable
Lander				
Dry Mass (with SEV)	7.0 t	6.2 t	5.9 t	5.8 t
Propellant Mass	13.4 t	16.5 t	17.5 t	18.4 t
Propellant Mass Fraction	66%	73%	75%	76%
Stage Height (on surface)	6.3 m	5.5 m	5.2 m	5.2 m
Stage Diameter	7.8 m	5.8 m	5.1 m	5.1 m

SpaceWorks

Figure 68: Results of propellant trade study [Spaceworks]

Again, to make an honest trade, we also may need to consider also cost, reliability, durability, etc. of each of the vehicles. Additionally, it may be interesting to not only look at the vehicle cost, but also at the cost of the mission, as a larger and heavier spacecraft may necessitate a larger and more costly launcher.

Example four is again about a propellant trade, but this time for an advanced launch vehicle with two reusable boosters, see figure, as taken from the work of [Burkhardt].

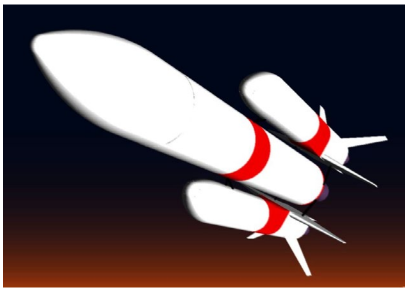


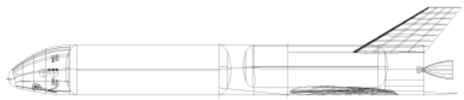
Figure 69: Advanced SLV with two reusable boosters [Burkhardt]

Burkhardt aimed at trading the benefits of using Methane-LOX versus Kerosene-LOX as propellant for the reusable booster stage. The specific impulse of a Methane-LOX motor is about 10 s higher than for a comparable Kerosene-LOX engine. It was considered that this would allow for a reduction in propellant mass and hence launch mass. However, the comparison of the performance of both propellant combinations for a complete vehicle revealed that the advantage of the higher energetic content of methane was counterbalanced by an increased motor mass and an increased booster size, hence higher aerodynamic drag and increased mass. The payload performances of the reusable kerosene and methane booster are therefore almost identical with some edge for kerosene. In view of the increased size and dry mass of a reusable methane booster stage, one can expect a cost disadvantage for CH_4 from an SLV system level point of view.

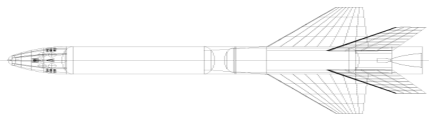
	X11	X14
Booster propellant	Kerosene	Methane
Length [m]	33.96	34.14
Fuselage Diameter [m]	3,80	3.95
Wing Span [m]	15.00	15.36
Booster mass empty* [kg]	31 914	35 209
Ascent Propellant [kg]	200 000	200 000
Residual Propellant [kg]	2 295	2 263
Reserve Propellant [kg]	1 800	1 800
Fly-Back Fuel [kg]	7 800	8 700
Booster Structural Index [-]	0.1506	0.1655
Booster GLOW [kg]	243 809	247 972
LV GLOW [kg]	740 754	749 068
Take-off T/W [N/kg]	12.06	12.01
GTO Payload [kg]	**11 667**	**11 656**
Separation Altitude [km]	58.5	60.2
Separation Mach number [-]	5.74	5.86

* incl. margins

Principal Launch Vehicle Characteristics



Reusable Kerosene Booster Design X11



Reusable Methane Booster Design X14

Figure 70: Comparison of Kerosene versus Methane booster [Burkhardt]

The next example is of a propellant trades for a future A5 RCS system. The current Ariane 5 RCS system is based on six 400 N thrusters that each use hydrazine as monopropellant. For a normal mission a total impulse of 150 kNs (all thrusters together) is given using about 75 kg propellant stored in two tanks pressurized with nitrogen gas. Next to the thrusters and the tanks, the system consists of a range of valves, sensors, tubing and structure elements. The dry mass is about 59 kg. Each tank has a volume of 58 litres and all together the system volume comprises about 155 litres. Replacement systems have been studied based on a range of propellants. Resulting designs are compared on a mass and volume basis in Figure 71. Interesting is that none of the options both has lowest mass and volume. So mass and volume should be traded against each other, meaning we have to consider how important mass and volume are for the design.

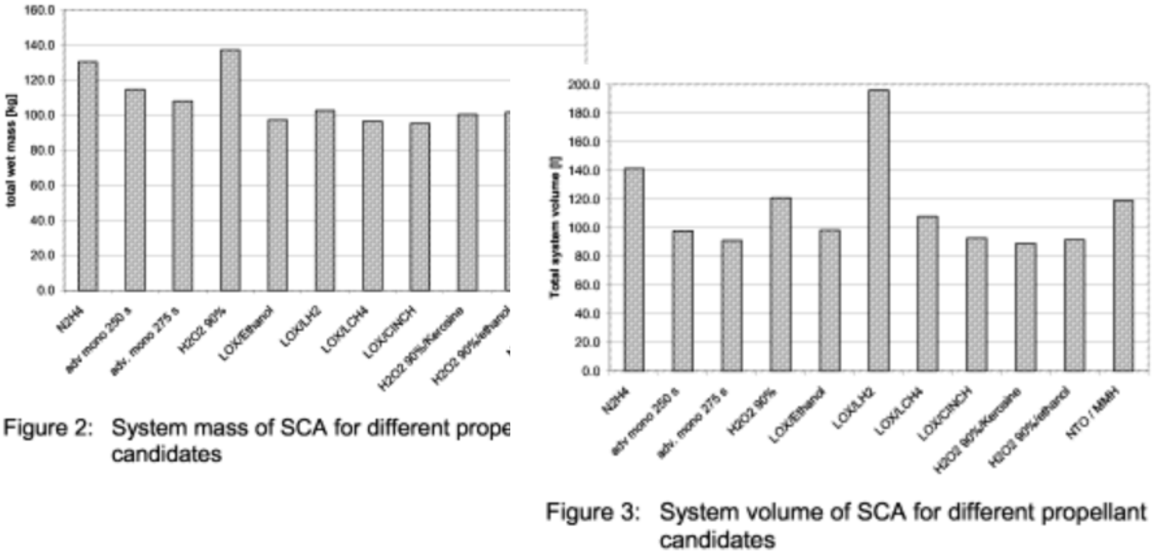


Figure 71: System mass and volume of SCA for different propellant candidates [Haeseler]

The example demonstrates that trades are not limited to vehicle level studies, but can also be performed at subsystem level etc. It also shows that trading of different characteristics is needed.

Sixth and final example is a solid/liquid system reliability trade. Appendix B, Table 3 shows information on the success rate of different technologies that can be used for SLV concepts. Clearly these data show that solid chemical propulsion is more favourable than liquid chemical propulsion. However, most current designs use liquid propulsion technology. So a legitimate question is why? As a possible reason is offered that the added risk is considered less important than payload mass into orbit, which in general is higher for liquid propelled systems as of the higher specific impulse.

Summarizing, we may conclude that many different trades are possible at all design levels. Note that with trades, we mean that we are assessing the effect of different design choices on performances of the design and or whether the constraints can be met. Design choices may include such diverse issues as:

- Reusable versus expendable
- Single stage versus multi-stage
- Air-launch versus ground- and/or sea-launch'
- Use of different structural materials
- Steering (gimbaled thrust versus verniers or using fixed thrusters about multiple axis, etc.)
- Different propellants (solid versus liquid, etc)
- Etc.

To allow for trading we need to be able to analyze all designs on the trade parameters, i.e. the parameters that are traded upon. These parameters include payload mass into orbit, payload size, launch cost, safety, availability, feasibility, etc. Some of these parameters may be easy to quantify and some are not. How to deal with the latter type of parameters will be dealt with in detail in later lectures on systems engineering.

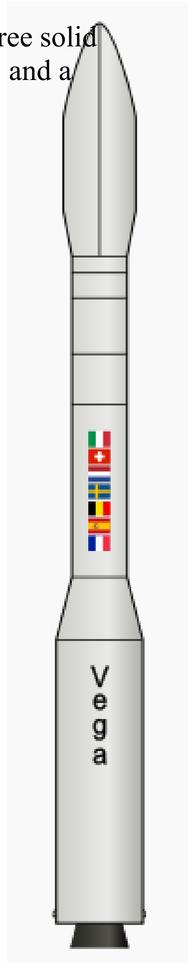
Some further examples of launch vehicle trade studies can be found in:

- **Performance Comparison of Reusable Launch Vehicles, by** Mark Ayre, Tom Bowling, Cranfield University, Bedfordshire, MK43 OAL.
- **Cost Comparison of Expendable, Hybrid, and reusable Launch Vehicles,** THESIS, Greg J. Gstattenbauer, Second Lieutenant, *Air Force Institute of Technology*
- **Launch Vehicles Then and Now: 50 Years of Evolution,** http://www.aerospace.org/2013/12/11/launch-vehicles-then-and-now-50-years-of-evolution/. Last retrieved 31 May 2015.

Problems

Problem 1: Vega rocket (mass fractions and total velocity change)

The Vega rocket, see figure, is a single-body launcher (no strap-on boosters) with three solid rocket stages, the P80 first stage, the Zefiro 23 second stage, the Zefiro 9 third stage, and a liquid rocket upper module called AVUM.



Vega rocket

Some typical data are provided in the next table.

	Stage 1	Stage 2	Stage 3	Stage 4	Fairing	Payload
Mass at lift off [ton]	95.796	25.751	10.948	0.968	0.49	1.50
Propellant mass [ton]	88.365	23.906	10.115	0.55		
Effective exhaust velocity [m/s]	2550	2834	2893	3094		

You are asked to determine for this rocket launcher:

1. Payload, propellant and structures mass fraction for each of the 4 sub-rockets
2. The total ΔV delivered by the rocket given that fairing separation takes place at burn-out of the 2^{nd} stage.

Check the Launch Vehicle Catalogue (available on Brightspace) for when the fairing is jettisoned.

Problem 2: Two stage rocket design

Suppose we are designing a two-stage[27] rocket that should be capable of delivering a 5 ton payload a ΔV of 9.8 km/s. For this rocket we select a propulsion system capable of delivering an effective exhaust velocity of 3750 m/s (constant with altitude). Taking a constant structural mass ratio of 0.1, determine the minimum launch mass needed to bring this payload into orbit. Hint, you should vary the ΔV distribution over the two stages to find out which distribution gives lowest total mass.

---

[27] Single stage rocket:
For a single stage launcher, we find a mass ratio of 13.64.

$13.64 = M_o/M_e$
$M_o - M_e = M_F$
$M_e = M_s + M_p$

$M_e/M_o = M_s/M_o + M_p/M_o$

Filling in numbers gives: $1/13.64 = 0.1 + 5000/M_o \rightarrow M_o =$ No solution possible (only when structural mass ratio is much lower a solution might be possible. Take 5%, we get $5000/M_o = 0.0233 \rightarrow M_o =$ 214.5 ton

Problem 3: Mass fractions of parallel-staged rocket

The Ariane 5 launcher, see figure, is the main work horse of the European Space Agency (ESA). It essentially consists of a core stage (EPC) with two boosters (EAP) attached. On top of the core stage there is an upper stage (EPS), which carries the actual payload, a vehicle equipment bay (VEB, which carries essential equipment to power and control the rocket) and a fairing.

From the Launch Vehicle Catalogue [LVC] we obtain the following data for this launcher.

STAGE	0	1	2
Designation	EAP	EPC	EPS = L9
Manufacturer	EADS ST	EADS ST	EADS ST
Length (m)	31.24	30.7	3.356
Diameter (m)	3.05	5.4	3.936
Dry mass (t)	37 x 2	12.23	1.25
Propellant:			
➤ Type	Solid	Cryogenic	Liquid (storable)
➤ Fuel	PBHT	Hydrogen	MMH
➤ Oxidizer		Oxygen	N_2O_4
Propellant mass (t):	237.8 x 2	157.3	9.7
➤			3.2
➤			6.5
➤			-
Ta (ba..)			MMH: 19 N_2O_4: 19
Total lift-off mass (t)	274.8 x 2	170.2	11

Additionally we give:

- Payload mass into GTO: 6640 kg
- Fairing mass: 1970 kg
- VEB mass: 1300 kg

This rocket essentially consists of three sub-rockets[28]:
1: 1st[29] sub-rocket with the 2 boosters and the main stage actively providing thrust
2: 2nd sub-rocket after separation of the two boosters (EAP) with only the main stage firing
3: 3rd sub-rocket after burn-out and separation of the first stage

You are asked to determine for this rocket payload mass fraction, propellant mass fraction and structural mass fraction of the first and second sub-rocket.

Problem 4: Effective exhaust velocity of parallel staged vehicle stage

1st sub-rocket of European Ariane 5 rocket consists of a central core stage with a thrust of 1MN and two parallel booster stages each with a thrust of 5.4MN. Effective exhaust velocity of core stage is 4000 m/s and for each of the two booster stages 3000m/s. Core and booster stages both are operative (producing thrust) at lift-off. It is assumed that thrust of both the booster stages and the core stage does not vary with altitude. The rocket as a total with core and booster stages active is referred to as first sub-rocket. Determine for this rocket the average effective exhaust velocity of this first sub-rocket.

[28] In the following, we assume that the fairing is carried all the way in space. In reality, this is not the case.
[29] Note that because the booster stages together are referred to as the 0th stage, the first sub-rocket sometimes is referred to as the 0th sub-rocket and so on.

Design problems related to Space Shuttle

The next few problems are about the Space Shuttle. The Space Shuttle is considered a space launcher, but it also shows many of the features of an aircraft. To some extent it is even more of an aircraft than a conventional rocket. To show that many of the methods used in aircraft design can also be used to design (highly advanced) space launchers, we have included a number of problems focusing on the Space Shuttle. Of course also some issues are tackled that are very rare or uncommon for ordinary aircraft.

About the Space Shuttle (background information)[30]

The US Space Shuttle consists of a rust-colored external tank (ET), two white, slender Solid Rocket Boosters (SRBs), and the orbiter, a winged spaceplane which is the Space Shuttle in the narrowest sense.

A typical mission profile is shown in the next figure:

At lift off all engines are operating providing a total thrust of about 30 MN (MegaNewton; 1 MN is 1 million Newton). After 124 seconds the SRBs burn out and separate. After that time only the main engines of the Orbiter keep on functioning until main engine cut off (MECO).

[30] All pictures related to Space Shuttle are by courtesy of NASA.

Final orbit insertion is achieved by another propulsion system contained on board of the Space Shuttle. Some technical data are provided hereafter.

Technical data

Orbiter specifications (for *Endeavour*, OV-105)

- Length: 37.237 m
- Wingspan: 23.79 m
- Height: 17.86 m
- Empty mass: 78,000 kg
- Gross liftoff mass: 110,000 kg
- Maximum landing mass: 100,000 kg
- Main engines: Three, each with a sea level thrust of 1.752 MN.
- Maximum payload: 25,060 kg
- Payload bay dimensions: 4.6 m × 18 m
- Operational altitude: (190 to 960 km)
- Speed: 7,743 m/s (27,870 km/h)

External tank specifications

- Length: 46.9 m
- Diameter: 8.4 m
- Propellant volume: 2,025 m^3
- Empty mass: 26,535 kg
- Gross liftoff weight: 756,000 kg

Solid Rocket Booster specifications

- Length: 45.6 m
- Diameter: 3.7 m
- Empty mass (per booster): 63,272 kg
- Gross liftoff mass (per booster): 590,000 kg
- Thrust (per booster, sea level, liftoff): 12.5 MN
- Burn time: 124 sec.

System Stack specifications

- Height: 56 m
- Gross liftoff mass: 2,000,000 kg
- Total liftoff thrust: 30.16 MN

Problem 5: ΔV delivered

Estimate for the Space Shuttle the total ΔV delivered until Main Engine Cut Off (MECO) using ideal rocket equation. You may assume a constant thrust level for both the SRBs and the main engines equal to the values defined in the foregoing for sea level conditions. In addition, you may use for the main engines a constant specific impulse of 455 sec.

