Estimating launch vehicle trajectories and atmospheric emissions

by

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Submitted to the Department of Aeronautics and Astronautics in partial fulfillment of the requirement for the degree of

Master of Science in Aeronautics and Astronautics

at the

MASSACHUSETTS INSTITUTE OF TECHNOLOGY

May 2021

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Abstract

Launch vehicles enable Earth observation, navigation or space exploration. In doing so, they cause direct anthropogenic emissions in the troposphere and above. This comes with an environmental cost. They can emit carbon dioxide (CO_2), water vapor, chlorine, aluminum oxide, black carbon and nitrogen oxides that lead to atmospheric changes such as ozone depletion. Historically, because launches were part of national security concerns, rockets have not been subjected to environmental regulation. The few studies in the 1990s due to the decreasing number of launches have helped rockets avoid any policy regulation. However, the development of commercial launches and the evolution of engine designs has created a need to reassess the impact of rocket launches. My work presents an inventory of stoichiometric emissions of rocket launches between the years 2009 and 2018. I first compile all publicly available data of launches between 2009 and 2018, then I design a program to simulate the trajectory of any launch vehicle under an altitude of 100 km. This model gives a profile of fuel burn and stoichiometric emissions as a function of altitude for many launch vehicles. The exhaust products of interest are CO_2, water vapor, chlorine and aluminum oxide. Between 2009 and 2018, 140.5 kt of CO_2 were emitted in the atmosphere, 78.9 kt of water vapor, 5 kt of chlorine and 7.8 kt of alumina were emitted above the tropopause. The increase in the number of launches has made CO_2 emissions grow by 73% between 2009 and 2018, while water and chlorine emissions have decreased by 25 and 58% since 2009 because of the retirement of the Space Shuttle. The rise in kerosene-fueled rockets launches is making CO_2 emissions increase faster than the number of launches and suggests that water vapor emissions are going to increase again. Launch vehicles are compared and reveal that trade-offs are necessary to minimize the different emissions.

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Acknowledgements

I would like to thank my advisor, Prof. Steven Barrett, for giving me the opportunity to join the Laboratory for Aviation and the Environment and for supporting my work from the other end of the world.

I am especially grateful to Dr. Seb Eastham for supervising my research with patience and enthusiasm, and for always finding time to help me when I needed it. I would like to also thank Dr. Sabnis for sharing his experience and his insights on the project.

My time at LAE would not have been the same without the people in it. Thank you to the lab leadership for being understanding of the challenges we faced, especially during the pandemic, and to my lab-mates for your presence, your help, and your jokes.

Thank you to all the people at MIT I have met who have made me feel at home in Cambridge and who have made my stay enjoyable despite the pandemic.

Thank you to my friends for supporting and listening to me on both sides of the Atlantic. Thank you to my family for their love, their advice and their unconditional support.

Contents

Chapter 1: Introduction

Launch vehicles are used to send into Space satellites for Earth observation, navigation or communication, and spacecraft for space explorations with probes or humans. They are designed to send payloads with a specific mass to a defined orbit the cheapest way possible.

Launch vehicles designs have undergone a great deal of innovation to answer those demands. The American launch vehicles that were popular thirty years ago such as the Titan IV and the Space Shuttle are now retired. Chinese and Indian space programs have grown very fast over the past decades to compete with other countries. Crewed missions are being planned to the Moon and to Mars. Private companies such as SpaceX or Rocket Lab have developed cost-effective launchers with reusable stages that have allowed a growth in commercial launches. As a result, the number of launches per year started rising again around 2006 (Figure 1) and is expected to increase even more.

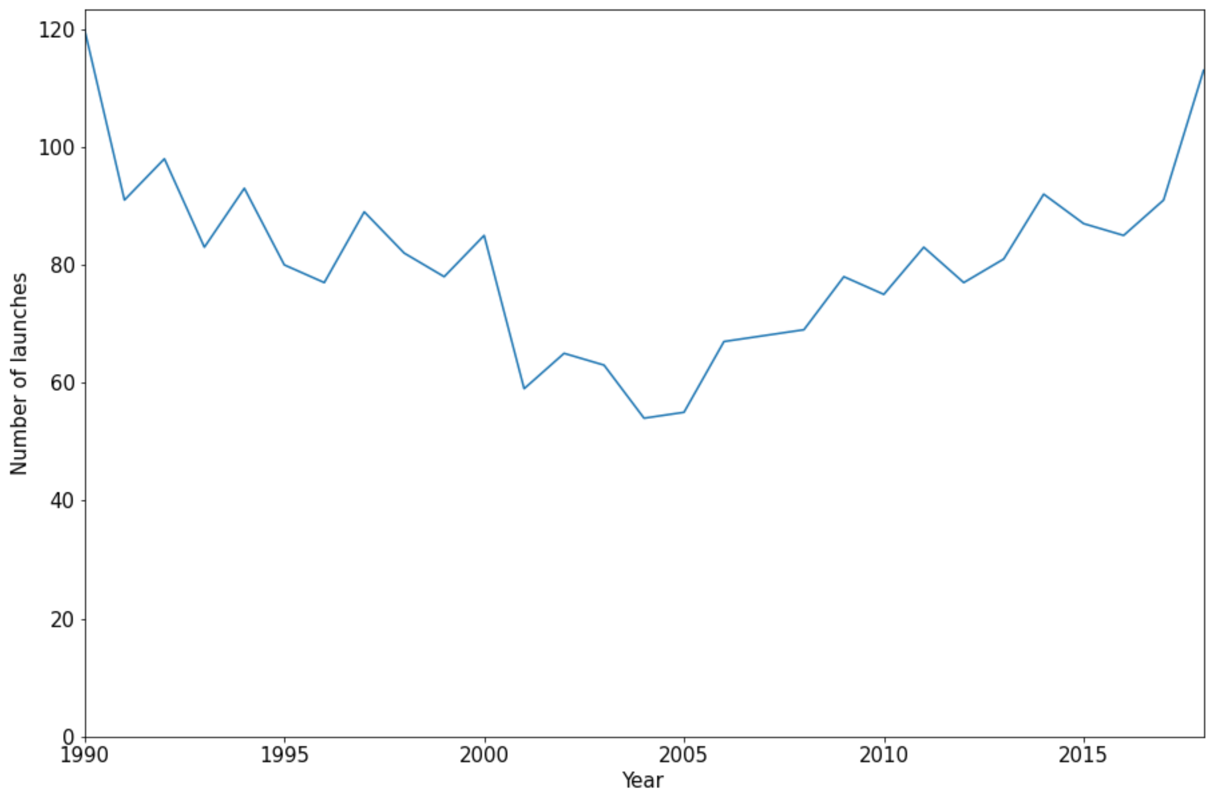


Figure 1 Evolution of the number of launches between 2009 and 2018 derived from [1]

However, it is known that rocket exhaust products can change the atmosphere, and that their impact depends on the propellants used by the rocket. Those propellants emit different products, such as carbon dioxide (CO_2), water vapor (H_2O), alumina (Al_2O_3) and chlorine. Incomplete combustion can also produce carbon soot, sulfate or nitrogen oxides (NO_x). Carbon soot, also called black carbon (BC) because of its dark color, is a solid substance consisting in a mixture of carbon and hydrogen.

The first atmospheric impact of interest is the ozone layer. It is a zone in the stratosphere (mostly between 15 and 30 km) where an important concentration of ozone absorbs the Sun's ultraviolet radiations. Without this layer, life could not exist on Earth.