Problem 6: Approximate ET tank and SRB size

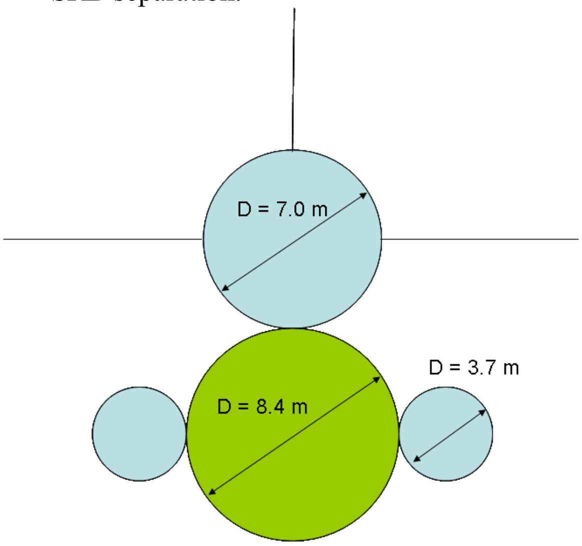
Given an oxidizer fuel mass ratio of the main engines of 6.0, a liquid hydrogen mass density of 76 kg/m^3 and an oxidizer mass density of 1140 kg/m^3, determine the propellant volume as well as the length of a cylindrical tank holding the volume (assume straight end caps). Diameter may be taken equal to the diameter indicated in the figure. Discuss if and why you expect the calculated length to differ from the real length.

Problem 7: Location of CoM

Consider the space shuttle. There are 5 rocket nozzles working during liftoff. The resultant thrust vector has to pass thru the shuttle's CoM or the vehicle would start to rotate.

a) Given that the SRBs, ET and orbiter can be approximated as cylindrical bodies with homogeneous mass distribution compute the location of the CoM at lift-off in the plane shown in the figure relative to a body axis system with its origin in the geometric centre of the ET with y-axis being the vertical axis pointing up and z-axis in the horizontal plane with positive direction to the right.

b) Discuss (do not calculate) the shift in CoM that occurs during the flight up to the point of SRB separation.



Problem 8: Location of CoP

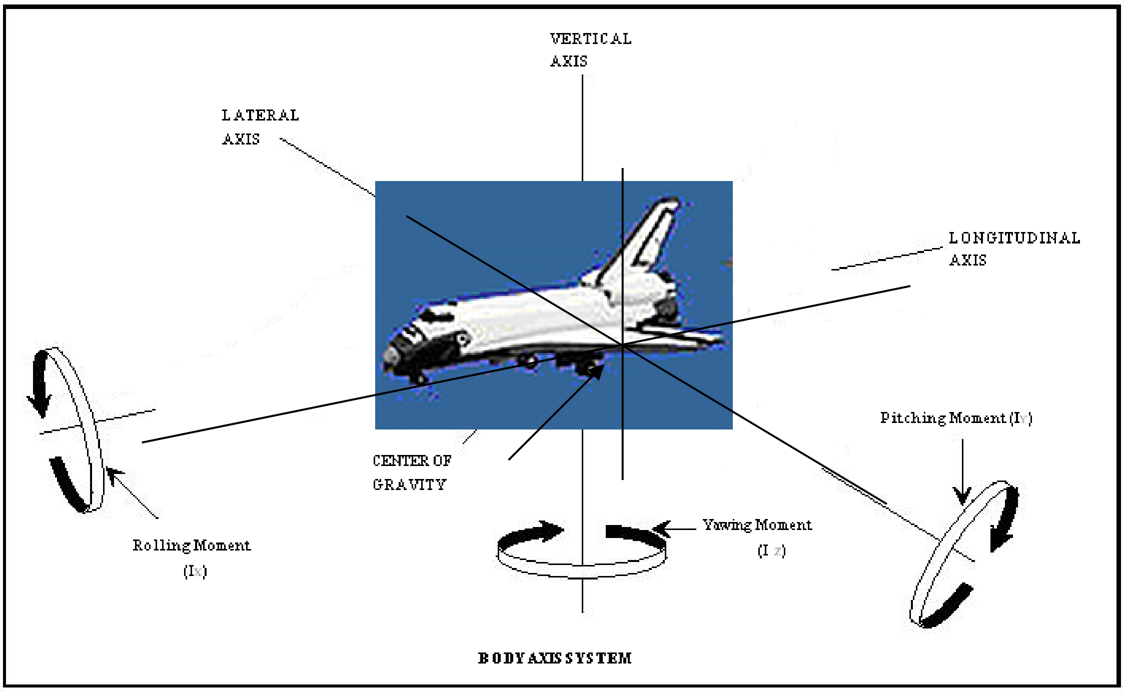
Given that the drag coefficient of the SRBs, ET and Orbiter based on the cross-sectional area is 0.2, 0.225 and 0.3 respectively and that interference effects can be neglected, you are asked to calculate the location of the CoP of the complete vehicle in the plane shown in the figure.

Problem 9: MMOI

Calculate the MMOI of the Space Shuttle Orbiter, SRBs and ET about the bodies principal axis in longitudinal direction at lift-off assuming that SRB, ET and Orbiter can be approximated as long cylinders with a cylinder length equal to the length indicated in the foregoing data table and given that the CoM is placed on the centre line of the ET 0.414 m above the line connecting the centre of ET and the two SRBs.

Problem 10: Control torque

During the ascent flight the Shuttle Orbiter has to rotate about its longitudinal (roll) and lateral (pitch) axis, see figure for axis definitions.



Values of MMOI of the Orbiter are given in the next table.

Dimensions, mass and moments of inertia of the Shuttle Orbiter.

Dimensions	$x = 37.24$ m
	$y = 23.79$ m
	$z = 17.25$ m
Mass[a]	$m = 90\,700$–$104\,330$ kg
Principal moments	$I_{xx} \approx 1\,310\,000$ kg m^2
of inertia[b]	$I_{yy} \approx 10\,220\,000$ kg m^2
	$I_{zz} \approx 10\,650\,000$ kg m^2

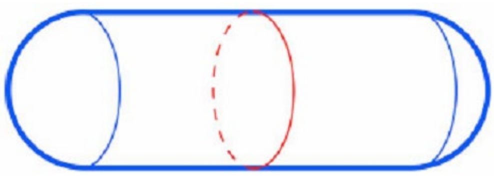
[a] At landing, depending on mission. The rendezvous mass will be nearer to the larger value.
[b] Typical values, depending on mission and payload.

Calculate for the Orbiter:

- minimum roll torque needed to rotate the vehicle over 180 degrees in 20 seconds
- minimum pitch torque (y-axis) to rotate the vehicle over 10 degrees in 1 minute.

Problem 11: Propellant tank design (based on internal pressure)

Consider Shuttle Orbiter hydrogen tank. Given is that this tank is of a cylindrical design with spherical end caps, see figure, with a tank diameter of 8.4 m. Total tank volume = 1500 m^3 (~10% larger than needed to allow for tank pressurization and/or thermal expansion of the liquid). Tank maximum expected operating pressure (MEOP) is 3.5 bar. Tank material is aluminum with E = 69 GPa, σ = 145 MPa and ρ = 2700 kg/m^3.



Calculate thickness of tank wall and tank mass based on shell thickness in case we design the tank based on internal pressure only and using a tank safety factor (design x ultimate load factor) of 1.5. You may neglect any effect of welds, etc.

Problem 12: Hydrogen boil-off

Consider hydrogen tank of Space Shuttle ET. The ET contains 104.2 ton of hydrogen propellant at a temperature of 14 K. Surface area of hydrogen tank = 790 m^2. Specific heat of hydrogen = 7.32 kJ/kg-K. Temperature of hydrogen should not exceed 20 K else boiling will occur. ET experiences a flight and surface average heat flux of 3 W/cm^2 for about 600 seconds. Determine the raise in temperature that would be experienced by the initial mass (104.2 ton) of hydrogen propellant and discuss ways to limit the heat flow to the propellant.

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Solutions to problems

1) Vega rocket

Mass fractions of the 1st sub-rocket are:
- Payload mass fraction: 0.293
- Propellant mass fraction: 0.652
- Structural mass fraction: 0.055

As a check we should find that the sum of the three mass fractions is equal to 1.

Mass fractions of the other sub-rockets are to be added in the future.

Velocity increment delivered by the 1st sub-rocket is 2694 m/s. Velocity increment of the other sub-rockets are to be added in the future.

2) Two stage rocket design

In case we divide the velocity increment equally over the 1st and 2nd sub-rocket, we find a rocket mass at take-off of 171.3 ton.

3) Mass fractions of parallel staged rocket stage

To determine the asked for mass fractions of the first sub-rocket, we first need to determine the propellant mass expelled at end of operation of the first sub-rocket. To determine the propellant mass, we first determine the burn time of the EAP. According to the LVC this is 129.8 seconds. Now we determine the propellant mass expelled during this time by the core stage (EPC). From the LVC, we determine a thrust of the main stage in vacuum of 1120 kN and a vacuum specific impulse of 433 seconds. This indicates a mass flow rate of 1120 kN/(433 x 9.81 m/s2) = 263.7 kg/s. Assuming that during operation of the main engine, mass flow rate is constant it follows that the engine uses 236.7 kg/s x 129.8 = 34.2 ton.

Using these data, we find for the first sub-rocket:
Mass at lift off is 2 x 274.8 + 170.2 + 11 + 6.64 + 1.97 + 1.3 = 740.71 ton.
Total propellant mass used is 2 x 237.8 + 34.2 = 509.8 ton.
Mass at end of burn is 740.41-509.8 = 230.91 ton
Payload mass is 230.91 – 2 x 37(empty mass of the EAP) = 156.91 ton.

This gives:
- Payload mass fraction: 156.91/740.71=0.212
- Propellant mass fraction: 509.8/740.71 = 0.688
- Structural mass fraction: 1-0.212-0.688 = 0.010

For the second sub-rocket we find:
Mass after separation of EAB = (170.2 – 34.2) + 11 + 6.64 + 1.97 + 1.3 =156.91 ton. Here the term between brackets represents the mass of the core stage at separation of the two EAB.
Total propellant mass used is 157.3-34.2 = 123.1 ton.
Mass at end of burn is 12.23 + 11 + 6.64 + 1.97 + 1.3 =33.14 ton
Payload mass is = 20.91 ton.

This gives:
- Payload mass fraction: 20.91/156.91=0.133
- Propellant mass fraction: 123.1/156.91 = 0.785
- Structural mass fraction: 1-0.133-0.785 = 0.082

4) Effective exhaust velocity of parallel staged vehicle

To determine the average effective exhaust velocity of the first sub-rocket of the Ariane 5 rocket, we determine:

- Propellant mass flow rate of core stage. Given the thrust level of the core stage of 1MN and an effective exhaust velocity of 4000 m/s, it follows a mass flow rate m = F/w = 250 kg/s
- Propellant mass flow rate of the two booster stages. Each booster stage has a thrust of 5.4MN. Effective exhaust velocity of each booster stage is 3000m/s. This gives a propellant mass flow rate per booster: m = F/w = 1800 kg/s
- Total thrust is 11.8 MN (1 MN + 2 x 5.4 MN= 11.8 MN), total mass flow rate is 3850 kg/s (250 kg/s + 2 x 1800 kg/s = 3850 kg/s). This gives an average effective exhaust velocity for the first sub-rocket of 11.8 MN/3850 kg/s = 3065 m/s.

5) ΔV delivered

The total velocity increment delivered by the Space Shuttle is 9.99 km/s. It is made up of a velocity increment of 2863 m/s until burn out and separation of the two SRBs and 7131 m/s after Main Engine Cut Off (MECO).

6) Approximate ET and SRB size

ET
Tank volume is 1919 m^3 (1371 m^3 hydrogen and 548 m^3 oxygen).
Tank length is 1919 m^3 * $4/\pi/(8.4)^2$ = 34.6 m

SRB propellant volume is 329.2 m^3. For the given diameter of 3.7 m follows a length of the cylindrical volume of 30.6 m.

Actual length will be different as for the liquid tanks we have neglected the ullage volume and for the SRB, we neglected the free volume or the volume of the grain port. In addition, we did not consider the addition of stream line caps (SRB + ET), recovery system (SRB), insulation which reduces available diameter (mostly ET), and a likely alternative shape of end-caps (mostly ET).

7) Location of CoM

CoM is on line of symmetry through orbiter and ET approximately 0.414 m above the combined CoM of ET and SRB's.

8) Location of CoP

CoP is on line of symmetry approximately 3.14 m above the combined centre of pressure of ET and SRBs.

9) MMOI

First calculate the MMOI of the individual elements:
MMOI of the orbiter is ½ x 110 ton x $(3.5\ m)^2$ = 673,350 kgm^2.
MMOI of the ET is ½ x 756,000 kg x $(4.2\ m)^2$ = 6,667,920 kgm^2.
MMOI of each SRB is ½ x 590,000 kg x $(1.85m)^2$ = 1,009,637 kgm^2.

Next compute the MMOI of the total vehicle about its principal axis using Parallel axis theorem. It follows:

$$MMOI = MMOI_{orbiter} + MMOI_{ET} + 2 \times MMOI_{SRB} + M_{orbiter} \times (3.5 + (4.2-0.414))^2 + (M_{ET} + 2 \times M_{SRB})(0.414)^2 = 68{,}617{,}600 \text{ kgm}^2.$$

Note that when comparing the MMOI value for the orbiter alone with the value as given in figure belonging to problem 10, we find our estimated value is a bit low. This illustrates that the result obtained from assuming a cylinder with a homogeneous mass distribution is limited.

10) Control torque

Minimum roll torque is 41.2 kNm.

(We design for a constant torque maneuver. We use the full 20 seconds as specified in the question as this allows for the minimum torque needed. We take 10 seconds for accelerating and 10 seconds for decelerating. In both periods, the vehicle should rotate over 90 degrees.)

Minimum pitch torque: Not yet included.

11) Propellant tank design

First we calculate the length of the cylindrical part of the tank. The spherical head ends have a total volume of 310.3 m^3. This leaves a volume of 1189.7 m^3 for the cylindrical part. Using the know diameter, it follows a length of the cylindrical part of 21.47 m and a total length of the tank of 29.87 m.

Using the relations valid for stresses in pressurized, thin-walled cylindrical tanks and taking into account the given safety factor, we find for the thickness (t) of the cylindrical part (Hoop stress is dimensioning load):

$t = MEOP\ FoS\ D/(2\ \sigma) = \sim 1.52$ cm

Thickness of spherical part is:

$t = \sim 0.76$ cm (half the thickness of the cylindrical part).

Surface area of cylindrical part: $\pi\ D\ L = \pi\ 8.4 \times 21.47 = 566.5\ m^2$
Surface area of spherical end caps: $4\ \pi\ r^2 = 4\ \pi\ 4.2^2 = 221.7\ m^2$

It follows a total tank mass based on shell thickness:

$M_{tank} = (566.5 \times 0.0152 + 221.7 \times 0.0076) \times 2700\ kg/m^3 = 27.8$ ton.

Actual tank mass (including propellant management devices and local strengthening for welding and mounting) is estimated at 1.2 x 27.8 t = 33.4 t, wherein a tank constant K = 1.2 is assumed.

12) Hydrogen boil off
Heat flow to tank per unit of time is 23.7 MW.
Over 600 s this represents a flow of energy of 14.22 GJ to the propellants.

Using the specific heat of hydrogen, it follows that the 104.2 ton of hydrogen propellant will heat up 19 K.

To reduce the heat flow to the propellant we may consider insulating the tank from the hot environment. This can be done by applying an insulator material on the outside of the tank and/or by enclosing the tank by a vacuum (as to reduce convection and conduction).

Appendix A: Launch vehicle data

In the early stages of design, simple (statistics-based) models are used to estimate the main characteristics of the object (here the launch vehicle) to be designed. Such models are not always at hand and must be developed by the designer him/herself. For this data needs to be collected, but that may take quite some time. To assist you in this task, we have collected some hard to come by data in this (and next two) appendix/appendices.

Table A-1: Launch Vehicle Data [Francis]; Cost data are in FY1994 US$.