Species such as chlorine, NOx, or H_2O in the stratosphere originate catalytic cycles that can generate ozone destruction. The chemistry being different in different layers of the atmosphere, the amount of depletion will depend on the altitude of the emissions. Chlorine can also be created by a chemical reaction on the surface area of alumina, an aerosol. This reaction depends on the aerosol's main properties such as size distribution, which is poorly understood. [2]

Jackman et al. (1998) [3] calculated that on a global scale, HCl and alumina contributed together to an annually averaged global ozone decrease by 0.025%, updated later to 0.03% by Ross et al. (2009) [4]. Those studies had qualified rocket ozone depletion as "not significant", but the changes of engines, launch vehicles and the increase in launch frequency call to re-evaluate that concern. [4]

Rocket launch emissions are also known to contribute to climate change because of CO_2, H_2O, BC and alumina. CO_2 and water vapor are greenhouse gases that trap outgoing thermal radiation. BC and alumina are solid particles that can either absorb solar flux, or trap outgoing radiations. The net change in the energy system due to those perturbations is called radiative forcing (RF). A positive RF means that the perturbation causes an increase in the temperature of the atmosphere. Greenhouse gases RF are necessarily positive, but this is not always the case for the other products. Ross and Sheaffer (2014) [5] calculated that both BC and alumina induced positive RF, with BC contributing to 70 % of total rocket RF, and alumina 28%. The RF calculation will vary with the altitude, because of the changes in species lifetime and chemistry.

Climate and ozone loss are also interlinked. Greenhouse gases lead to stratospheric cooling, which can cause not only troposphere warming but an increase in transport of water vapor from the troposphere to the stratosphere. This change in water vapor cycle can affect ozone loss, especially in the Arctic [6] and further cool down the stratosphere.

Rocket launches could finally lead to the formation of clouds observed near the poles during their summers: the mesospheric clouds. Defined as the highest clouds in the atmosphere, they are believed to form from water vapor in the mesosphere (between 50 and 80 km of altitude). Stevens et al. (2005) [7] determined that a single Space Shuttle launch led to a 10-20% increase in the mass of polar mesospheric clouds during the 2002-2003 summer. Mesospheric cloud RF is also very poorly understood [8].

Rockets are potentially harmful to the environment, and their current development is not guided by atmospheric impact concerns but costs. To estimate the impact of rocket launches, emissions need to be known as a function of altitude first. While emissions from incomplete engine combustion are poorly understood, stoichiometric emissions can be calculated. Previous works have either tried to calculate impacts without emissions profiles [5], or obtained those profiles for only some launch vehicle types. [8] The Aerospace Corporation also released a report in 2014 to estimate the worldwide launch emissions and their impact on climate and ozone depletion for launchers around the world, but they simply identified the different layers from launch profiles publicly available. [9]

This work presents the first comprehensive profile of stoichiometric emissions for all the rocket launches between 2009 and 2018. By compiling all publicly available data of launches between 2009 and 2018, we design a program to simulate the trajectory of any launch vehicle under an altitude of 100 km. The 100 km limit was chosen because emissions above are considered to fall back down in the atmosphere. Appendix A provides more information on this decision. To our knowledge this is the first model to give a comprehensive profile of fuel burn and stoichiometric emissions as a function of altitude for all types of launch vehicles, and to give a comparison of different mission achievements.

This thesis will first introduce in Chapter 2 a launch program that calculates fuel burn as a function of altitude for a given mission. Then Chapter 3 will explain how stoichiometric emissions are derived from those missions. Chapter 4 will present the inventory of the emissions from rocket launches between 2009 and 2018. Chapter 5 will compare how emissions vary from different launch vehicles and targeted missions. Finally, Chapter 6 will conclude this thesis.

Chapter 2: Modeling launch trajectories

Stoichiometric exhaust products calculations must be a function of the altitude of their emissions to be able to properly estimate their impact. A program is designed to simulate any rocket launch and extract fuel consumption as a function of altitude.

2.1 Typical launch timeline

We describe the different steps of a rocket launch that will help guide building the program, as displayed in Figure 2. When a rocket launches, it takes off vertically to clear off the launch site: its pitch angle (angle between the rocket's axis and the horizon) is 90° (a). Then the launch vehicle makes a maneuver called the gravity turn or pitch over. (b) The thrust engines are gimballed to suddenly decrease the pitch angle of an angle called the kick angle. This maneuver creates a torque that uses gravity to make the vehicle pivot and acquire horizontal velocity. The gravity turn has several benefits: thrust is not used to change the rocket direction, and it allows decrease in the aerodynamical effects on the rocket.

Once the rocket has cleared off the denser atmosphere and aerodynamical forces are negligible, pitch angles may be controlled through open or closed loop. (c) An open loop guidance is based on the vehicle relative velocity and uses a table with angle commands for corresponding velocity. A closed loop system computes where the vehicle ought to be in the sky to hit the targeted performance. It corrects error from possible previous open loop. There is no general rule or theory for manufacturers to follow, as their goal is to reach payload insertion conditions. From observation, this phase usually starts at Main Engine Cut Off (MECO).

The payload fairing, used to protect the payload, separates at a time determined by the manufacturer when it is no longer of use (d). As the rocket gains altitude and speed, it jettisons its stages one after another until separation (e). Separation with the payload can occur in varied ways: the rocket can inject the payload in its final orbit or in an intermediate orbit and let the payload propel itself to its final destination. The rocket can also reach several orbits before the injection orbit in accordance with its performance. In that case, it often requires an upper stage that will burn to change orbit. It can ignite several times during the launch.

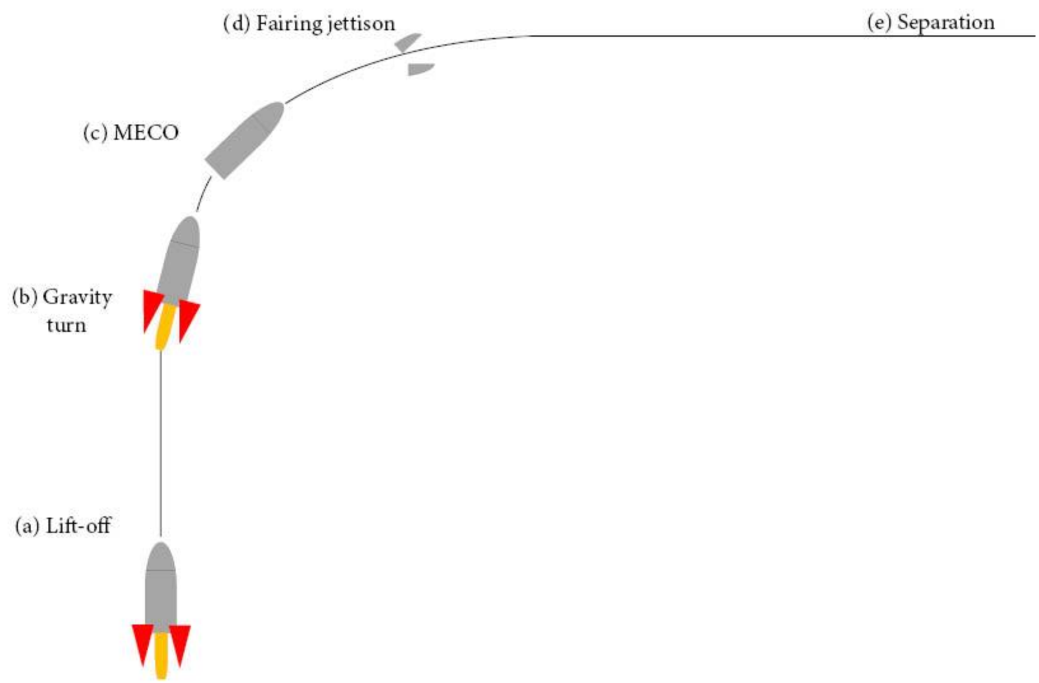


Figure 2 Different steps of a rocket launch.