Vehicle	Country	Length (m)	GLOW (kg)	Payload (kg)	Payload Fraction (%)	Orbital Inc. (deg)	Orbital Alt. (km)	Launch Cost ($ x 10^6)	Cost Per kg ($/kg)
Saturn V*	USA	102.0	3,038,500	118,821	3.91%	28.5	200	N/A	N/A
Energia	Russia	97.0	2,524,600	87,982	3.48%	51.6	200	N/A	N/A
Proton D-1	Russia	59.0	669,130	20,860	3.12%	51.6	200	90	$4,314
Ariane V	Europe	53.9	710,000	18,000	2.54%	5.2	185	120	$6,667
Atlas IIAS	USA	47.5	234,000	8,610	3.68%	28.5	185	105	$12,195
Ariane IV (44L)	Europe	58.5	470,000	7,700	1.64%	5.2	185	100	$12,987
Soyuz	Russia	50.6	310,000	7,000	2.26%	51.6	200	35	$5,000
Delta II (7925)	USA	38.1	231,670	4,971	2.15%	28.5	185	52	$10,461
Titan II	USA	42.9	150,530	3,100	2.06%	28.5	185	40	$13,006
Rokot	Russia	22.0	97,170	1,859	1.91%	62.0	300	8	$4,216
Kosmos	Ukraine	26.3	107,500	1,400	1.30%	51.6	400	11	$8,000
Taurus	USA	27.4	73,000	1,300	1.78%	28.5	200	24	$18,462
Conestoga	USA	15.2	87,407	890	1.02%	40.0	463	20	$22,652
Athena	USA	18.9	68,930	794	1.15%	28.5	200	20	$25,189
Pegasus XL	USA	17.5	24,000	460	1.92%	28.5	200	14	$30,444
Shavit	Isreal	15.0	23,400	160	0.68%	143.0	185	N/A	N/A

Notes:

*Retired
- GLOW = Gross Liftoff Weight
- Launch costs are estimates.
- Payload Mass does not include fairings.

*Retired
-Variations in Inclination and Altitude will result in differences in perfomance.
-Shavit launch vehicle launches in a retrograde orbit. Posigrade performance would be better.
-Energia, Shavit, and Conestoga vehicles have not launched since 1988, 1990 and 1995, respectively.

Note that 'weight' in this table actually should be read as 'mass' as the unit used is 'kg' and not 'N', the latter being the common base unit for force.

Table A-2: Length to Diameter (L/D) and thrust to weight ratio* of some launch vehicles [Francis]

Vehicle	Vehicle Mass (kg)	Thrust to Weight Ratio	L/D
Energia	2,524,600	1.48	7.54
Ariane V	737,000	1.73	10.00
Proton D-1	712,460	1.27	7.97
Ariane IV (44L)	470,000	1.17	15.37
Soyuz	297,400	1.38	11.50
Delta II (7925)	230,000	1.56	15.88
Titan II	150,530	1.28	11.61
Kosmos	107,500	1.40	10.96
Rokot	97,170	1.63	8.80
Conestoga	87,407	1.99	12.67
Taurus	73,030	1.80	11.25
Athena	64,820	2.02	6.25
Pegasus XL	24,000	2.07	13.54
Shavit	23,400	1.80	11.54

*Thrust to weight ratio is given at take-off (using take-off thrust). During flight it usually increases as propellant is used and the weight goes down.[31]

[31] It is mentioned that in literature the term thrust to weight (denoted by T/W) is often used. Still, one should be careful as next to the thrust to weight ratio of the launch vehicle, there is also the thrust to weight ratio of a sub-rocket, stage, and engine. Moreover, since thrust varies with ambient pressure, we have the vacuum thrust based T/W ratio, sea-level thrust bases T/W ratio, etc.

Table A-3: Small launch vehicle payload mass, cost and reliability data [Karabeyoglu]

Launcher	Payload*, kg	Cost#, M$	Cost/Payload, $/kg	Reliability
US Launchers				
Pegasus XL	190	20.0	105,263	34/39
Minotaur	317	19.0	59,936	7/7
Taurus	660	36.0	54,546	6/7
EU Launchers				
Vega	1,395	20.0	14,337	0/0
Russian Launchers				
Dnepr	300	10.0	33,333	9/10
Kosmos	775	12.0	15,484	422/448
Start	167	9.0	53,892	6/6
Strela	700	20.0	28,571	1/1
Others				
Long March 2	1,600	23.0	14,375	22/22
PSLV	900	15.0	16,667	4/7

Space Propulsion Group, Inc. *Sunsynchronous Orbit: 800 km, 98.7° #FY02 Values

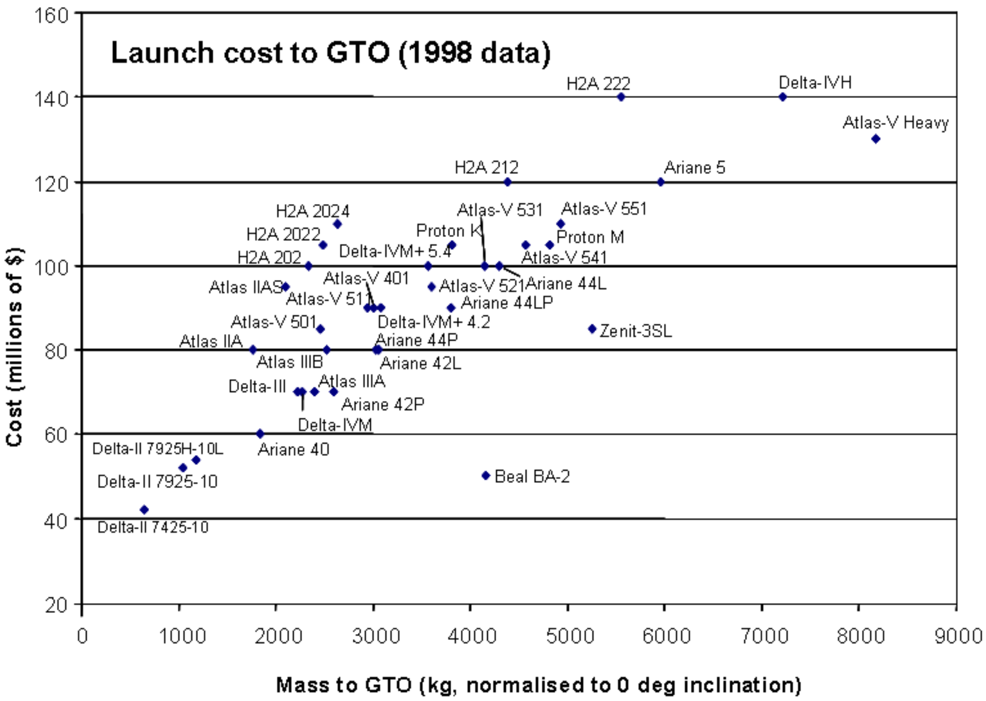

Figure A-1: Launch vehicle cost data and payload mass in to orbit [Buursink]

Table A-4: International Expandable Launch Vehicle Data for Planetary Missions [NASA A]

International Expendable Launch Vehicle Data for Planetary Missions

Launch Vehicle		Incl.		Status		Cost Range	Lead Time	Payload Dimensions (m)				Payload Weight (kg) to:					
Family	Model	(deg)	Country	O/D	Year	(FY'94 $M)	(mo.)	Dia 1	Len 1	Dia 2	Len 2	LEO	GTO	C3=0	C3=10	C3=50	C3=100
Ariane	5	28.5	Europe	O	1996	118-130	36	4.57	5.25	4.57	11.75	18,000	6,800	-	-	-	-
Energia	EUS	51.6	Russia	O	1987	120-120	-	5.50	37.00	5.50	37.00	88,000	-	35,680	31,091	17,446	8,006
	EUS/RCS	51.6	Russia	O	1987	120-120	-	5.50	37.00	5.50	37.00	88,000	-	-	-	-	9,735
H	II	30.4	Japan	O	1994	157-157	24	3.70	10.00	4.60	9.85	10,500	4,000	2,466	1,950	281	-
	II SPKS	30.4	Japan	O	1994	157-157	24	3.70	10.00	4.60	9.85	10,500	-	2,793	2,338	1,148	472
	IIA	30.4	Japan	D	2001	-	-	-	-	-	-	10,000	4,000	-	-	-	-
	IIA LRB	30.4	Japan	D	2001	-	-	-	-	-	-	14,000	6,000	-	-	-	-
J	1	30.4	Japan	O	1996	-	-	-	-	-	-	1,000	-	-	-	-	-
Long March	2C	28.5	China	O	1975	24-24	24	1.80	2.50	3.07	5.00	3,500	1,000	-	-	-	-
	2E	28.5	China	O	1990	47-47	24	3.80	6.50	3.80	6.50	9,200	3,370	-	-	-	-
	3A	28.5	China	O	1993	-	24	3.00	5.25	3.00	5.25	7,000	2,300	-	-	-	-
	3B	28.5	China	D	1994	-	24	3.80	6.50	3.80	6.50	13,600	4,850	-	-	-	-
	3	28.5	China	O	1984	39-39	24	2.32	4.05	2.70	5.33	5,000	1,330	-	-	-	-
	4B	97.8	China	-	-	-	24	2.50	2.91	2.90	6.51	1,440	-	-	-	-	-
M	V	31.2	Japan	D	1996	45-47	30	2.20	6.11	2.20	6.11	1,950	1,215	-	-	-	-
Proton	M	51.6	Russia	O	1996	-	-	-	-	-	-	22,500	5,500	-	-	-	-
Soyuz		51.6	Russia	O	1963	18-18	12	2.85	9.00	2.85	9.00	7,000	-	-	-	-	-
Tsyklon		51.6	Russia	O	1977	12-12	12	2.30	5.90	2.30	5.90	4,000	-	-	-	-	-
Zenit	2	51.6	Russia	O	1970	77-82	36	3.30	9.70	3.30	13.65	13,740	-	-	-	-	-
	3	51.6	Russia	O	1970	77-82	36	3.25	9.72	3.20	12.22	-	4,300	2,911	2,238	398	-

Original prepared by SAIC: March 1994

Table A-5: US expendable launch vehicles for planetary missions [NASA B]

Launch Vehicles		Incl.		Status		Cost Range	Lead Time	Payload Dimensions (m)				Payload Weight (kg) to:					
Family	Model	(deg.)	Country	O/D	Year	(FY'94 $M)	(mo.)	Dia. 1	Len 1	Dia. 2	Len 2	LEO	GTO	C3=0	C3=10	C3=50	C3=100
 Atlas	IIAS	28.5	USA	O	1993	110-142	36	3.30	7.75	4.19	9.74	8,640	3,379	2,698	2,215	845	-
	IIAS-Star 48B	28.5	USA	O	1993	126-158	36	3.30	3.89	4.19	5.88	6,499	-	2,625	2,220	1,156	602
	IIIA	28.5	USA	D	1999	-	-	-	-	-	-	-	3,400	-	-	-	-
	IIIB	28.5	USA	D	-	-	-	-	-	-	-	-	4,500	-	-	-	-
	V (400 series)	28.5	USA	D	2001								5,100				
	V (500 series)	28.5	USA	D									8,200				
 Delta	II-7325	28.7	USA	-	-	55-60	30	2.54	4.67	2.79	4.08	2,760	-	754	614	270	97
	II-7920	28.7	USA	O	1991	49-60	30	2.54	6.38	2.79	5.78	5,045	1,270	692	379	-	-
	II-7925	28.7	USA	O	1990	55-60	30	2.54	4.67	2.79	4.08	-	-	1,277	1,041	461	167
	III	28.7	USA	D	-	-	-	-	-	-	-	-	8,345	3,800	-	-	-
	IV (EELV)	28.7	USA	D	-	-	-	-	-	-	-	-	-	-	-	-	-
Minotaur	Minotaur		USA	O	2000												
Pegasus	Pegasus	28.0	USA	O	1990	8-14	18	1.27	1.90	1.27	2.14	455	125	-	-	-	-
	XL-C Star 24C	23.0	USA	-	-	14-14	20	1.27	1.39	1.27	1.53	544	-	98	77	30	-
	XL-C Star 27	23.0	USA	-	-	14-14	20	1.27	1.14	1.27	1.38	544	-	112	90	36	-
Taurus	Taurus	28.5	USA	O	1994	21-26	18	1.37	2.54	1.37	2.54	1,420	514	329	263	107	35
	XL Star 37XFP	28.5	USA	D	-	23-26	18	1.37	2.54	1.37	2.54	1,565	595	372	296	120	39
	XL/S Star 37FM	28.5	USA	D	-	23-26	18	1.37	2.36	1.37	2.36	1,980	736	474	378	153	50
 Titan	IIG/Star 37	28.7	USA	O	1988	33-38	36	2.84	5.13	2.84	6.66	2,655	-	248	192	-	-
	IIG/Star 48B	28.7	USA	O	1988	33-38	36	2.84	5.13	2.84	6.66	2,655	-	610	492	67	-
	IIS-SSPS	99.0	USA	O	1988	44-55	36	2.84	2.41	2.84	3.94	2,445	-	-	-	-	-
	IIS-PAM D2	99.0	USA	O	1988	44-55	36	2.84	5.58	2.84	7.11	2,885	-	-	-	-	-
	IIS-4SRM-SSPS	99.0	USA	O	1988	44-55	36	2.84	2.41	2.84	3.94	3,342	-	-	-	-	-
	IIS-4SRM-PAM D2	99.0	USA	O	1988	44-55	36	2.84	5.58	2.84	7.11	3,665	-	-	-	-	-
	IIS-10GEM	28.7	USA	O	1988	44-55	36	2.84	5.13	2.84	6.66	5,470	-	1537	1116	-	-
	III/TOS	28.6	USA	O	1989	165-245	33	3.65	10.58	3.65	10.58	14,515	11,000	3,610	2,693	730	-
	IV/Centaur	28.6	USA	O	1994	435-475	33	4.57	15.70	4.57	15.70	18,144	-	7,477	6,330	3,025	775
	IV/SRM/Centaur	28.6	USA	-	-	435-475	33	4.57	15.70	4.57	15.70	18,144	-	9,323	7,867	3,989	1,707
	IV/SRM/IUS	28.6	USA	-	-	330-435	33	4.57	10.50	4.57	10.50	23,350	2,360	3,988	3,193	986	-

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Appendix B: Launch Vehicle Reliability Details

Launch vehicle reliability

Launch vehicles do differ in their degree of reliability. According to 2015 data the number one in reliability is the Soyuz FG. To date, it has gone up 41 times. The number two in reliability, the US Delta 2 has failed only 2 times out of a total of 153 tries. The number three in reliability is the Russian Soyuz U, which is the main booster for their manned missions. It has been launched over 700 times. Below are the satellite launch vehicles that have gone up 30 times or more, and the percentage of those launches that failed.

Table B-1: Historical failure rate of selected space rocket launchers (2015) [Space launch report]

Failure rate	Tries	Successes	Launcher
0%	41	41	Soyuz FG (Russian)
1%	153	151	Delta 2 (US)
3%	772	752	Soyuz U (Russian)
2%	49	50	Ariane 5-ECA (European)
2%	55	54	Atlas 5 (US)
2%	43	42	CZ-4 (China)
3%	74	76	CZ-2 (China)
4%	28	27	H2A (Japan)
7%	65	70	CZ-3 (China)
7%	30	28	PSLV (India)
11%	82	73	Proton-M/Briz-M (Russian)
11%	36	32	Zenit 3SL/DMSL (Russian)
12%	42	37	Pegasus XL (US)

During the early stages of developing and prototyping complex systems, like launch vehicles, reliability often does not meet expectations. In the early stages of the design life a team of engineers and technicians (design, quality, reliability, manufacturing, etc.) analyze every failure that occurs. This team comes up with root causes for the failures and develops design and/or assembly improvements to hopefully eliminate or reduce the future occurrence of that type of failure. The improvements the team comes up with are incorporated into the launcher with an associated increase in reliability, see Figure B-1. After about 40-150 flights, reliability seems to reach its limit. It may therefore be expected that reliability of launchers with flights below 40-150 may still improve.