2.2 Physical description of the program

We define 4 objects in Python illustrated in Figure 3:

- Propulsion: describes the propulsion systems being simulated, liquid or solid. It can refer to the main engine or a vernier, which is used to help with attitude control. The inputs are propellant and structure mass, burn times and thrust inputs. When applicable, it also provides throttling profile.
- Stage: describes the propulsion systems burning at the same time, for parallel staging. It is only applied on the stages that are going to be simulated burning. Its inputs are the list of propellant objects that compose this stage, its length and its diameter.
- Rocket: describes the complete launch vehicle. Its inputs are a list of the different stages, extra mass for the upper stages that are not being simulated, and fairing parameters to calculate drag.
- Launch: describes the conditions of the specific launch being simulated. The inputs are the payload mass, targeted orbit, launch site and initial conditions. Initial conditions are as follow: initial velocity of 0 m/s, initial flight path angle of 90°, initial height and downrange distance of 0 m each.

Those inputs are taken from a database built on Excel and imported in Python through the library pandas. Data was collected through available sources in books and online. Since many sources are not official, and since some launch vehicles have kept modifying their engines, we tried to find values validated by different sources when possible. References are provided in Appendix D.

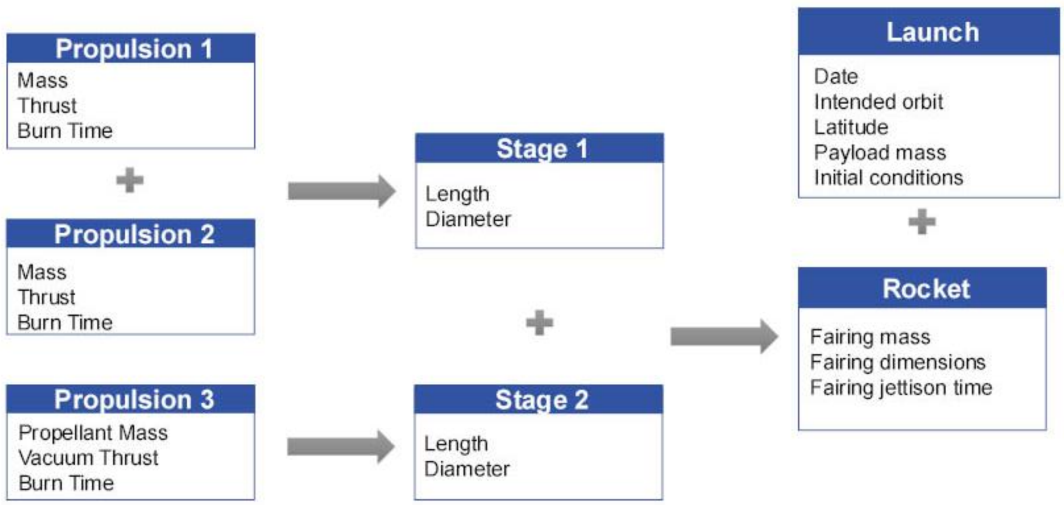


Figure 3 Construction of a mission, defined by a Launch object and a Rocket object. The latter is built on Stage and Propulsion objects.

2.3 Mathematical formulation of the model

The aim of the program is to obtain propellant consumption as a function of altitude of a launch. To that end, it simulates the different steps of a rocket launch and calculates propellant mass flow rate, rocket altitude and vertical velocity as a function of time. They depend on other physical quantities such as speed. Changes in speed *v*, altitude *h* and downrange distance *x* follow the ordinary differential equations (ODEs) taken from [10]

$$\frac{dv}{dt} = \frac{T\cos\alpha - D}{m} - \frac{\mu}{(h + R_E)^2}\sin\gamma\,, \tag{2.1}$$

$$\frac{dh}{dt} = v\sin\gamma\,, \tag{2.2}$$

$$\frac{dx}{dt} = v\cos\gamma\ ; \tag{2.3}$$

where T is the thrust, D the drag, R_E the Earth radius, m the rocket mass, μ the standard gravitational parameter, α the angle of attack and γ the flight path angle. The angle of attack represents the angle between the thrust direction (approximated as the direction of the vehicle) and the velocity direction, while the flight path angle is the angle between velocity and the horizon. Together, they form the pitch angle β

$$\beta = \alpha + \gamma \tag{2.4}$$

We assume that the angle of attack is zero during take-off and gravity turn, as aerodynamic loads are limited, and lift can be neglected. The flight path angle is initially at 90° and maintains this value until the pitch over. Afterwards, it starts decreasing by following the following ODE

$$\frac{d\gamma}{dt} = \frac{T\sin\alpha}{mv} + \left(\frac{v}{R_E + h} - \frac{\mu}{(R_E + h)^2 v}\right) \tag{2.5}$$

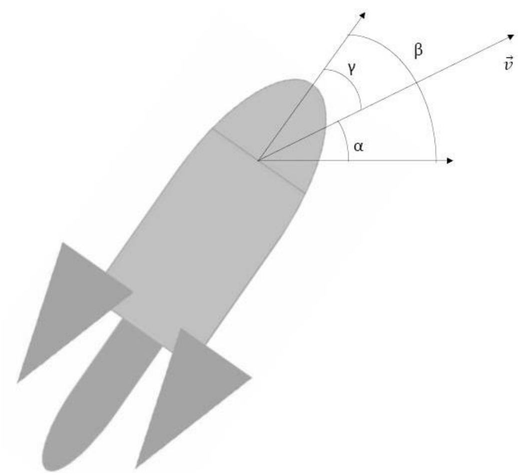


Figure 4 Representation of the angle of attack α, the flight path angle γ and the pitch angle β. $\vec{v}$ is the velocity of the rocket.

The ODEs are all coupled and can be numerically solved through a 4th order Runge-Kutta method.

In those equations, three variables are introduced that do not have an expression to calculate them: drag, thrust and mass, or more precisely mass flow rate.

Drag coefficient is here estimated using Fleeman's *Missile Design and System Engineering* (2012) [11]

$$C_D = \begin{cases} 0.053\dfrac{l}{d}\left(\dfrac{M}{0.5 \times \rho v^2} l\right)^{0.2} \; if\ M < 1 \\ \left(1.586 + \dfrac{1.834}{M^2}\right)\left(\tan^{-1}\dfrac{0.5}{\frac{l_n}{d_n}}\right)^{1.69}\left(\dfrac{{d_f}^2 - {d_n}^2}{{d_f}^2}\right) + 0.665\left(1.59 + \dfrac{1.83}{M^2}\right)\dfrac{{d_n}^2}{{d_f}^2} \; else; \end{cases} \tag{2.6}$$

where M is the Mach number, l the rocket length, d the rocket diameter, d_f the fairing diameter, l_n the nose fairing length and d_n the nose fairing diameter. Skin friction drag is predominant for subsonic conditions while for supersonic conditions wave drag is predominant. As for thrust and mass flow rate, calculation will be different depending on whether the propellant used is liquid or solid. For both, the propellant mass needs to be provided.