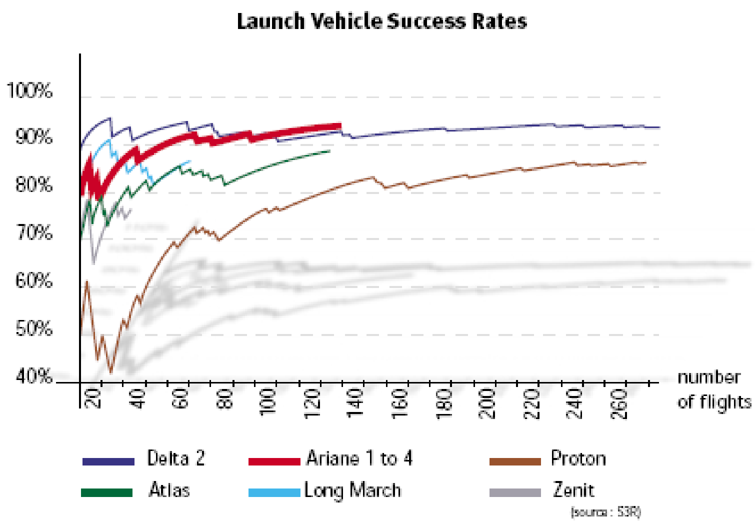

Figure B-1: Launch vehicle success rate (June 2000 data) [Chang]

Launch vehicle reliability details

Chang in 2001 reported subsystem failure rates based on launch vehicle failures over the period 1980-1999 [Chang]. According to Chang available launch-failure data reveals much about patterns in the possible causes of failure. Many failure causes fall into the category of human error: poor workmanship, judgment, or launch-management decisions. Some failures are the result of defective parts. Failure can have its root in any phase of launch vehicle development-difficulties have been noted in inadequate designs and component tests; in improper handling in manufacturing and repair processes; and in insufficient pre-launch checkouts. Many past failures could have been prevented if rigorous reliability-enhancement measures had been taken.

Table B-2: Launch vehicle subsystem failures, 1980-1999 [Chang]]

Country	Propulsion	Avionics	Separation	Electrical	Structural	Other	Unknown	Total
U.S.	15	4	8	1	1	1		30
CIS/USSR	33	3	2			1	19	58
Europe	7	1						8
China	3	1			2			6
Japan	2	1						3
India	1	1	1	1		1		5
Israel	1							1
Brazil	2							2
N. Korea							1	1
Total	64	11	11	2	3	3	20	114

Launch vehicle failure is usually attributed to problems associated with a subsystem, such as propulsion, avionics, separation/staging, electrical, or structures. In some cases failure is ascribed to problems in another area altogether (e.g., launch-pad, ground power umbilical, ground flight control, lightning strike), or to unknown causes (usually when subsystem failure information is not available). The foregoing table clearly demonstrates that among the causes of failure for space launch vehicles worldwide from 1980 to 1999, propulsion subsystem problems predominated. That particular subsystem appears to be the Achilles' heel of launch vehicles. Fifteen of the 30 U.S. failures were failures of the propulsion subsystem, whereas for Europe this was seven out of 8 failures and for Russia 33 out of 58 failures.

According to Chang, propulsion subsystem failures can be divided into failures in solid-rocket motors and liquid-rocket engines. For the United States (US), solid-propellant launch systems include Taurus, Conestoga, Athena, Pegasus, and Scout. Liquid-propellant launch systems include Titan II, Titan IIIA, Titan IIIB, Atlas (except Atlas IIAS), and Delta DM19, A, B, and C. Hybrid launch systems, consisting of liquid-propellant and solid-propellant rockets, include STS, Titan IV, all other Titan III, Atlas IIAS, and all other Deltas. The success rate of the propulsion subsystem in the US from 1980 to 1999 was 98.8 percent for solid-rocket motors and 97.5 percent for liquid-rocket engines).

[SPIAG] in 1999 reported slightly better reliability with an overall reliability for solid propulsion systems of 0.9946 (99.46 percent) and for liquid propulsion systems of 0.9803 (98.03 percent). This is part is attributed to that the failure data also includes much older stages with little or no TVC. Table B-3 provides some more detail.

Table B-3: US Space Flight History (1964 - May 1999) [SPIAG]

Motor type	Failures	Attempts	Success rate
Solid propulsion			
Upper stage	10	627	0.9841
Monolithic	6	2464	0.9976
Segmented	3	402	0.9925
Liquid propulsion			
Cryogenic	9	268	0.9664
Other	28	1612	0.9826

From the data reported in Table 3, it follows:

- Cryogenic (LOX/LH_2) systems fail two times as often as other systems
- Upper stages fail 2-3 times more often than main stages
- Segmented[1]solid rocket stages fail 3 times more often than monolithic rocket stages.

In 2001, Lee published a special project report aiming to identify reliability drivers for advanced liquid propelled space vehicles. He considered 11 liquid propelled vehicles that together accounted for 1642 launches of which 126 were not successful (R = 1 – 126/1642 = 92.3%) due to a failure of one of the launch vehicle subsystems. Of these 89 failures could be identified for which the cause of failure could be traced to a single system. An overview of these 89 failures and their distribution over the various subsystems is given in Figure B-2.

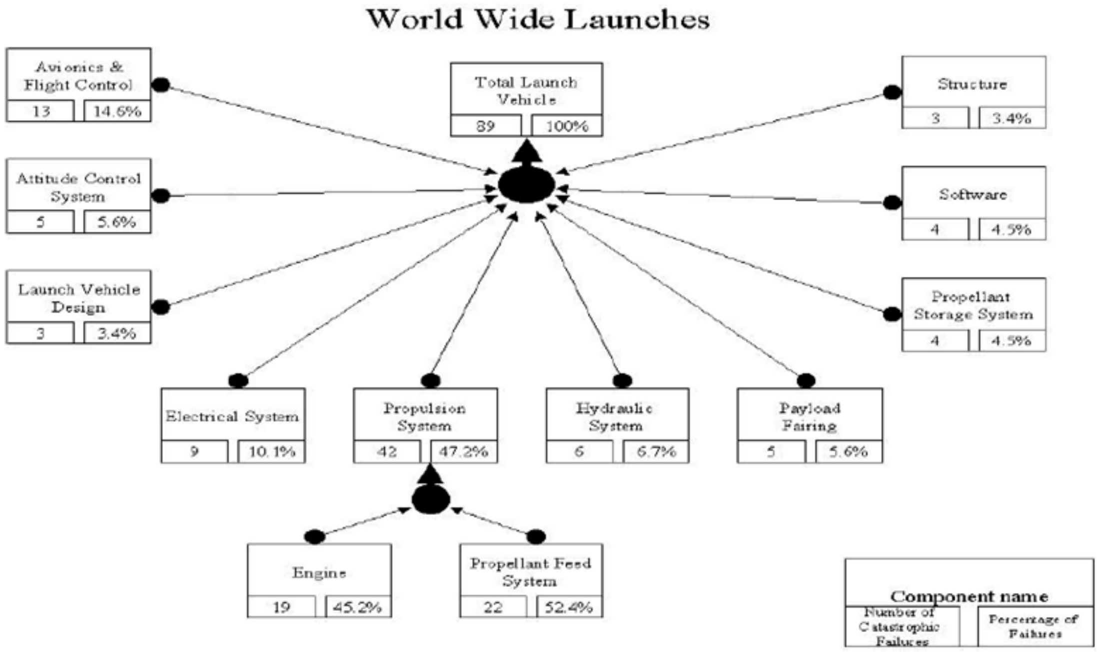

Figure B-2: Historical catastrophic failure ratios of worldwide launches [Lee]

Also indicated in the figure is the percentage contribution to the total number of failures. Note that we neglect that in some of the remaining 37 cases (126 – 89 = 37) also the failure may be attributed to one of the systems shown in figure.

[1] The term "segmented" is used for a solid rocket motor that is manufactured in segments for later assembly. In case of a monolithic rocket motor, the case consists of a single piece (segment). All "large" solid rockets, such as the Shuttle and the Ariane 5 boosters, are manufactured in segments and assembled at the launch site.

In 2004, the Futron corporation [Futron] published subsystem failure rates for US-built launch vehicles from October 1984 through September 2004 showing that separation events and engine/motor failures are the cause of approximately 80% (25 out of 470 orbital launches) of all US launch vehicle failures over the period considered, see Table B-4.

Table B-4: Subsystem failure rates for US-built vehicles from October 1984 through September 2004 [Futron]

Failure Type	Failures	Total Events	Individual Percent Failure Rate
Liquid Propulsion (Start)	3	1255	0.239%
Liquid Propulsion (In-flight)	3	1255	0.239%
Total Liquid Failure	**6**	**1255**	**0.478%**
Solid Propulsion (Shell)	4	1831 (all solids)	0.218%
Solid Propulsion (TVC)	3	571 (TVC only)	0.525%
Solid Propulsion with TVC (TVC and Shell Failure Modes)	--	--	**0.743%**
Stage, Booster, and Payload Separations	6	2577	0.233%
Fairing Separation	1	357	0.280%
Small Solid Booster Separations	1*	1165	0.086%
Electrical	2	470	0.426%
Avionics	2	470	0.426%
Other	1	470	0.213%

*Did not result in total mission loss.

From Table B-4, we learn that the Thrust Vector Control (TVC) system significantly contributes to the failure rate of solid rocket stages or solid propulsion systems. For liquid propulsion, we see that start-up of the liquid stage significantly contributes to the failure rate of liquid rocket systems.

[Arianespace] in 2002 provided some background on 9 catastrophic failures of the European Ariane space launcher on a total of 157 tries, see the next table. Data provided includes an identification of the flight number, where the problem/failure originated and what caused the problem/failure. Of the 9 failures listed, we find that 6 are propulsion related, 1 is related to Guidance, Navigation and Control (GNC) and 2 are related to other causes (cleaning).

Table B-5: Catastrophic failures in Ariane 1-5 [Arianespace]

Flight ID	Origin of the failure	Diagnosis
V02	Combustion Chamber.	Vibrations
V05	Turbo-Pump.	Gear box between two pumps
V15	Combustion Chamber.	LH2 leakage
V18	Combustion Chamber.	Ignition
V36	Piping Obstruction	Cleaning
V63	LOX Pump.	Bearing blockage
V70	Gas Generator.	Cleaning
V88	Guidance, Navigation & Control	Software malfunction
V157	Propulsion /Structure	Nozzle thermo-mechanical collapse

The finding that for Ariane most failures are related to the propulsion system correlates well with the data earlier reported for US launch failures be it that the percentage of propulsion system failures is slightly higher for the European Ariane rocket (67% compared to 50-59% for US systems). Data also shows no failure of a solid rocket booster stage, even though all solid rocket booster stages have TVC. This seems to be a remarkable difference with respect to the US data.

Closing remarks

Launch vehicles do fail. Still different sources may provide seemingly different reliability data. How is this possible? The main reasons are:

1. Investigators using different subsets of data, like:
    a) US only, Russian only or worldwide launch vehicles; Different technologies are in use, which tends to influence the reliability levels attained.
    b) Liquid propelled or solid propelled launch vehicles only; Liquid propelled vehicles tend to be more complex due to the presence of a propellant feed system, which ends up in a lower reliability.
    c) Vehicles that have launched over 50 times or 100 times or more; Vehicles that have only launched a couple of times may still suffer from "infancy" problems that will be avoided in more mature systems.
    d) Different time periods such as Futron corp which focussed on the period 1984-2004 versus Chang who focussed on the period 1980-1999; Modern systems tend to have a higher reliability unless there is added complexity, like a TVC system.
2. Subsystem boundaries may differ so some failure may be allocated to different systems in different studies;
3. Interpretation of data (is it done well?); This is especially important in case we have failures that could not be assigned to any of the subsystems defined.
4. Etc.

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Appendix C: Chemical rocket system component data

Table C-1: Thrust to weight* ratios of various rocket engines [Francis]

Engine	Launch Vehicle(s)	Propellants	Thrust (vac.) (N)	Mass (kg)	Thrust to Weight Ratio
RD-253	Proton	N_2O_4 / UDMH	1,635,000	1,280	130
RD-210	Proton	N_2O_4 / UDMH	582,100	566	105
RS-27A	Delta 3	LO_2 / Kerosene	1,054,200	1,091	99
F-1	Saturn V	LO_2 / Kerosene	7,740,500	8,391	94
RS-56-OSA	Atlas IIAS	LO_2 / Kerosene	386,400	460	86
Vulcain	Ariane V	LO_2 / LH_2	1,075,000	1,300	84
LR 91-7	Titan II	N_2O_4 / Aerozine-50	444,800	565	80
RD-180	Atlas III	LO_2 / Kerosene	4,152,000	5,393	79
J-2	Saturn V	LO_2 / LH_2	1,033,100	1,438	73
SSME	Space Shuttle	LO_2 / LH_2	2,278,000	3,177	73
RL-10A-4	Atlas IIAS	LO_2 / LH_2	73,400	141	53
RL-10	Saturn IV	LO_2 / LH_2	66,700	131	52
L7	Ariane V	N_2O_4 / MMH	27,400	110	25

* Based on vacuum thrust

Table C-2: Properties of some typical chemical propellant combinations (taken from the work of [Francis]); cost data are in FY 1994* US$.

Oxidizer properties				Fuel properties				Propellant properties		
Name	Type	Mass density	Cost per unit mass	Name	Type	Mass density	Cost per unit mass	MR	Mass density	Isp
		[kg/m^3]	[$/kg]			[kg/m^3]	[$/kg)]	[-]	[kg/m^3]	[sec.]
Nitrogen Tetroxide	Storable	1440	6.00	Hydrazine	Storable	1010	17.00	1.1	1197	318
Nitrogen Tetroxide	Storable	1440	6.0	MonoMethylHydrazine (MMH)	Storable	880	17.00	1.8	1173	312
Nitrogen Tetroxide	Storable	1440	6.00	Unsymmetrical DiMethyl Hydrazine (UDMH)	Storable	790	24.00	2.20	1145	309
Hydrogen Peroxide (70%)	Storable	1308	4.04	Ethanol	Storable	870	0.52	6.25	1223	258
Hydrogen Peroxide (95%)	Storable	1418	5.48	Ethanol	Storable	870	0.52	4.25	1266	294
Hydrogen Peroxide (98%)	Storable	1431	5.66	Ethanol	Storable	870	0.52	4.00	1268	297
Liquid Oxygen	Cryogenic	1140	0.08	Liquid Hydrogen	Cryogenic	70	3.60	3.70	268	431
Liquid Oxygen	Cryogenic	1140	0.08	Ethanol	Storable	870	0.52	1.60	1018	312
Cost data for Hydrogen and Ethanol from 1999, Nitrogen tetroxide, Hydrazine and its derivatives from 1990, and liquid Oxygen and Hydrogen from 1985. MR is Mass Mixture ratio of oxygen to fuel										

* To convert FY 1994 US$ to FY 2022 US$, multiply values with a factor 2. Note that this factor 2 is based on an average US inflation rate of 2.5% and that actual values may differ.

Table C-3: Large Solid rocket motor design and technology data [Boury]

MOTOR LAUNCHER	RSRM SHUTTLE	SRMU TITAN IVB	P230 ARIANE 5	SRB-A HIIA	S 138 PSLV
Diameter	3,7 m	3,2 m	3,0 m	2,5 m	2,8 m
Motor length	38 m	31 m	27 m	12 m	20 m
Propellant mass	503 t	314 t	240 t	65 t	138 t
Binder	PBAN	HTPB	HTPB	HTPB	HTPB
A.P. / Al.	70/16	69/19	68/18	68/18	-
Segment	4	3	3	1	5
MEOP pressure	70 b	86 b	69 b	118 b	59 b
Case material	D6AC	Carbon	D6AC	Carbon	Maraging
Throat material	Phenolic	Phenolic	C/C	C/C	Phenolic
Actuation	Flex-seal Hydraulic	Flex-seal Hydraulic	Flex-seal Hydraulic	Flex-seal E.M.A.	Liquid Injection
Motor Inert mass	68 t	29 t	29 t	6 t	18 t
Nozzle Σ	7,5	16	11	18	8

Table 2-1: Large Motors for Heavy Launcher - Design and Technology Data

MOTOR LAUNCHER	RSRM SHUTTLE	SRMU TITAN IVB	P230 ARIANE 5	SRB-A HIIA	S 138 PSLV
Propellant mass	503 t	314 t	240 t	65 t	138 t
Stage inert mass	88 t	36 t	36 t	11,5 t	28 t
Average thrust	11,8 MN	6,1 MN	5,0 MN	1,8 MN	3,6 MN
ISP – vacuum	267 s	284 s	275 s	280 s	269 s
Burning time	123 s	135 s	128 s	100 s	107 s
Vectoring	± 5°	± 6°	± 5°	± 5 °	± 3°

Table 2-2: Large Motors for Heavy Launcher - Stage Propulsive Data

Above table C-3 nicely shows that there is a difference between stage mass and motor mass. This is attributed to additional elements being added to make a stage, like e.g. separation system including separation motors, avionics, nose fairing, and parachutes (if available), see figure.