2.4 Simulating solid and liquid propulsion

Rockets can use propellants that are either in a liquid state or a solid state. For either type of propulsion, thrust can be defined as

$$T = \dot{m}V_e + (P_e - P_a)A_e \; ; \tag{2.7}$$

with T the thrust, $\dot{m}$ the massflow rate at the exit of the nozzle, V_e the exhaust velocity, P_e the exhaust pressure, P_a the atmospheric pressure and A_e the exit area [12]. The exhaust velocity can be approximated as constant throughout the whole launch, but that is not always true for mass flow rate.

2.2.1 Liquid propulsion

In this case, the mass flow rate is constant except when there is throttling, which reduces the amount of mass leaving the chamber every second. If we write vacuum $_{vac}$, and sea level as $_{sl}$, we can estimate

$$T_{vac} = \dot{m}V_e + P_eA_e \tag{2.8}$$

$$T_{sl} = \dot{m}V_e + (P_e - P_{sl})A_e \tag{2.9}$$

By combining the two equations, we obtain a simpler expression of thrust

$$T = T_{vac} - \frac{T_{vac} - T_{sl}}{P_{sl}} P_a \tag{2.10}$$

Manufacturers often provide sea level and vacuum thrust values for liquid propulsion systems. P_{sl} is the atmospheric pressure at sea level and is a constant. Atmospheric pressure varies according to altitude and time of the year. The date of launch is a given input, and altitude is calculated in the program. We use the CIRA 86 data to identify the atmospheric pressure for a given month and altitude. The propulsion system's thrust is then calculated using each engine's sea level and vacuum thrusts, altitude, month

$$T(h, \text{month}) = T_{vac} - \frac{T_{vac} - T_{sl}}{P_{sl}} P(h, \text{month}) \tag{2.11}$$

In addition to that, the initial propellant mass M_p and the burn time t_b give us the mass flow rate for liquid propulsion

$$\dot{m} = -\frac{M_p}{t_b} \tag{2.12}$$

However, some engines can be throttled: the mass flow rate is reduced, and in turn, thrust and acceleration. This can happen before the vehicle reaches the maximum dynamic pressure (max Q), to limit the impact of aerodynamical forces on the rocket. Throttle is introduced as the percentage of mass flow rate reduction *a(t).* If *a* is 1, then there is no throttling. Throttle is defined at each timestep h. The new expression of mass flow rate for a propulsion system burning propellant between t_l and t_m, is

$$\dot{m}(t) = -a(t) \frac{M_p}{h t_b \sum_{k=l}^{k=m} x(t_k)} \tag{2.13}$$

The final expression of thrust is

$$T(h, \text{month}) = a(t) \left[T_{vac} - \frac{T_{vac} - T_{sl}}{P_{sl}} P(h, \text{month}) \right] \tag{2.14}$$

The expression of the mass flow rates ensures that exactly M_p kg of propellant will have been burned by the engine.

2.2.2 Solid propulsion

Solid propulsion physics is described in more details in Appendix B. We approximate solid propulsion thrust and mass flow rate by using prescribed vacuum thrust profiles. Mass flow rate can be approximated by

$$\dot{m} = \frac{T_{vac}(t)}{gc}; \tag{2.15}$$

with *c* the specific impulse (in s), *g* the gravity constant and $T_{vac}(t)$ the vacuum thrust. With access to the exterior area, A_e, thrust can be estimated as a function of altitude as

$$T = T_{vac} - P_a A_e \tag{2.16}$$

Contrary to liquid propulsion, the mass flow rate value is overestimated, so the total propellant mass consumed is often greater than M_p.

Engines using solid propulsion are called Solid Rocket Motors (SRMs). When they are used as strap-on boosters, they are called Solid Rocket Boosters (SRBs).

2.5 Approximating missing or incomplete data

The program relies on many inputs that have been described in the previous subsections. Most of this data is obtained from publicly available sources, but some of it is not accessible. This can involve inputs that are missing from only a few launch vehicles, and there is nothing to be done. It can also be parameters hard to find for any launcher. We describe here how we obtain the latter.

2.5.1 Propellant mass

The propellant mass burned by each engine is essential to derive emissions. The value used in the program is usually given in payload user guides, published by manufacturers. This value is the maximum capacity of the tanks. But depending on the payload mass and the targeted orbit, less propellant mass may be required, and adjustments need to be made. We understand that manufacturers have different ways of facing this problem: adding a booster, choosing a launcher from the same family with different performances, or making the upper stages consume less propellant. In that case, extra propellant could be vented in the atmosphere, or the tank may be less filled. Since this is more likely to occur above 100 km and that the difference in propellant consumption will be less than 1%, we set propellant mass to the maximum capacity.

2.5.2 Latitude

Knowing the latitude λ of the emissions can be chemically important to estimate atmospheric impacts, especially from water vapor. Its evolution can theoretically be calculated alongside the longitude Λ solving

$$\frac{d\Lambda}{dt} = \frac{v\cos\gamma\sin\chi}{(R_E + \mathrm{h})\cos\lambda}, \tag{2.17}$$

$$\frac{d\lambda}{dt} = \frac{v\cos\gamma\cos\chi}{(R_E + \mathrm{h})}; \tag{2.18}$$

where χ is the launch azimuth. The launch azimuth is an indication of the direction taken by the rocket towards orbit relatively to the north. It is defined by the targeted orbit of the launch. We first obtain the launch azimuth in inertial space, with no Earth rotation, with angle geometry in three dimensions

$$\cos\lambda \sin\chi_{\text{inert}} = \cos i \; ; \tag{2.19}$$

where *i* is the orbit inclination. Now, taking into consideration Earth rotational speed v_E, the launch needs to produce the speed v_{rot} to reach the orbit velocity v_{orb} which is a function of perigee and apogee

$$v_{rot,x} = \sin\chi_{inert}\, v_{orb} - \cos\lambda\, v_{eq,} \tag{2.20}$$

$$v_{rot,y} = \cos\chi_{inert}\, v_{orb} \; ; \tag{2.21}$$

where v_{eq} is the rotation speed at the Earth equator. The final launch azimuth χ will then be defined as

$$\tan\chi = \frac{v_{rot,x}}{v_{rot,y}} \tag{2.22}$$

The payload orbit is often available, but we do not necessarily have information on the injection orbit or, when applicable, the intermediate orbits targeted by the rocket. However, calculations of downrange distance have shown that below 100 km of altitude, the rocket approximately stays within the same latitude. For the program, we can consider that the latitude of emissions corresponds to the latitude of launch.

2.5.3 Throttling

This parameter only applies to certain liquid propulsion systems.

Some engines have throttling profile provided, such as the Atlas V's Core Booster. Others, like the Falcon 9's Merlin engines, do not, and probably vary from one launch to another. For those that do not have predefined profiles, we decide to impose some limits on possible achievable load factors, based on their Payload Users' Guide. Load factor is defined as the force-to-weight ratio on the rocket

$$L = \frac{T\cos\alpha - D}{mg} \tag{2.23}$$

It can also be called axial acceleration and is expressed in *g* units. For example, SpaceX provides in the Falcon User's Guide [13] the designed load factors for the Falcon 9 and the Falcon Heavy. Restrictions on axial acceleration (or load factor) depend on the lateral acceleration, that is defined as being "*driven by wind gusts, engine gimbal maneuvers, first-stage engine shutdown and other short-duration events*." Since our simulation does not include some of those parameters, we impose a wider limit: the load factor cannot exceed 6 g.