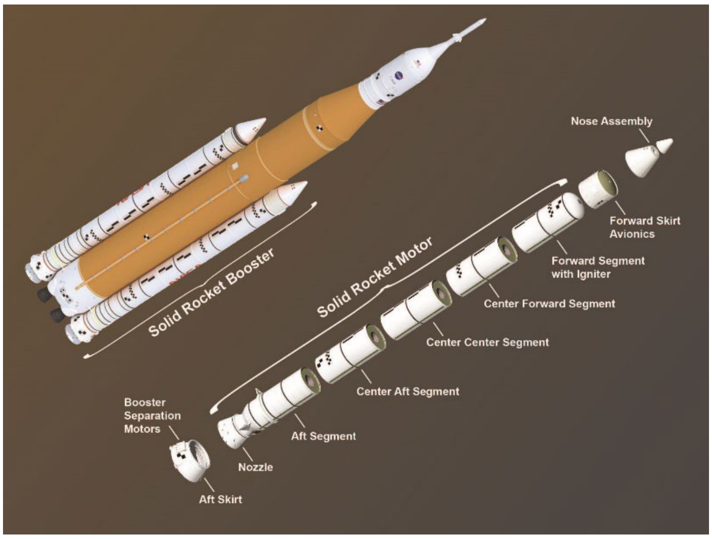


Figure C-172: SLS solid rocket booster consisting of SRM, forward and aft skirt with avionics and separation motors and nose assembly (courtesy NASA)

References

1. Francis Jr., R.J., A systems study of very small launch vehicles, MIT, December 1999.
2. Boury, D., Thinat F-X, Space Launcher Large Solid Propulsion Overview, Market Analysis and General Trends, Snecma Propulsion Solide – SAFRAN Group – France http://enu.kz/repository/2009/AIAA-2009-5519.pdf

Appendix D: Advantages of liquid, solid & hybrid chemical rockets

Advantages of liquid, solid & hybrid (chemical) rockets

The various advantages and disadvantages of liquid, solid and hybrid propellant rockets are described with as purpose to allow initial trade-offs of the three types of rocket motors against the requirements. The numbers mentioned in the text have been taken from existing engines and are considered typical values for space applications. By no means should these values be interpreted as extremes.

1. Performance: High performance liquid rockets using high-energy bipropellants offer a sea level specific impulse in the range 270-360 s. High performance solid rockets are more limited, offering a sea level specific impulse in the range 210-265 s. Hybrid rockets offer a specific impulse in the range 230-270 s, which is similar to those obtainable with bipropellant motors (apart from the very high performing ones, like liquid oxygen – liquid hydrogen). Monopropellant liquid rocket motors offer a specific impulse in the range 160-190 s. Further information can be obtained from Tables D-1 and D-2.

Table D-1: *Sea level specific impulse of some typical liquid chemical propellants*

Propellant combinations	*Isp Range (sec)*
Monopropellants (liquid):	
Low-energy monopropellants	160 to 190
Hydrazine	
Ethylene oxide	
Hydrogen peroxide	
High-energy monopropellants:	
Nitromethane	190 to 230
Bipropellants (liquid):	
Low-energy bipropellants	200 to 230
Perchloryl fluoride-Available fuel	
Analine-Acid	
JP-4-Acid	
Hydrogen peroxide-JP-4	
Medium-energy bipropellants	230 to 260
Hydrazine-Acid	
Ammonia-Nitrogen tetroxide	
High-energy bipropellants	250 to 270
Liquid oxygen-JP-4	
Liquid oxygen-Alcohol	
Hydrazine-Chlorine trifluoride	
Very high-energy bipropellants	270 to 330
Liquid oxygen and fluorine-JP-4	
Liquid oxygen and ozone-JP-4	
Liquid oxygen-Hydrazine	
Super high-energy bipropellants	300 to 385
Fluorine-Hydrogen	
Fluorine-Ammonia	
Ozone-Hydrogen	
Fluorine-Diborane	

Table D-2: *Sea level specific impulse of some typical solid chemical propellants*

Propellant combinations:	*Isp Range (sec)*
Oxidizer-binder combinations (solid):	
Potassium perchlorate:	
Thiokol or asphalt	170 to 210
Ammonium perchlorate:	
Thiokol	170 to 210
Rubber	170 to 210
Polyurethane	210 to 250
Nitropolymer	210 to 250
Ammonium nitrate:	
Polyester	170 to 210
Rubber	170 to 210
Nitropolymer	210 to 250
Double base	170 to 250
Boron metal components and oxidant	200 to 250
Lithium metal components and oxidant	200 to 250
Aluminum metal components and oxidant	200 to 250
Magnesium metal components and oxidant	200 to250
Perfluoro-type propellants	250 and above

2. Size: High-density solid propellants have a mass density in the range of 1500 – 1900 kg/m^3 compared to about 1000 – 1350 kg/m^3 for high-density storable liquid propellants. This compares favorably to the 280 - 375 kg/m^3 attainable for the high performing liquid Oxygen – liquid Hydrogen propellant. For hybrid propellants, it is possible to obtain a density in the range 1000 – 1200 kg/m^3.
3. Flexibility: For SRM's, extinction and re-ignition is hard to realize. Hybrid and liquid rockets on the other hand are much easier to shut down and re-start. For example for monopropellant liquid rockets and hypergolic bipropellant rockets the propellant decomposes under the action of a catalyst (monopropellant) or is self-igniting (hypergolic bipropellant) this can be accomplished through simply opening and closing of a valve. For other bi-propellants, the same advantage holds, but an igniter may be necessary to start combustion. For example, liquid propellant motors using the monopropellant hydrazine or the hypergolic propellant combination of hydrazine and nitrogen tetroxide allow for a precision pulse-mode, where thrust is produced in accurately reproducible impulse bits with a total number of pulses that can easily reach 100000.
   Thrust magnitude control for liquid and hybrid rockets is simply through controlling the flow of the liquid propellant. For solid rockets the thrust is 'pre-programmed' and difficult to change during flight. For example, the Lunar Module Descent Engine had a 10:1 throttle range. For most applications, however, a range of 3:1 seems more than acceptable.
   Liquid rockets allow for easy steering of the thrust vector by use of a gimbal. Steering of the thrust vector for hybrid and solid rockets through gimballing is more complicated as a comparable solid or hybrid motor is much larger. To achieve thrust vector control for solid or hybrid motors, we nowadays use either liquid injection or a vectorable nozzle. Typical thrust vector control angles in practice are up to 9-10 degrees for both liquid and solid rocket motors.
4. Safety: All rocket propellants are explosives, i.e. a substance (or mixture of substances),

which is capable, by chemical reaction, of producing gas at such a temperature and pressure as to cause damage to the surroundings. Liquid and hybrid propellants are more apt to external stimuli than solid propellants. For example, there have been accidents where liquid oxygen was spilled onto asphalt, which caused an explosion when a truck was driven over the spill. The small amount of heat and pressure caused by the tire was enough to trigger an explosion in that concentration of oxygen.

For solid propellants although they require stronger stimuli to ignite, there is the added fact that fuel and oxidizer are intimately mixed so all ingredients are ready at hand. Once burning starts, it will be almost impossible to stop it. Solids also have potential for detonation of the propellant. The latter requires extensive safeguards during propellant manufacturing as well as launcher- and payload processing. For liquid propellants the risk of inadvertent ignition is limited to those cases where leakage occurs, e.g. caused by breakage or launching incidents. For hybrid rocket motors breakage or launching accidents are unlikely to result in an explosion or in involuntary ignition and operation.

Most liquid propellants, like fluorine, hydrazine, nitric acid, mono-methyl hydrazine, oxygen etc, are difficult to handle, because they are very toxic, or corrosive. This requires special pre-cautions; see e.g. Figure on liquid propellant loading. In contrast, solid propellants as well as the solid component of hybrid propellants are relatively harmless in human contact.

Figure 1: Liquid propellant loading (courtesy ESA)

5. Environmental load[33]: The major exhaust products of various solid and liquid systems are shown in the next table.

Table D-3: Major exhaust products of some typical rocket propellants

Propellant system	Major exhaust products
Ammonium perchlorate/aluminium	HCL, H_2O, Al_2O_3, CO_2, N_2
Liquid Oxygen/liquid Hydrogen	H_2O
Liquid Oxygen/hydrocarbon	CO_2, hydrocarbons, H_2O
Nitrogen tetroxide/dimethylhydrazine	NO_x, CO_2, N_2

On a single launch basis, the Space Shuttle injects the greatest mass of exhaust products into the atmosphere of any current propulsion system. Each launch vehicle consists of about 1000 tons of solid propellant and about 800 tons of liquid propellant. Typical concerns related to rocket exhaust products are toxicity, acid rain, Ozone depletion, and the ‘Greenhouse effect’. Further information on the environmental effects of rocket exhaust products can be obtained from e.g. [R.R. Bennet, et al., 1992].

[33] Global impact of rocket exhaust on stratospheric ozone concentration and ground level ultraviolet radiation is estimated at maximum 0,02%.

6. Reliability: SRMs have a simple structure containing few parts. They consist of a pressure vessel, an igniter, a solid grain and a nozzle. Liquid propellant rocket motors are more complicated. They generally consist of one or more tanks; a propellant feed system, a hydraulic system and the engine itself. Liquid monopropellant systems are somewhat less complicated as only a single propellant must be stored and fed to the rocket engine. Hybrid rocket motors can be compared to liquid monopropellant rockets, because like monopropellant systems, they have a single feed system. [Andrews and Haberman, 1991] reported a reliability of 0.998 for SRMs and 0.985-0.989 for liquid propellant systems. More recent data [SPIAG, 1999] for launcher stages indicates an overall reliability for solid propulsion systems of 0.9946 and for liquid propulsion systems of 0.9803. Table below provides some more detail.

Table D-4: US Space Flight History (1964 - May 1999)

Motor type	Failures	Attempts	Success rate
Solid propulsion			
Upper stage	10	627	0.9841
Monolithic	6	2464	0.9976
Segmented	3	402	0.9925
Liquid propulsion			
Cryogenic	9	268	0.9664
Other	28	1612	0.9826

7. Costs: Compared to liquid propulsion, solid propulsion is considered low cost. The main reason is that they are much simpler in design. For illustration, [Andrews and Haberman, 1991] compared <u>large</u> rocket systems and reported a solid propulsion cost of US$ 0.017 per Ns of total impulse and US$ 0.045 Ns for liquid propulsion. More recent data [SPIAG, 1999] for launch vehicle stages with a total impulse equal or larger than 28 MNs indicate a first unit cost for solid propulsion systems of 0.0171 $/Ns and for liquid propulsion systems of 0.0445 $N/Ns, see table below for more detail, and a development cost for liquid systems 6 times higher than for solids. Hybrid rocket motor costs are expected to be somewhere in between those of liquids and solids.

Table D-5: First Unit Cost[34] [SPIAG 1999]

Motor type	Approximate first unit cost		
	$/kg mass	$/N thrust	$/Ns total impulse
Solid propulsion	41.0	1.52	0.0171
Liquid propulsion	113.1	8.88	0.0445

* To convert FY1999 US$ to FY2022 US$, multiply all values with 1.78. Note that this factor 1.78 is based on an average US inflation rate of 2.5% and that actual values may differ.

[34] First unit cost is cost of first unit produced. Later units can be built at lower cost as experience allows later units to be built in less time. This is referred to as the learning experience.

8. Development time: Because of their simplicity, solid rockets have a shorter time to develop than liquid rockets. For example, for the Ariane 5 solid rocket booster the development time is 7 years whereas for the cryogenic main engine (Vulcain) this is 9-12 years including a 3-year long technology preparation period preceding the actual development.
9. Operability: Solid rocket motors require short preparation time since solid propellants can be stored, loaded into the solid rocket motor, over long periods of time. Liquid propellant rockets require a long preparation time due to propellant loading. In case cryogenic liquid propellants are used, extensive cooling is needed. This makes cryogenic propellants extremely difficult to store for long periods of time. Hence, they are only filled a few hours prior to launch. However, even then the propellants are constantly evaporating, causing the formation of ice on the storage tanks, which may cause damage. Also in case of a launch abort, the propellants must first be off-loaded before the rocket can be moved back to the assembly station.

References

- Andrews W.G., and Haberman E.G.; Solids Virtues a Solid Bet, Aerospace America, June 1991.
- Bennet R.R., et al.; Chemical Rockets and the Environment, Aerospace America, May 1991
- SPIAG (Solid Propulsion Industry Action Group); Solid Rocket Motor Briefing, June 1999. Briefing focus is on rocket motors with total impulse equal or larger than 28.0 MNs

Appendix E: Mass estimating relationships for <u>Earth-based</u> rocket type space launch vehicles

A) TU-Delft (B.T.C. Zandbergen)

In this model, Gross Lift-of Mass (here denoted by GLOW) of Earth-based, chemical rocket propelled, expendable launch vehicles is determined directly based on payload mass into Low Earth Orbit (LEO)[35].

Inputs for the model include:

1. Mass of payload ($M_{payload}$)

Launch vehicle Gross Lift Off Mass:

- <u>Payload mass into LEO of up to 7500 kg</u>
  GLOM [ton] = 42.827 $M_{payload\ mass}$ [ton] + 12.996 (based on 10 data points, N_{data} = 10)
  $R^2 = 0.9677$
  RSE = 30.1%

- <u>Payload mass into LEO from 7500 kg up to 120,000 kg</u>
  GLOM [ton] = 25.138 $M_{payload\ mass}$ [ton] + 169.358 (N_{data} = 8)
  $R^2 = 0.9866$
  RSE = 25.7%

The mass data[36] used to obtain the above two relations are shown in next two tables.

- GLOM for payload mass up into LEO of up to 7500 kg:

Launch Vehicle Data				
Vehicle	Country	Length [m]	GLOM [kg]	Payload [kg]
Soyuz-2	Russia	43.5	311700	7100
Delta II (7925)	USA	38.1	231670	4971
Titan II	USA	42.9	150530	3100
Vega	*Europe*	*22.1*	*95100*	*2300*
Rockot	Russia	22	97170	1859
Kosmos C-1	Russia	32.4	109000	1400
Taurus	USA	27.4	73000	1300
Falcon 1	USA	21.3	27670	1010
Pegasus XL	USA	17.5	24000	460
Shavit	Israel	15	23400	160

- GLOM for payload mass into LEO from 7500 kg up to 120,000 kg:

Launch Vehicle Data					
Vehicle	Country	Length [m]	GLOM [kg]	Payload [kg]	Orbital altitude [km]
Saturn V	USA	110	3,038,500	118821	200
Energia	Russia	97	2,524,600	87982	200
Ares I	USA	99.7	900,700	26900	300
Proton D-1	Russia	53	669,130	20860	200
Ariane V	Europe	51.4	746,000	18000	185
H2A (4S fairing)	Japan	52.5	285,000	10000	250
Atlas IIAS	USA	47.5	234,000	8610	185
Ariane IV (44L)	Europe	58.5	470,000	7700	185

[35] For launch from Earth surface into LEO mission characteristic is about 10 km/s. For other missions, like launch from Moon surface to low lunar orbit mission characteristic is quite different (about 2.2-2.6 kms/) and hence these relations are not applicable.
[36] All data taken from "A systems study of very small launch vehicles", by R.J. Francis jr., except Energia, Ares I, H2A, Atlas IIAS, and Falcon 1, which were obtained from Encyclopedia Astronautica and/or Wikipedia.