2.5.4 Pitch angle

Equations of motion solved numerically are necessary to obtain altitude and velocity and depend on flight path angle. The latter is also determined through an ODE, but also relies on some parameters that need to be optimized: the kick angle and the angle of attack profile.

The kick angle sets how much the engines are gimbaled at the gravity turn: the new flight path angle is the previous angle (usually 90°) minus the kick angle and follows its ODE. Once the rocket has left the densest part of the atmosphere, the angle of attack is no longer constrained to 0. The pitch program is controlled through thrust vector control (TVC), following an open or close loop. Its evolution must be estimated.

One important aspect is the ending time of the simulation. Our simulation is of interest for altitude below 100 km. This point can be reached after MECO (e.g.: Falcon 9) or before (e.g.: Ariane 5 ECA). We try to choose the first staging that occurs above 100 km. In the case of Ariane 5, it means stopping at MECO. For other rocket launches like the Proton M, we stop before the upper stage Briz M ignites. Another perk of doing this is that we do not have to simulate re-startable upper stages who can have burn profiles very specific to a launch.

2.5.4.1 Kick angle

The time of kick does not need to be known with precision. Sometimes we know when the launch vehicle begins its gravity turn, but often we do not. However, since the kick varies with the time of kick and we never know the real value of the kick angle that needs to be optimized, we can decide of an arbitrary time of kick; the kick angle will be estimated accordingly. Therefore, the kick time will not be optimized.

The value of the kick angle is of extreme importance for the first phase of the flight for vertical velocity, as defined in equation (2.5). If you look at a vertical velocity profile, it increases until reaching a peak at MECO. This peak is defined by the kick angle. However, we do not know what determines the value of the kick angle.

We do know that vertical velocity is very sensitive to the kick angle, as illustrated by Figure 5. We compare our vertical velocity with specific kick angles to Ariane 5 ECA's VA242 mission vertical velocity. The latter is obtained by calculating the instantaneous velocity from an altitude profile available online. For different launch vehicles, we find some kick angle values that vary between 0.08 and 18.5°.

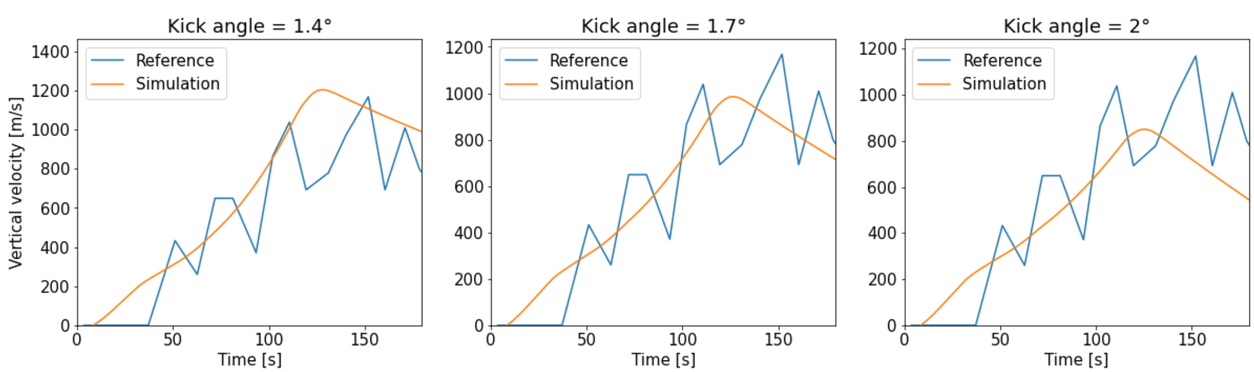


Figure 5 Vertical velocity of Ariane 5 ECA launch with three different kick angles: 1.4 (left), 1.7 (middle), and 2 (right).

2.5.4.2 Angle of attack

After MECO and staging, the angle of attack starts changing. However, there is no rule or equation to follow, as it can be adjusted with open or closed loop, that we do not have. Since it is not possible to do so, the evolution of the angle of attack must be estimated.

Several documents mention a pitch rate. It led to the decision that, because the launch vehicle after the gravity turn tends to become horizontal, the pitch would decrease at a certain rate *b*. The angle of attack would be the difference between the pitch and the flight path angle. This gives the benefit of having only one parameter *b* to optimize in the expression of pitch angle.

$$\beta(t) = \begin{cases} \gamma(t) \; if \; t \leq t_0 \\ \gamma(t_0) - b(t - t_0) \; else \end{cases}; \tag{2.24}$$

with t_0 the time of MECO. For most cases, the pitch rate evolution does not matter on an important portion of the altitude below 100 km.

2.5.4.3 Optimization

Optimization problems usually add the angles to a list of parameters optimized to minimize propellant mass or maximize payload mass injected into a nominal target orbit, but that is not what we are trying to do as we want to imitate real launches. We first select some launches to test the feasibility of the work. We obtain the kick angle and pitch rate by minimizing the error in altitude and speed profile, speed being defined as the magnitude of velocity. Altitude and speed are chosen because their profiles are more likely to be available and that they are the two main physical quantities to describe a trajectory. Given references, we minimize the cost function

$$f(U, X) = \frac{1}{n}\sum_{i=1}^{n} \frac{\left|V_{ref,i} - V_{sim,i}\right|}{V_{ref,i}} + \frac{1}{n}\sum_{i=1}^{n} \frac{\left|H_{ref,i} - H_{sim,i}\right|}{H_{ref,i}}; \tag{2.25}$$

where *U* are the parameters, *X* the other inputs. With 17 launches we get altitude and speed profiles as close as possible. Once this is done, we simulate all the launches of the launch vehicles we have a reference of for the decade of reference. The profile can change because of changes in payload mass and launch site. We provide by default the kick angle and pitch rate that we found with our optimizations, and we simulate altitude profile below 100 km. If the payload is too heavy compared to our reference (for example because of the use of an upper stage), the kick angle can be too important and cause the rocket to crash before reaching 100 km. We decide to reduce the kick angle until the launch vehicle reaches 100 km before 260 seconds of launch. This time was decided arbitrarily from observation of all launches of interest.

2.6 Different steps of the program

The program, written in Python, aims at simulating a given mission and provide propellant mass flow rate, rocket altitude and vertical velocity as a function of time. The previous subsections have highlighted the key elements to make it succeed. We combine them together to build the simulation, described below, and link them to the launch timeline defined in 2.1.

At every timestep *h*=1 the differential equations defined in 2.3 are solved through a 4th order Runge-Kutta method. Mass flow rate and thrust are the sum of respectively the mass flow rate and thrust of every propulsion system burning at time *t*. During the first part of the launch, flight path angle is set to 90 (a). When time reaches kick time mentioned in 5.2.1, gravity turn begins (b). The flight path angle is set to

90-γ_k, with γ_k the kick angle, and follows the ODEs until the end of the simulation. The angle of attack is set to 0 during the first part of the launch and during gravity turns, that ends at MECO (c). After that, we derive it from pitch angle calculated in equation (2.24).

During the launch, the mass of the rocket keeps decreasing as propellant is being burned and staging occurs. Propellant mass flow rate is calculated using the formulas derived in 2.5. Remaining mass is obtained using the total mass flow rate

$$m(t+h) = m(t) - h \times \dot{m}(t) \tag{2.26}$$

When the propulsion system cuts off (reaching burn out), we set $\dot{m}(t) = 0$ until the propulsion system is jettisoned. We subtract structural mass from the rocket mass at stage jettison, and the payload fairing mass at fairing jettison (d). For emission calculations, we also keep a list of separated mass flow rates corresponding to each propulsion system, necessary for emission estimations as they use different propellants with different properties.