B) TU-Delft (B.T.C. Zandbergen)

In this model, launch vehicle mass is determined as the summed mass of stages (dry mass including inter stages etc. + propellant mass), fairing and Vehicle Equipment Bay[37] (VEB). So this is excluding payload mass.

$$M_{launch\ vehicle} = \sum_{i=1}^{N} M_{stage\ dry,i} + \sum_{i=1}^{N} M_{propellant,i} + M_{fairing} + M_{VEB}$$

Inputs for this model include:

1. Mass of propellant ($M_{propellant}$); From rocket equation. Mass may include margin to take into account residual propellant (propellant not being used)
2. Vehicle dry mass (M_{dry})
3. Number of stages (N)
4. Vacuum thrust (F_{vac})
5. Surface area of fairing ($S_{fairing}$)

Data on which these models are based are taken from the following launch vehicles: Saturn 1 and V, Atlas II and V, SLS, Zenit, Proton, Soyuz, Kosmos, Delta, Falcon 1 and 9, Ariane 4 and 5, Vega. Titan I, II and III, Delta 2, 3 and 4, GSLV, PSLV, Long March 3, H1, H2, and H2A. Space Shuttle was excluded from the table as it is different from most expendable rockets. The number of data points used is indicated by N_{data}.

Fairing mass

$M_{fairing}$ = 10.3 kg/m^2; SSD = 2.3 kg/m^2 ; $S_{fairing}$ in range 15-250 m^2

Cryogenic (liquid oxygen and liquid hydrogen) rocket stage dry mass (N_{data} = 29)

$M_{stage\ dry}$ [ton] = 0.1011 $M_{propellant}$ [ton] + 1.201
Valid in propellant mass range 8 – 985 ton
R^2 = 0.9914
RSE = 26%

Semi-cryogenic (liquid oxygen and liquid kerosene/RP-1) rocket stage dry mass (N_{data} = 23)

$M_{stage\ dry}$ [ton] = 0.0668 $M_{propellant}$ [ton] + 1.468
Valid in propellant mass range 3.5 – 2040 ton
R^2 = 0.9932
RSE = 25%

Storable (liquid hydrazine and its derivatives and liquid nitrogen-tetroxide) rocket stage dry mass (N_{data} = 25)

$M_{stage\ dry}$ [ton] = 0.0701 $M_{propellant}$ [ton] + 0.768
Valid in propellant mass range 0.5 – 420 ton
R^2 = 0.9668
RSE = 25%

Solid Rocket stage (N_{data} = 13):

$M_{stage\ dry}$ (kg) = 0.1554 $M_{propellant}$ (kg)
Valid in propellant mass range 2 – 500 ton
R^2 = 0.9857
RSE = 22.1%

[37] Not all launch vehicles have a separate VEB. This is mainly the case for the Ariane vehicles, Saturn I and V and SLS. For most of the others the VEB is integrated into the upper stage. Hence this brings some inaccuracy in the final vehicle mass.

The relation for solid rocket stage dry mass and the data points used are plotted in next figure.

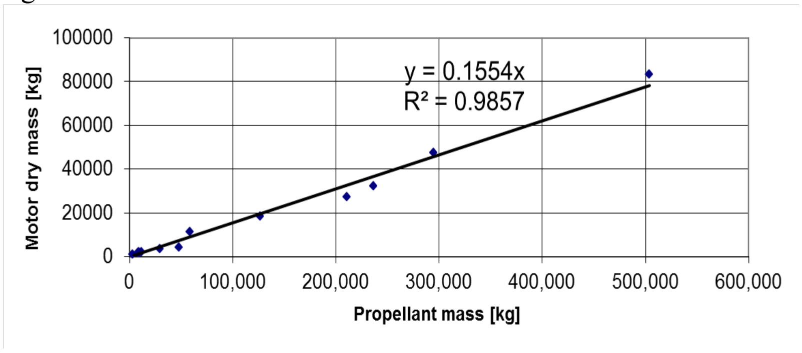


Figure shows the data spread about the estimation relationship. This spread in part is explained from the use of different SRM motor materials. Typical materials in use include steel, titanium, aluminum and composites (fiber based materials). More detailed relations have been derived for steel and composite materials and are given hereafter. In case you are already able to select the case material in an early stage, it is advised to use the latter. If not, use the former.

Steel: N_{data} = 11 (stages included Ariane 5 SRB stage, Space Shuttle SRB stage, PSLV S138, HII SRB stage, Titan III SRM, J1 rocket stage 1 and stage 2, Titan IV UA 1207, M-V rocket M14 stage, Delta Castor 4A, Ariane 4 PAP).

$M_{stage\ dry}$ (kg) = 0.1726 x $M_{propellant}$ (kg) + 1173 kg
$R^2 = 0.9907$
RSE = 20.1%

Kevlar/carbon/graphite composite: N_{data} = 11 (Vega stages P80, Zefiro 23 and Zefiro 9, Athena 1 stage 1 and 2, Pegasus stage 1 and 2, HIIA SRB-A, Titan IVB SRMU, M-V M34 stage).

$M_{stage\ dry}$ (kg) = 0.1173 x $M_{propellant}$ (kg)
$R^2 = 0.9833$
RSE = 24.1%

Clearly, the use of composites allows for a significant reduction in stage dry mass as compared to steel.

Vehicle Equipment Bay (VEB, N_{data} = 11):

M_{VEB} [kg] = 0.345 (M_{dry} [kg])$^{0.703}$
$R^2 = 0.9381$
RSE = 72%

To increase the level of detail and hopefully also the level of accuracy, stage dry mass of liquid propelled rocket stages can also be computed using:

$M_{stage\ dry} = M_{const} + N_{engine}\ M_{engine}$ (N_{engine} is number of engines)

M_{const} is mass of stage construction. It is essentially equal to dry stage mass minus total engine mass.

I) Construction mass[38] (dry stage mass less engine mass) MERs:

Cryogenic (liquid propellant) rocket stages (N_{data} = 26; propellant mass range is 8.5 – 427 ton):

M_{const} [ton] = 0.0872 $M_{propellant}$ [ton] + 1.013
$R^2 = 0.991$
RSE = 30%

Semi-cryogenic (liquid propellant) rocket stages (N_{data} = 34; propellant mass range is 3.5 – 2040 ton):

M_{const} [ton] = 0.046 $M_{propellant}$ [ton] + 1.848
$R^2 = 0.9882$
RSE = 29%

Earth storable liquid propellant rocket stages (N_{data} = 25; propellant mass range is 0.5 – 420 ton):

M_{const} [ton] = 0.0528 $M_{propellant}$ [ton] + 0.8201
$R^2 = 0.9488$
RSE = 27%

II) Engine MERs[39]:

Pump-fed cryogenic bipropellant rocket engine

M_{engine} [kg] = 0.0016 (F_{vac} [N]) + 36.1
$R^2 = 0.9882$
RSE = 10.1%

Pump-fed storable bipropellant rocket engine

M_{engine} [kg] = 0.0011 (F_{vac} [N]) + 33.4
$R^2 = 0.9877$
RSE = 17.66%

Pressure fed storable bipropellant rocket engine

M_{engine} [kg] = 0.107 (F_{vac} [N])$^{0.621}$
$R^2 = 0.9487$
RSE = 39.2%

Improved estimations of mass and size of large bipropellant engines can be obtained from [Zandbergen]. An alternative MER on cryogenic rocket stage structures mass can be obtained from [Pietrobon]. In the latter work some analysis is given that allows for determining the effect of the use of a shared bulkhead and the use of aluminum-lithium as the stage material. This is left for you to explore for yourself.

To be able to determine engine mass, a decision has to be made on stage thrust and number of engines. The thrust level depends on the thrust-to-weight (T/W) ratio that should be realized for the rocket and more specifically the sub-rocket. For stages that act when the rocket lifts off, stage T/W is in range 1.2 – 2.4[40] and for upper stages stage T/W is in range 0.1 – 0.7.

[38] Most liquid propelled rocket stages in use today use aluminium or steel as the main construction material. When replacing these materials by stronger and lighter ones, like Al-Li 2195 or composites, it should be possible to reduce stage construction mass.

[39] Relations do differ a bit from the relations presented in appendix F, but the results are in reasonable agreement (note: consider quantifying the quality of the agreement).

[40] One should be careful here as with Ariane 44L, 4 boosters are used in conjunction with the lower core stage. The combined thrust of these boosters and the core stage is the thrust the vehicle experiences and that should be above 1 otherwise the vehicle never lifts off. However, a single booster stage may have a stage T/W that is less than 1. In that case,

C) University of Maryland model

In this model taken from the work of [Akin, 2002], launch vehicle mass of hydrolox[41] propelled SLV is determined as the summed mass of the propellant tanks, insulation, engines, thrust structures, other tankage systems, structural elements, like inter-stages, aft and front skirt, fairing, avionics, and wiring.

$$M_{launch\ vehicle} = M_{tanks\ dry} + M_{insulation} + M_{propellant} + M_{engine} + M_{thrust\ structure} + M_{fairing} + M_{interstages} + M_{avionics} + M_{wiring}$$

The model also allows for taking into account for solid rocket stages instead of hydrolox. In that case, the first 5 terms on the right hand side (RHS) in above relation are replaced by the mass of the SRM.

Inputs for the model include:

1. Mass of liquid oxygen (M_{LOX})
2. Mass of liquid hydrogen (M_{LH2})
3. Surface area of LOX tank
4. Surface area of LH2 tank
5. Mass of storable propellants
6. Mass of tank content for small storable propellant (monopropellant like hydrazine)
7. Mass of gas (M_{Gas})
8. Surface area of fairing
9. Total rocket mass (M_o)
10. Length of stage (l)
11. Engine thrust (T)
12. Area expansion ratio of liquid rocket engine (ε)
13. Solid propellant mass (M_{Solid})

LOX Tank MERS

- Mass of tank

$$M_{LOX\ Tank}\ (kg) = 0.0152\ M_{LOX}\ (kg) + 318$$

- Mass of insulation

$$M_{LOX\ Insulation} = 1.123\ kg/m^2$$

LH2 Tank MERS

- Mass of tank

$$M_{LH2\ Tank}\ (kg) = 0.0694\ M_{LH2}\ (kg) + 363$$

- Mass of insulation

$$M_{LH2\ Insulation} = 2.88\ kg/m^2$$

Other Tankage MERS

- Storable Propellants:

$$M_{Storables\ Tank}\ (kg) = 0.316\ [M_{Storables}\ (kg)]^{0.6}$$

- Storable Propellants:

$$M_{Small\ Liquid\ Tank}\ (kg) = 0.1\ M_{Contents}\ (kg)$$

- Small Tank (pressurized gases):

$$M_{Small\ Gas\ Tank}\ (kg) = 2\ (kg)$$

[41] Hydrolox is short for a propellant combination consisting of liquid hydrogen and liquid oxygen.

Liquid propellant rocket engine and thrust structure

- Pump-Fed Liquid Propellant Rocket Engine Mass

$$M_{Engine}\ (kg) = 7.81 \times 10^{-4}\ T(N) + 3.37 \times 10^{-5}\ T(N)\ \varepsilon^{0.5} + 59$$

- Thrust Structure Mass[42]:

$$M_{Thrust\ Structure}\ (kg) = 2.55 \times 10^{-4}\ T(N)$$

Solid Rocket Motors:

$$M_{Motor\ dry}\ (kg) = 0.135\ M_{Solid}\ (kg)$$

Other Structural MERS

- Fairings and shrouds (inter-stages, etc)

$$M_{Fairing}\ (kg) = 13.3\ kg/m^2$$

- Avionics

$$M_{Avionics}\ (kg) = 10[M_o\ (kg)]^{0.361}$$

- Wiring

$$M_{Wiring}\ (kg) = 1.058\ SQRT(M_o\ (kg)*l^{0.25}$$

The estimation relationships given are offered as is. It is mentioned that for none of the relationships an indication is given of the range of applicability and the uncertainty in the estimate. Hence it is for the designer/user to determine for him/herself how accurate the estimation relationships are in specific situations. For this a comparison of estimated date with real data may serve to do the job. This process is referred to as validation. Based on the results of the validation (validation results) the designer can consider adding appropriate margins/contingencies to allow for taking into account such uncertainties.

[42] Little to no data can be found on the mass of the thrust structure/frame of rocket engines, except for two rockets. For the Saturn V first stage, it is reported a mass of the engine structure of 21000 kg (vacuum thrust level is 22.0 MN) and for the Ariane 5, lower core stage a value of 1500 kg (vacuum thrust level is 1 MN). Filling in numbers show quite a difference between estimated and actual mass. Hence, this shows that the relation provided in text is at least a bit dubious. Hence, when using a relation taken from literature, one should make sure to check whether the relation is sufficiently accurate. That the relation is a bit dubious may also follow if one considers that the mass of a thrust frame not only depends on thrust, but also on the stage diameter and the type of structure (cylindrical with cross-beams or truncated cone) considered as well as the used structural materials.

D) Cryogenic Rocket Mass Budgeting

The model[43] presented in this section has been taken from the work of [Apel].

Inputs for the model include:

1. Mission characteristic velocity (Δv)
2. Specific impulse
3. Propellant mass ($M_{propellant}$)
4. Propellant mass density ($\rho_{propellant}$)
5. Tank volume (V_{Tank})
6. Vacuum thrust (F_{vac})
7. Initial vehicle mass (M_o)
8. Vehicle mass recovered ($M_{Recovery}$)
9. Vehicle mass at landing ($M_{Landing}$)
10. Usable propellant mass ($M_{Prop.\ Usable}$)

- Mass of structure (body + tankage) of a rocket stage:

$$M_{Structure}\ [\text{ton}] = 0.1 \left(\frac{M_{Propellant}\ [\text{ton}]}{6\ \rho_{propellant}\ [\text{ton/m}^3]} \right)^{0.95} + 0.02\ M_o\ [\text{ton}]$$

- Tankage mass (aluminum tanks):

$$M_{Tank}\ [\text{kg}] = \left(2.903 + \frac{46.25}{\ln\left(V_{Tank}\ [\text{m}^3]\right)} \right) \cdot V_{Tank}\ [\text{m}^3]$$

- Mass of a Pump-Feed Rocket engine:

$$M_{Engine}\ (\text{kg}) = 8.09 \times 10^{-3}\ (F_{vac})^{0.9}\ [\text{N}]$$[44]

- Avionics

$$M_{Avionics}\ [\text{ton}] = [5.938 - 0.00404\ (M_{Structure} + M_{Engines})]\ [0.01\ (M_{Structure} + M_{Engines})]$$

  With $M_{Structure} + M_{Engines}$ expressed in [ton]

- Mass of Recovery System (Parachute):

$$M_{Parachute} = 0.1\ M_{Recovery}$$

- Mass of Landing Gear:

$$M_{Landing\ Gear} = 2\text{-}3\%\ M_{Landing}$$

- Mass of Residual and Reserve Propellants:

$$M_{Prop.\ residual} = M_{Prop.\ reserve} = 0.01\ M_{Prop.\ Usable}$$

---

[43] Relations are offered as is. It is not clear for this model how accurate the relations are. It is left for the designer/user to determine for him/herself how accurate the model is in specific situations. For this a comparison of estimated date with real data may serve to do the job.

[44] Applying this relation to determine the engine mass of the SpaceX Merlin 1D engine with a vacuum thrust of 981 kN gives an engine mass of 1997 kg. On Wikipedia, look for Merlin 1D, a value of 470 kg is reported. This then would signify that the here used expression for thrust is about a factor 4 off.

E) Example results

The models described under B to D have been applied to the design of a single stage cryogenic rocket with:

- propellant mass of 100 ton of which 2 ton remains as reserve and residual propellant
- oxidizer to fuel mass ratio of 4:1 (to allow calculating oxidizer and fuel mass separately)
- hydrogen mass density of 76 kg/m^3 (to allow calculating fuel tank volume)
- lox mass density of 1140 kg/m^3 (to allow calculating oxidizer tank volume)
- fairing surface area of 40 m^2
- 1 engine with vacuum thrust of 1.8 MN
- nozzle area ratio of 12 (arbitrarily selected)
- no landing gear
- no recovery system

The results are given in next table.