The simulation runs until the last propulsion system to be simulated either cuts off or is jettisoned. As we are only interested in the launch profile below 100 km, when they are not needed, upper stages are either not simulated or only their first burn is. We still use their masses to simulate the launch correctly and to estimate the total emissions above 100 km.

Finally, we optimize the parameters described in 2.3.2. We use references of altitude and speed profiles as a function of time when available. From altitude, we extract vertical velocity, that we use to obtain our first guess of kick angle and pitch rates and adjust kick time if needed. The kick time can be provided from data compilation. If not, we set it arbitrarily at t = 15 s. With our first guess, we minimize the cost function from (2.25) using the least square methods from the *optimize* library in Python. Details of the launch vehicles that we simulated are in Appendix C.

2.7 Validation

In this section, we run some simulations to validate the program and the information we can extract from it. First, we need to assert what simulations are to be run.

2.7.1 Launches between 2009 and 2018

We compile the list of all the launches that occurred between 2009 and 2008, with their payload mass and their targeted orbit when possible. We exclude launches which had an accident. An accident is qualified as an unexpected event that prevented the launch to take place the way it was supposed to. It can refer to the explosion of the rocket, staging failure, fairing jettison failure, or injection in the wrong orbit. Figure 6 depicts the cumulative number of launches with no accident.

Each launch vehicle, to be simulated, requires data on the launch vehicle engines, and trajectory and speed profiles**.** The most common launch vehicle, the Soyuz, represents 18% of total launches, and the next three each to 8%. The tenth most common launcher, the Delta IV, contributes to 3%, and the thirty least common launch vehicle each represent less than 1% of total launches**.** As we can see on the figure, the launch vehicles of interest represent 81% of total launches with no accident. For some launches, the payload mass is not publicly available, for security reasons. As a result, we simulate 77% of the launches. The launch vehicles selected are: Soyuz, Proton-M, CZ-3, Atlas-V, Falcon-9, Ariane 5, CZ-4, CZ-2D, PSLV, Delta-IV, H-2A, CZ-2C, Rokot-KM and Space Shuttle. We added the Space Shuttle launches because data is easier to find, and they represent an important launch vehicle of the past decades. While some of those launch vehicles do not have any variants (e.g.: Rokot-KM), others can refer to different versions. The Atlas-V is adjusted with a varied number of solid rocket boosters to accommodate payloads and targeted orbits. This will affect the emissions, as the version with no booster will not emit any chlorine nor alumina, for example.

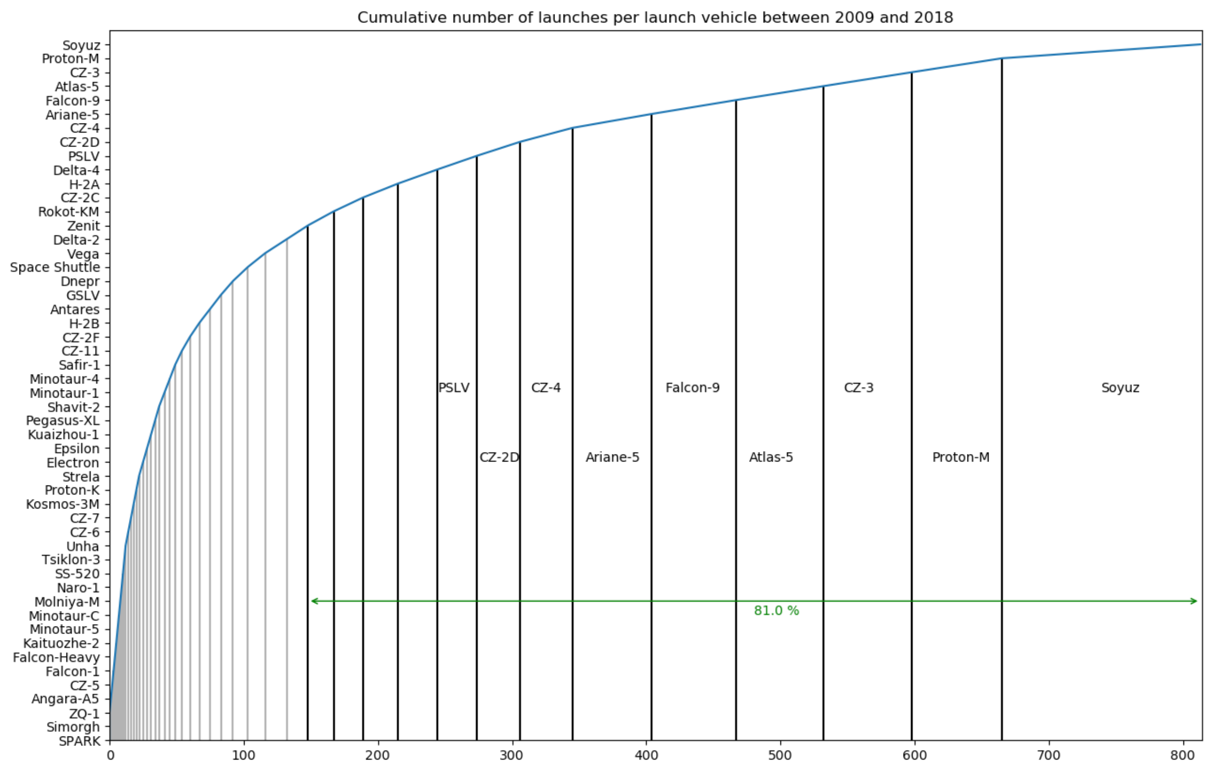


Figure 6 Cumulative number of launches classified by launch vehicles. The arrow represents the launchers that we simulate.

We run simulations of the selected launch vehicles. For each launcher, we select a reference and find the angle parameters that optimize altitude and speed profiles. We give here two examples: the Ariane 5-ECA and Falcon 9 v1.2. The Falcon 9 v1.2 optimization, as explained in Appendix C, is affected by throttling. The cost functions for the Ariane-5ECA and Falcon-9 v1.2 simulations are respectively 0.44 and 0.07.