Table E-1: Comparison of results of various launcher mass models (ton = metric ton)

Item	Model B1 (Zandbergen)	Model B2 (Zandbergen)	Model C	Model D (Apel)
LOX tank mass			1534 kg	
LOX tank insulation mass			92 kg	
LH_2 tank mass			1751 kg	
LH_2 tank insulation mass			572 kg	
Tank mass			3959 kg	3.62 ton
Inter-tank fairing			1825 kg	
Aft fairing mass			1321 kg	
Thrust structure mass			459 kg	
Stage structure mass		9733 kg	7554 kg	6.80 ton
Engine mass		2916 kg	1675 kg	3.45 ton
Wiring mass			667 kg	
Stage dry mass	11316 kg	9396 kg	9913 kg	10.25 ton
VEB/avionics mass	258 kg	274 kg	660 kg	0.60 ton
Fairing mass	412 kg	412 kg	532 kg	-
Total SLV dry mass	11981 kg	13335 kg	11105 kg	10.85 ton
Residual + reserve mass	2000 kg	2000 kg	2000 kg	2.00 ton
Total SLV empty mass	13981 kg	15335 kg	13105 kg	12.85 ton
SLV loaded mass	113981 kg	115335 kg	113105 kg	112.85 ton

Notes:

1. Tank mass is sum of LOX and LH_2 tank mass + insulation mass.
2. Stage structure mass is sum of tank mass, inter-tank mass, aft-fairing mass, and thrust structure mass.
3. For the model C a total tank volume was found of about 333 m^3 (LOX tank volume ~ 70 m^3 and hydrogen tank volume ~ 263 m^3). Propellant volume was found to be 0.3 ton/m^3. Stage length, and tank surface area were determined assuming spherical tanks. This gives a stage length of about 13.1 m (LOX tank diameter is 5.1 m and LH_2 tank diameter is 8.0 m), a LOX tank surface area of 82 m^2 and a hydrogen tank surface area of 199 m^2. Aft fairing is taken to be cylindrical with diameter equal to hydrogen tank diameter and height equal to tank radius. Inter-tank fairing is taken to be a frustum with diameters determined by tank diameters and height equal to the sum of the radii of the LOX tank and hydrogen tank.

4 different values result for stage dry mass (and hence also for the other masses calculated). So naturally can ask oneself how to find out which method is best. This can only be answered in case we start comparing the outputs of the model with actual data (existing stages). This is left for the reader to explore for her/himself. Without giving proof, it is mentioned that most of the models presented here can be quite off when applying them to modern rockets like Falcon 9 (FT version).

F) References

1. Akin, David L., Presentation on Mass estimation relationships for rocket launch vehicles, from a lecture series on Space Systems Design, University of Maryland, 2002.

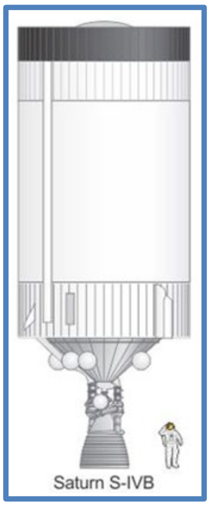
2. Apel, U., Presentation on Rocket Mass Budgets and Staging, presented by U. Apel from Hochschule Bremen at 6th CVA Summer School, Barcelona, July 18 to August 14, 2005.

3. Pietrobon S.S., Analysis of Propellant Tank Masses, Small World Communications, First Avenue, Payneham South SA 5070,Australia. email: steven@sworld.com.au, 2009.

4. Zandbergen, B., Mass and size estimation relationships of pump fed rocket engines for launch vehicle conceptual design, 6th European Conference for Aeronautics and Space Sciences (EUCASS), Krakow, Poland, July 2015.

Appendix F: Size data of chemical rocket propelled rocket stages

Size data and more particularly stage mass densities are listed for a number of stages of conventional, vertical take-off, expendable launch vehicles (SLV). Such data can be used to estimate the size of rocket stages. Stage mass density is based on stage loaded mass and stage volume. The latter is based on stage height and diameter, where stage height and diameter are based on the stage envelope and essentially make up a cylindrical box (the envelope) that encompasses the whole stage, see figure with the envelope marked in blue (thick line).



In the table, we have marked per stage the propellant combination used as this tends to be quite determining for the stage mass density. Additionally, we have also marked whether the stage is a booster stage (marked with 'B'), an upper stage ('U') or a lower first, second, or third stage (marked with '1' , '2', '3'). This is to allow investigating whether the stage position in the SLV stack affects the stage mass density. This can be the case for, for instance, upper stages as these generally carry most SLV avionics unless they are collected in a separate Vehicle Equipment Bay (VEB).

When looking at the values for some propellant combinations, sometimes large variations can be found. This brings of course some uncertainty and hence, this may necessitate further investigations into other (more detailed) size estimation methods.

Table F-I: Characteristic size data of expendable chemical rocket stages [1, 2, 3]

#	Stage	SLV	Propellants	Stage type	Loaded stage mass [ton]	Stage height [m]	Stage diam [m]	Stage volume [m^3]	Stage mass density [kg/m^3]	Developer	Origin
1	EPC	Ariane 5	Hydrolox	1	170.2	30.7	5.4	703.1	242.1	EADS ST	Europe
2	Core stage	H2B	Hydrolox	1	202.0	38	5.2	807.0	250.3	MHI	Japan
3	Core stage	H2A	Hydrolox	1	114.0	37.2	4	467.5	243.9	MHI	Japan
4	CBC	Delta 4M	Hydrolox	1	226.36	40.8	5.13	843.3	268.4	Boeing	USA
5	Energia	Energia	Hydrolox	1	905	58.77	7.75	2772.4	326.4	NPO Energiya	Russia
6	KVRB	Proton-M, Angara-A5	Hydrolox	U	22.0	8	4.1	105.6	208.3	Krunichev	CIS
7	H10	Ariane 44 LP	Hydrolox	U	12.1	9.9	2.6	52.6	230.2	Aerospatiale	Europe
8	H10+	Ariane 44 LP	Hydrolox	U	12.4	10.2	2.6	54.2	229.0	Aerospatiale	Europe
9	H10-3	Ariane 44 LP	Hydrolox	U	13.1	10.2	2.6	54.2	241.9	Aerospatiale	Europe
10	H8	Ariane 3	Hydrolox	U	9.5	9.08	2.6	48.2	196.9	Aerospatiale	Europe
11	Stage 2	H-IIB	Hydrolox	U	20.0	9.2	4	115.6	173.0	MHI	Japan
12	CZ-3A	Long march 7 (CZ-7)	Hydrolox	U	20.9	12.38	3	87.5	239.2	CASC	China
13	Centaur III (SEC)	Atlas 5	Hydrolox	U	22.6	12.68	3.05	92.6	243.8	Lockheed Martin	USA
14	Centaur III (DEC)	Atlas 5	Hydrolox	U	22.9	12.68	3.05	92.6	247.6	Lockheed Martin	USA
15	4M	Delta 4M	Hydrolox	U	23.26	12.2	4.07	158.7	146.5	Boeing	USA
16	4M+	Delta 4M	Hydrolox	U	30.69	13.7	5.13	283.2	108.4	Boeing	USA
17	Centaur G	Titan 4	Hydrolox	U	23.48	9	4.33	132.5	177.2	General Dynamics	USA
18	Stage 1	Zenit-2	Kerolox	1	318.8	33	3.9	394.2	808.7	Yuzhnoye	CIS
19	CCB	Atlas 5	Kerolox	1	305.3	32.46	3.81	370.1	824.9	Lockheed Martin	USA
20	Block A	Soyuz 2	Kerolox	1	101.9	27.8	2.95	190.0	536.3	TsSKB	CIS
21	CRM 1	Angara 1.1	Kerolox	1	124.4	25.14	2.9	166.1	749.2	Krunichev	CIS
22	Stage 1	Falcon 9 v1.1	Kerolox	1	299.8	41.2	3.66	433.5	691.6	SpaceX	USA
23	ELET Thor	Delta II	Kerolox	1	101.8	26.1	2.44	122.0	834.1	ULA	USA
24	K3-1	Long march 7 (CZ-7)	Kerolox	1	186	26	3.35	229.2	811.6	CASC	China
25	K3-2	Long march 7 (CZ-7)	Kerolox	2	71	11.5	3.35	101.4	700.5	CASC	China
26	Block I	Soyuz 2	Kerolox	2	25.2	6.74	2.66	37.5	672.8	TsSKB	CIS
27	Block 1E	Aurora	Kerolox	3	32.5	8.1	3.6	82.4	394.2	Energia	CIS
28	K2-1	Long march 7 (CZ-7)	Kerolox	B	81.5	26.5	2.25	105.4	773.5	CASC	China

29	Corvet	Aurora	Kerolox	U	12.8	5.28	3.454	48.5	262.7	Energia	CIS
30	DM(11C861-03)	Proton-M, Angara-A4	Kerolox	U	22.3	6.3	3.7	67.7	329.2	Energia	CIS
31	Stage 2	Zenit-2	Kerolox	U	80.6	10.4	3.9	124.2	648.8	Yuzhnoye	CIS
32	Stage 2	Falcon 9 v1.1	Kerolox	U	96.57	13.8	3.66	145.2	665.1	SpaceX	USA
33	Orion 50SXL1	Pegasus XL	Solid	1	16.4	10.27	1.27	13.0	1260.6	Alliant techsystems	USA
34	Stage 0	Taurus XL	Solid	1	53.02	12.8	2.38	56.9	931.1	Thiokol	USA
35	P80 FW	Vega	Solid	1	96.263	11.714	3.005	83.1	1158.7	Avio	Europe
36	Orion 50XL	Pegasus XL	Solid	2	4.306	3.07	1.27	3.9	1107.2	Alliant techsystems	USA
37	Stage 1	Taurus XL	Solid	2	17.93	10.3	1.28	13.3	1352.8	Alliant techsystems	USA
38	Z23 FW	Vega	Solid	2	26.3	8.454	1.907	24.1	1089.2	Avio	Europe
39	Stage 2	Taurus XL	Solid	3	4.33	3.6	1.28	4.6	934.7	Alliant techsystems	USA
40	Z9 FW	Vega	Solid	3	12	4.367	1.907	12.5	962.1	Avio	Europe
41	EAP	Ariane 5	Solid	B	274.8	31.24	3.049	227.8	1206.3	EADS ST	Europe
42	SPB	Ariane 44 LP	Solid	B	12.6	11.324	1.07	10.2	1237.4	BPD	Europe
43	SRB	Atlas 5	Solid	B	46.5	19.5	1.55	36.8	1263.6	Lockheed Martin	
44	SRB	Delta 4M	Solid	B	33.15	15.2	1.52	27.6	1201.9	Alliant techsystems	USA
45	SRB-A3	H-IIB	Solid	B	76.6	15.1	2.5	74.1	1033.4	MHI	Japan
46	SRM	Titan 4	Solid	B	315.7	34.41	3.11	261.4	1207.8	United Techn.	USA
47	Orion 38	Pegasus XL	Solid	U	0.8723	1.34	0.97	1.0	880.9	Alliant techsystems	USA
48	Star 48	Delta II	Solid	U	2.21	2.04	1.25	2.5	882.8	Boeing	USA
49	Star 37 FM	Delta II	Solid	U	1.147	1.676	0.933	1.1	1001.0	Boeing	USA
50	Stage 3	Taurus XL	Solid	U	0.98	2.1	0.97	1.6	631.5	Alliant techsystems	USA
51	L220	Ariane 44 LP	Storable	1	253.0	25.4	3.8	288.1	878.3	Aerospatiale	Europe
52	L140	Ariane 1	Storable	1	160.0	18.4	3.8	208.7	766.7	Aerospatiale	Europe
53	Proton M stage 1	Proton M	Storable	1	450.01	21.18	7.4	910.9	494.0	Krunichev	CIS
54	Tsyklon 3 stage 1	Tsyklon 3	Storable	1	127	18.75	3	132.5	958.2	Yushmash	CIS
55	Core 1	Titan 4	Storable	1	162.84	26.37	3.05	192.7	845.2	Martin Marietta	USA
56	L33	Ariane 44 LP	Storable	2	38.5	11.6	2.6	61.6	625.1	ERNO	Europe
57	L33	Ariane 1	Storable	2	38.5	11.6	2.6	61.6	625.1	ERNO	Europe
58	Delta-K	Delta II	Storable	2	7.0	5.97	2.44	27.9	249.0	ULA	USA
59	Proton M stage 2	Proton M	Storable	2	167.51	17.05	4.1	225.1	744.1	Krunichev	CIS
60	Tsyklon 3 stage 2	Tsyklon 3	Storable	2	53.3	10.08	3	71.3	748.1	Yushmash	CIS

61	Core 2	Titan 4	Storable	2	39.46	9.94	3.05	72.6	543.4	Martin Marietta	USA
62	Proton M stage 3	Proton M	Storable	3	50.26	4.11	4.10	54.3	926.2	Krunichev	CIS
63	LPB	Ariane 44 LP	Storable	B	43.7	18.6	2.17	68.8	635.3	ERNO	Europe
64	EPS	Ariane 5	Storable	U	11.0	3.396	3.356	40.8	269.4	EADS ST	Europe
65	Fregat	Soyuz 2	Storable	U	6.5	1.5	3.35	13.2	494.3	Lavotchkin	CIS
66	Breeze M	Proton M	Storable	U	22.17	2.61	4.1	34.5	643.4	Krunichev	CIS
67	S5M	Tsyklon 3	Storable	U	4.63	2.75	2.7	15.7	294.1	Yushmash	CIS
68	AVUM	Vega	Storable	U	1.265	2.04	1.952	6.1	207.2	Avio	Europe

References

1. Encyclopedia Astronautica, http://www.astronautix.com

2. Wikipedia

3. Wilson, A. (Ed.), Jane's Space Directory, 1995-1996.

Appendix G: Liquid rocket engine mass and size estimation relationships

Mass estimation

From theory, it can be shown that liquid rocket engine or thruster mass is closely related to thrust level. Below we will present relations valid for different types of rocket engines varying from large to small engines.

Large Liquid Rocket Engine Assemblies

The figure shown below presents a graph of engine mass versus (vacuum) thrust level for several large cryogenic (LOX-LH_2) and semi-cryogenic rocket engines. Data includes the mass of thrust chamber and turbo-pump assembly. The data for the graph has been taken from [Zandbergen-A].

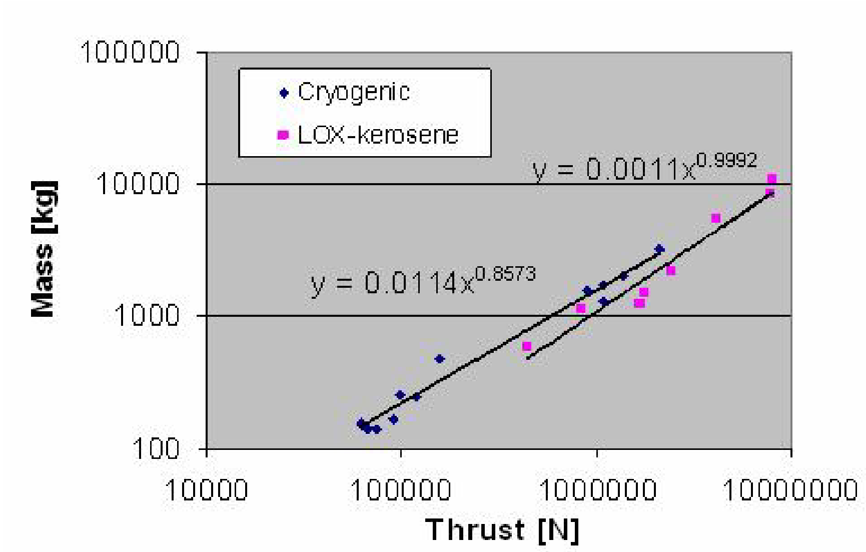


Figure 1: Mass estimation of large pump-fed liquid rocket engines

The Relative Standard Error (RSE) of the estimate for LOX-kerosene is 23.9 % and 18.6% for LOX/LH_2.

More detailed information can be obtained from [Zandbergen-C].

Pressure-fed, Storable Bipropellant, Thruster (thrust chamber assembly + flow control valve)

The next figure shows a graph of thruster mass versus (vacuum) thrust level for 29 actual bipropellant (hydrazine-nitrogen-tetroxide, MMH-nitrogen-tetroxide) thrusters. Mass data given include the mass of the thrust chamber and thruster valve(s). In the figure also a least squares curve fit of the thrust-to mass ratio versus thrust is given. The data for this graph has been taken from [Zandbergen-B].