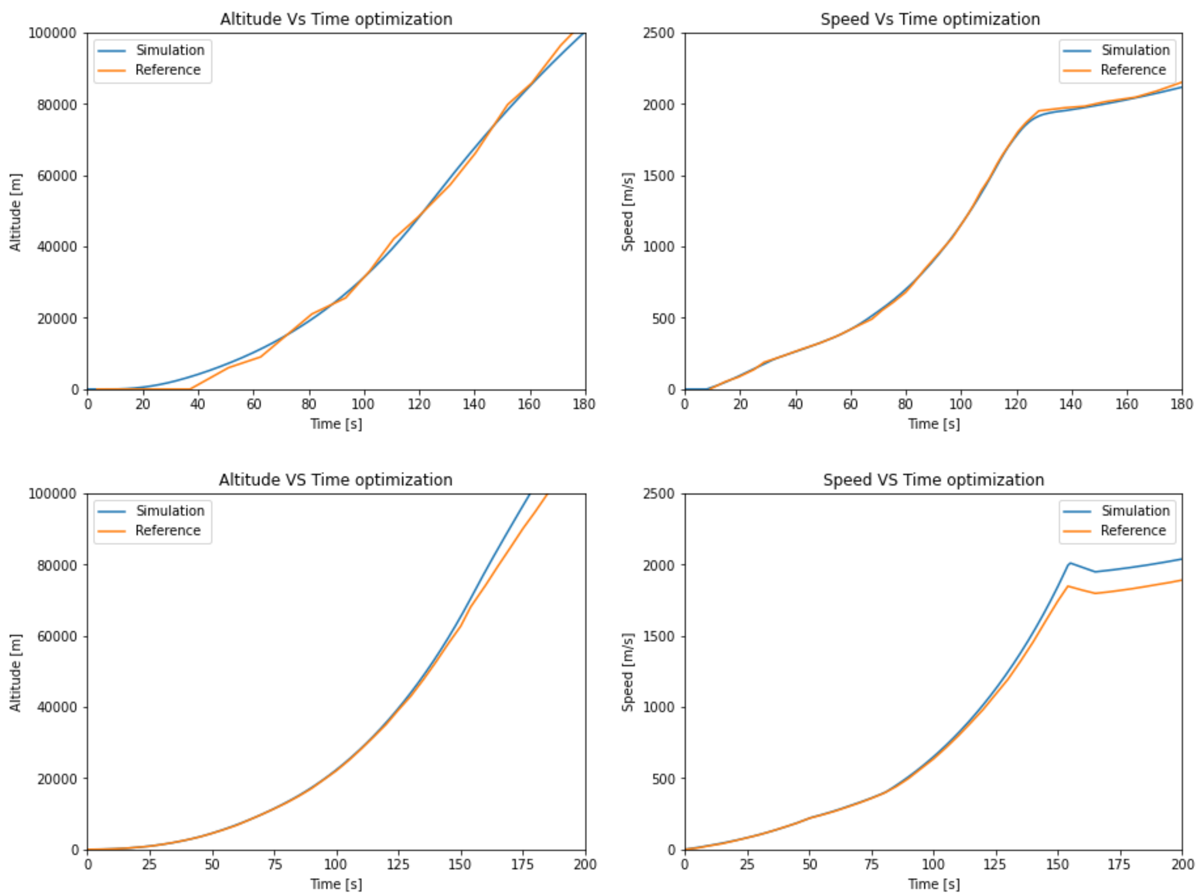


Figure 7 Two examples of simulations with optimization. Top: Ariane-5 ECA. Bottom: Falcon-9 v1.2 mission to the ISS.

Finally, we validate our optimization by simulating all the launches for each launch vehicle category. The main variation between each launch is the payload mass. The heaviest payloads can cause the rocket to crash if the kick angle is too great, so we set the kick angle to decrease until the rocket does not crash before reaching 100 km of altitude.

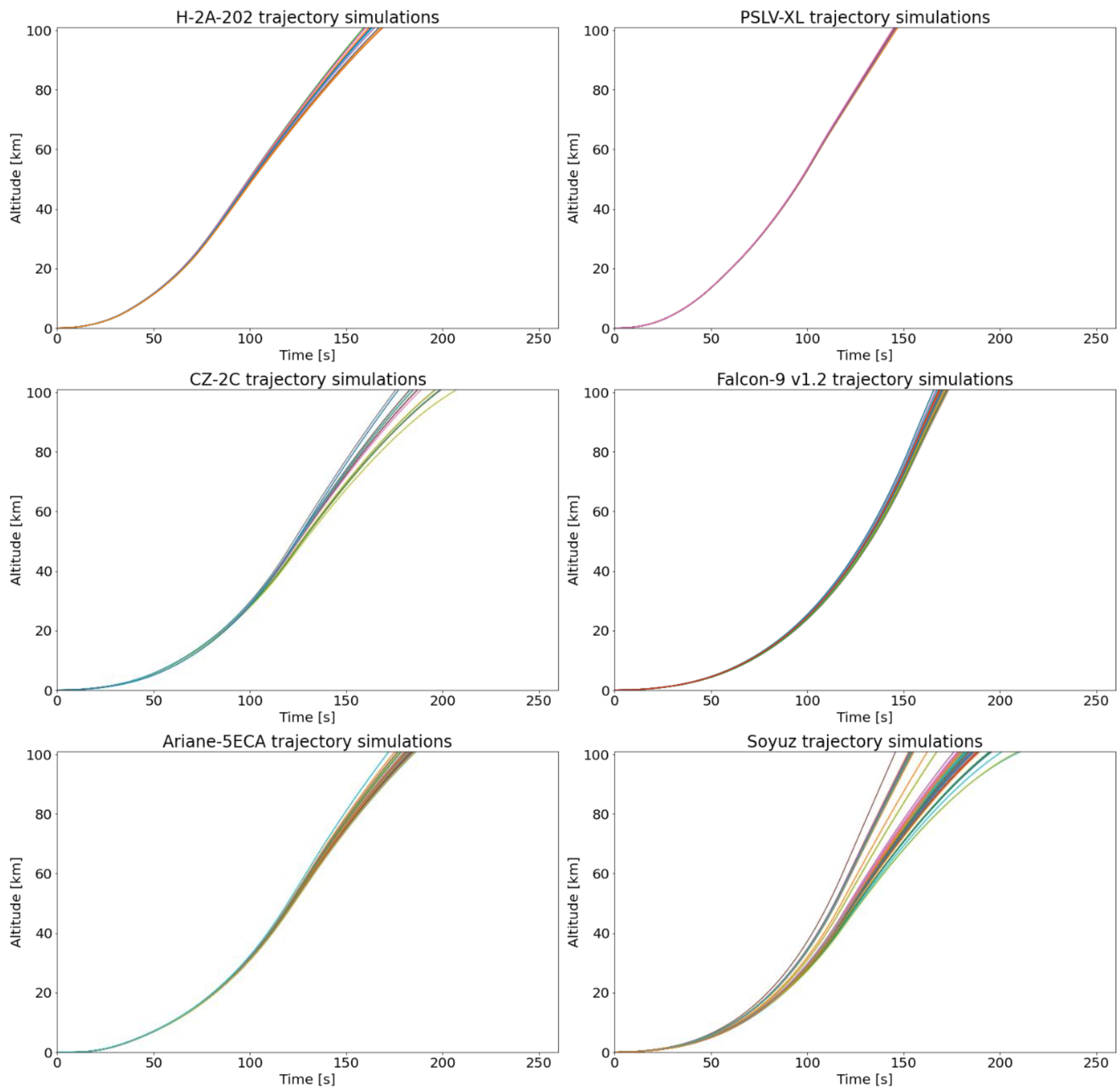


Figure 8 Simulations of altitude profiles for all the launches of six different rockets

We give six examples of validations for launch vehicles of different countries, manufacturers and payload capacities. We see that the discrepancy between launches depends on the difference in payload masses. Some launchers such as the CZ-2C can use upper stages that burn above 100 km, so to our purpose only act like an increase in payload mass. Finally, rockets like the Soyuz also fly different versions with changes in mass and thrust, and upper stages. It explains why there is more variation in the profiles.

2.7.2 Fuel burn by altitude

We derive from our simulated mass flow rate and vertical velocity the propellant mass burned per kilometer. We show as an example the launches of Ariane 5 ECA in Figure 8, and we mark the altitude of the stratopause and the tropopause. The stratopause is fixed at 50 km, and the tropopause is determined from the latitude of the launch and MERRA-2, the atmospheric reanalysis. This plot highlights that propellant emissions are greatest within the first kilometer before decreasing quickly. Simulations of different launchers show that the mass of propellant burned within the first kilometer of the launch is 14% of total propellant, in the tropopause 1.3%, and in the stratopause 0.4%.