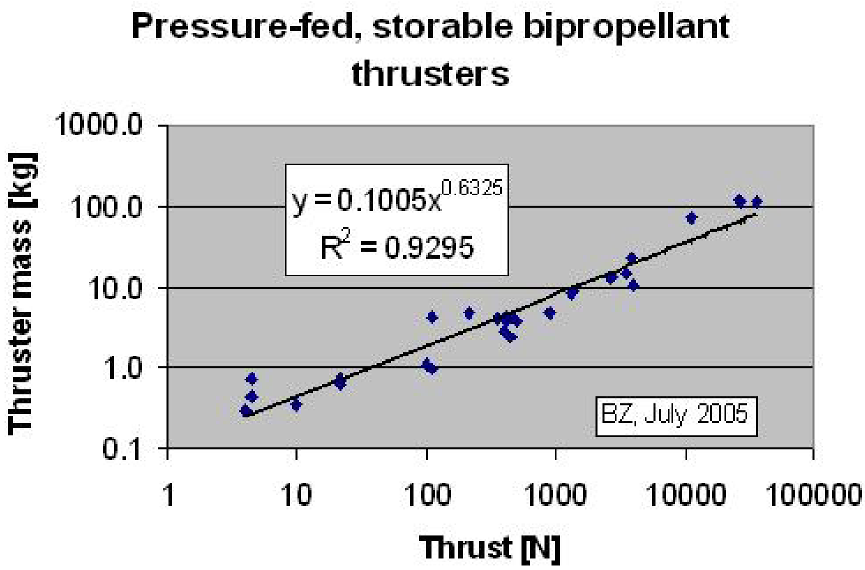


Figure 2: Mass estimation of small pressure-fed storable bipropellant rocket engines

Remarks

Next to thrust level, we find an influence of the propellant used. For illustration, the mass of a cryogenic engine typically is about a factor 1.5 higher than for a semi-cryogenic engine. Also we find that at identical thrust level (up to about 400 N) pressure-fed monopropellant thrusters have a mass advantage over pressure-fed bipropellant thrusters. The data presented however, does not provide insight in the effect of chamber material used, chamber pressure, nozzle expansion ratio, etc., which are also expected to be of influence. More detailed modeling is needed to find out how these parameters will affect thruster or engine mass. This modeling in part can be found in [Zandbergen-B].

References

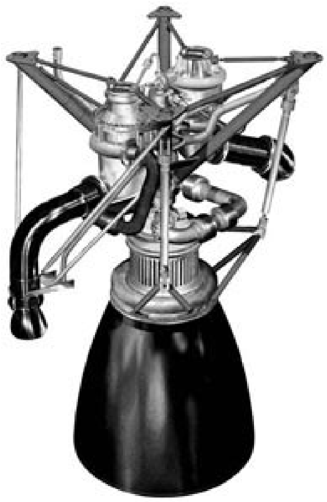
1. Zandbergen, B.T.C. (A), Part on space propulsion taken from Space Engineering & Technology II (course notes), Delft University of Technology, Faculty of Aerospace Engineering, 2003.
2. Zandbergen, B.T.C. (B), Thermal Rocket Propulsion (course notes), Delft University of Technology, Faculty of Aerospace Engineering, 2004.
3. Zandbergen, B. (C), Mass and size estimation relationships of pump fed rocket engines for launch vehicle conceptual design, 6th European Conference for Aeronautics and Space Sciences (EUCASS), Krakow, Poland, July 2015.

Size estimation

Humble et al in "Space Propulsion Analysis and Design" (1995) use a simple method to obtain a first estimate of the envelope (length and maximum diameter) of a liquid rocket engine. The method is based on that historic data shows engine thrust correlates well with engine length respectively diameter. Following the approach of Humble et al, we have plotted engine length and diameter versus thrust magnitude for existing liquid propellant rocket engines in various classes. Below the results are given for two different classes. For modeling applications the results also include a curve fit. First however, the various parameters are defined in some detail.

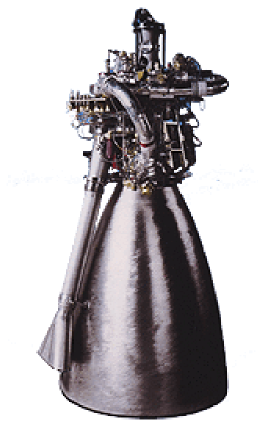
The envelope of a rocket engine is defined as the smallest enclosure that covers the rocket engine completely. Rocket engine is here taken to be:

- For small tank pressure-fed thrusters: thrust chamber (nozzle + combustion chamber) and flow control valve(s)
- For pump-fed rocket engines: thrust chamber + pump system + gimbal + TVC system



RS-68 (3.3 MN)

Vulcain (1.075 MN)



HM-7B (62 kN)

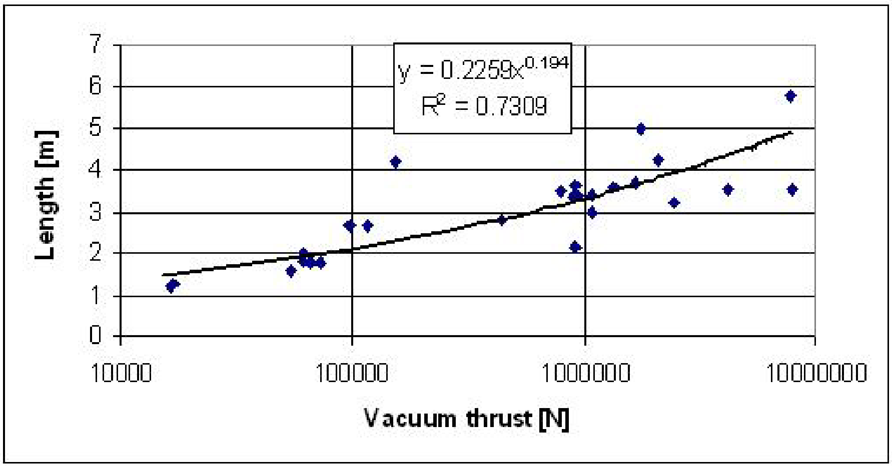
S4 thruster (4N)

Figure 3: Typical liquid rocket engines (courtesy Rocketdyne, ESA, Astrium)

From the above figures we learn that the largest diameter does not necessarily equal the nozzle exit diameter. Also engine length may be substantially different from the length of the thrust chamber.

Pump-fed liquid propellant rocket engine envelope

Engine length vs thrust for pump-fed liquid propellant rocket engines



Engine diameter versus thrust for pump-fed liquid propellant rocket engines

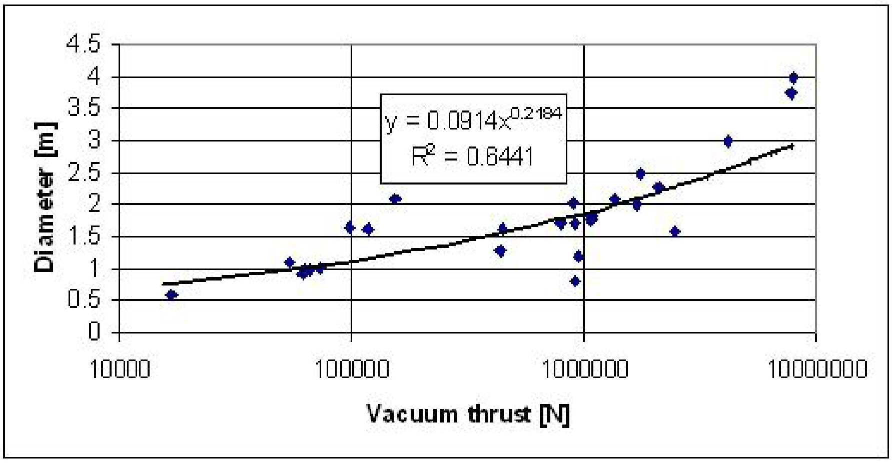
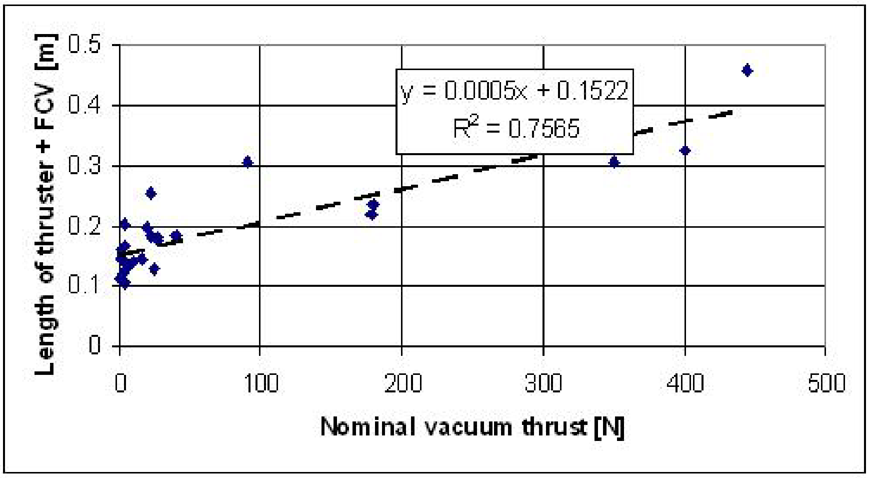


Figure 4: Size estimation of large pump-fed liquid rocket engines

More accurate relationships can be obtained from [Zandbergen].

Hydrazine monopropellant thruster (+ flow control valve) envelope

Length versus thrust of hydrazine monopropellant thrusters (including thruster control valve)



Diameter versus thrust of hydrazine monopropellant thrusters (including thruster control valve)

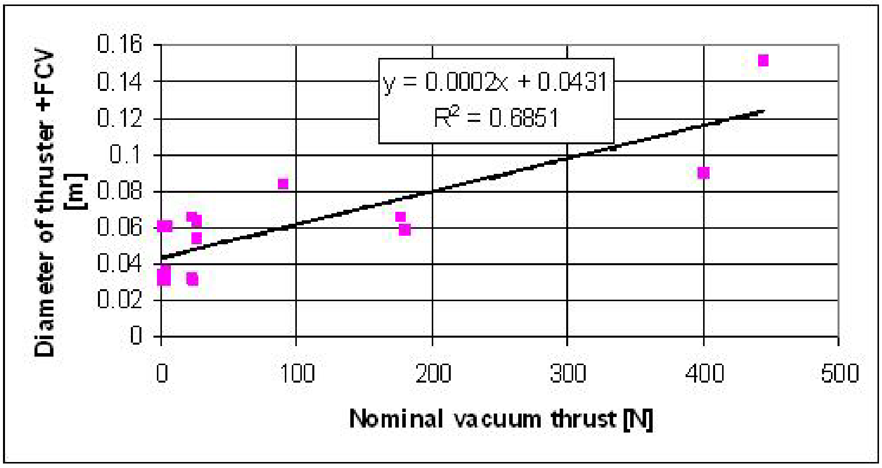


Figure 5: Size estimation of small monopropellant rocket thrusters

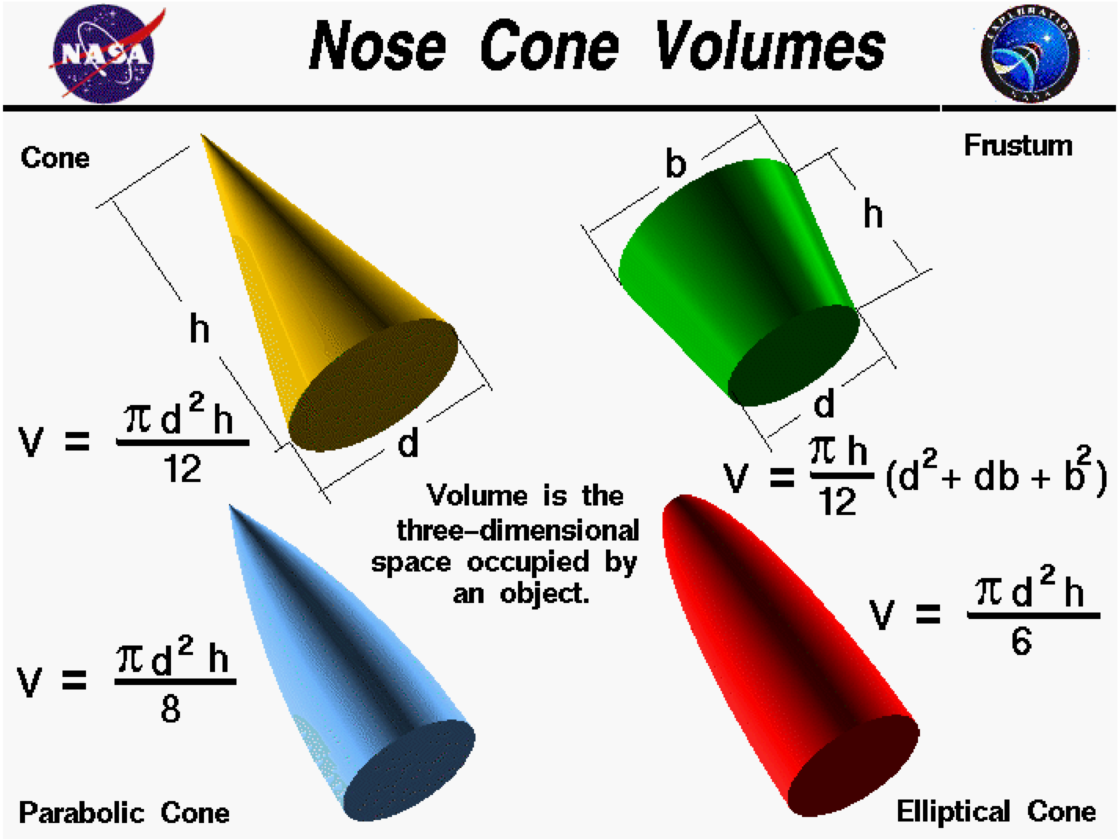
Discussion of results

The results confirm that length and diameter to some extent are correlated with thrust. However, the results also indicate that there is considerable spread about the curve fit. The cause of the spread is attributed to differences in amongst others expansion ratio (larger expansion ratio leads to a longer nozzle), chamber pressure (higher chamber pressure reduces throat area and for a constant expansion ratio also the nozzle exit area), and the propellant combination used for the different engines. When using the curve fit to estimate an envelope this spread should be taken into account by taking adequate margins for the envelope in the design.

References

1. Zandbergen, B., Mass and size estimation relationships of pump fed rocket engines for launch vehicle conceptual design, 6th European Conference for Aeronautics and Space Sciences (EUCASS), Krakow, Poland, July 2015.

Appendix H: Nose Cone volumes



For a **cone**, the distance from the tip to the base is called the **height**. The base is a circle of diameter **d**. The volume **V** of a cone is equal to **pi** (3.14159) times the diameter **d** squared times the height **h** divided by twelve;

V = pi * d^2 * h / 12

A **parabolic cone** has a smooth curved surface and a sharp pointed nose. On the standard cone there is an edge between the nose and the cylinder which forms the body of the rocket. But on the parabolic cone, the surface comes into the base with a slope equal to zero. There is no edge between the parabolic nose cone and the cylindrical rocket body . The equation for the volume is **pi** times the diameter **d** squared times the height **h** divided by eight;

V = pi * d^2 * h / 8

An **elliptical cone** is similar to the parabolic cone except the nose is blunted and not sharp. If the nose cone were cut in half, perpendicular to the base, the resulting cross-section would be half of an ellipse. The equation for the volume is **pi** times the diameter **d** squared times the height **h** divided by six;

The **frustum** of a cone is formed if the tip is cut off parallel to the base. Frustum shapes occur often on model rockets as **fairings** between cylindrical sections of the body. The equation for the volume is **pi** times the height **h** divided by twelve times the quantity: base diameter **b** squared plus base diameter times cut diameter **d** plus cut diameter squared:

V = (pi * h / 12) * (d^2 + d*b + b^2)

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Errata document:

Page II, footnote at bottom of page: “vehicle” should be “vehicles”

Page 26: Replace extend” by “extent”.

Page 27:

In example calculation shown on page, the structures mass reported should be 7201 kg instead of 72015 kg.

Page 115, footnote: 0,02% should read 0.02%.

Page 121, VEB mass estimation:

R^2 value should read 0.6381 instead of 0.9381.

Additionally, it is mentioned that from the ESA launch vehicle catalogue follows an identical VEB mass reported for Ariane 1, -2 and -3, whereas the dry mass of these vehicles differs. This leads to some doubts on the exactness of the relation obtained.

Page 122”, footnote 40: Remove “In that case,”