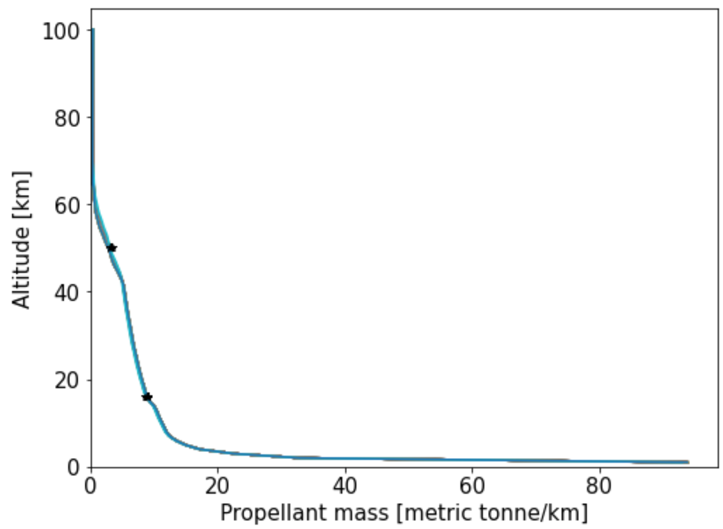


Figure 9 Mass of propellant burned per km as a function of altitude for the Ariane-5 ECA. The dots represent the altitudes of the tropopause (below 20 km) and the stratopause (at 50 km)

We have a program running and providing fuel burn profiles, illustrating that fuel burn decreases exponentially with altitude. But what we need now is to convert this propellant into stoichiometric emissions to obtain an emission inventory and compare missions and rockets.

Chapter 3: Emissions calculation

Once the launch profiles are obtained, we can calculate stoichiometric emissions for different regions of the atmosphere.

3.1 Method of calculation

The mass of an exhaust product is defined as

$$m = EI(X) \times m_f \tag{3.1}$$

with m_f the fuel mass and EI(X) the emission index of species X. The emissions index corresponds to the mass of the species emitted per kilogram of fuel burned. The emission index can be calculated because of matter conservation. The formation of CO_2, H_2O, HCl and Al_2O_3 depend on the conservation of the respective species: C, H_2, Cl and Al. Let us consider the combustion of a fuel represented by a (CH_2) groups. The combustion equation is

$$CH_2 + \frac{3}{2}O_2 \rightarrow CO_2 + H_2O$$

By definition of the emission index

$$\mathrm{EI(CO_2)} = \frac{m_{C_{O2}}}{m_f} = \frac{n_{C_{O2}} \times M_{\mathrm{C_{O2}}}}{n_{CH_2} \times M_{\mathrm{CH_2}}} \tag{3.2}$$

where m_x and M_x represent respectively the mass and molar mass of species X. Conservation of the species C allows us to obtain the emission index

$$\mathrm{EI(CO_2)} = \frac{12 + 2 \times 16}{12 + 2 \times 1} \tag{3.3}$$

We calculate all the emission indices using [9]. Emissions will vary with the propellant used.

Four propellants are used by main engines between 2009 and 2018. We denominate them in the form oxidizer-fuel:

- Liquid oxygen – liquid kerosene, or more specifically, refined petroleum with the designation RP-1
- Liquid oxygen – liquid dihydrogen
- Liquid nitrogen tetroxide (NTO) – liquid hydrazine. Also called earth-storable propellants, we distinguish different fuels: monomethylhydrazine (MMH), or unsymmetrical dimethylhydrazine (UDMH), a derivative of hydrazine, UH25 and Aerozine 50 which are UDMH mixed with hydrazine. The distinction is different because of the variations in emission indices.

- Solid ammonium perchlorate – solid aluminum. The oxidizer and the fuel are mixed with a binder. The binder is often Hydroxyl-terminated polybutadiene (HTPB), except for the Space Shuttle SRM that used Polybutadiene acrylonitrile (PBAN).

Most of the time, the engine specifications only provide total propellant mass. We use the oxidizer/fuel ratio *R* defined as the ratio of oxidizer mass flow rate on fuel mass flow rate. Fuel mass m_f is obtained from propellant mass m_p as

$$m_f = \frac{m_p}{R+1} \tag{3.4}$$

3.2 Carbon dioxide

Carbon dioxide can be formed through RP-1, hydrazine derivatives, or solid propellants. For refined kerosene, we approximate them as a series of ($C_nH_{1.953n}$) groups [14]. The mass of carbon dioxide from RP-1 fuel is

$$\mathrm{m_{CO_2}} = \mathrm{m_{RP-1}} \times \frac{12.01 + 2 \times 16.00}{12.01 + 1.953 \times 1.002} \tag{3.5}$$

The mass of carbon dioxide for UDMH is

$$\mathrm{m_{CO_2}} = \mathrm{m_{UDMH}} \times \frac{2 \times 12.01 + 4 \times 16.00}{60.104} \tag{3.6}$$

In solid propulsion systems, the fuel (ammonium perchlorate) does not produce carbon dioxide. However, a binder (most of the time HTPB) is used to maintain the fuel and oxidizer together in their solid form. CO_2 mass depends on m_p the total mass of propellant and the percentage of ammonium perchlorate x_{ammo} and aluminum x_{al} such that

$$\mathrm{m_{CO_2}} = \mathrm{m_p} \times (1 - x_{\mathrm{ammo}} - x_{\mathrm{al}}) \times \frac{12.01+2\times16}{14.026} \tag{3.7}$$

Since liquid hydrogen does not release any CO_2 during the combustion and that do not quantify liquid hydrogen production emissions, we do not attribute CO_2 emissions to cryogenic engines.

3.3 Water vapor

Some water vapor is formed during the combustion of all the propellants. For RP-1, we use the same approximation as above and obtain

$$\mathrm{m_{H_2O}} = \mathrm{m_{RP-1}} \times \frac{2 \times 1.008 + 16}{12.01 + 1.953 \times 1.008} \times \frac{1.953}{2} \tag{3.8}$$

Water production from Aerozine 50, UDMH, MMH and UH25 is respectively

$$m_{H_2O} = m_{UDMH} \times \frac{8 \times 1.008 + 4 \times 16.00}{60.104} \tag{3.9}$$

For SRMs, both ammonium perchlorate and binders are converted to water vapor, according to the expression

$$\begin{aligned} m_{H_2O} = m_p \times (1 - x_{ammo} - x_{al}) \times \frac{2 \times 1.008 + 16}{14.026} \\ + m_p \times x_{ammo} \times \frac{1.008 + 2 \times 16}{117.492} \end{aligned} \tag{3.10}$$

Water vapor is the main product from liquid hydrogen combustion. Usually, not all the liquid hydrogen is burned, leading to H_2 emitted in the atmosphere. However, after a few hours it ends up oxidized as water [9]. We therefore assume that propellants are in stoichiometric proportions and that water vapor is the only product of hydrogen combustion. We derive, as an expression

$$m_{H_2O} = (\text{Mass } H_2) \times 9 \tag{3.11}$$

3.4 Alumina and chlorine

Finally, alumina and chlorine are dominant products of the immediate exhaust plume of solid rocket engines. The liquid propellants of interest do not emit any of them during combustion. Alumina mass is determined by

$$m_{Al_2O_3} = m_p \times x_{al} \times \frac{2 \times 26.98 + 3 \times 16}{2 \times 26.98} \tag{3.12}$$

Chlorine appears first in the form of hydrochloric acid HCl, whose mass can be derived as

$$m_{HCl} = m_p \times x_{ammo} \times \frac{35.45 + 1.008}{35.45 + 4 \times 16 + 14.01 + 4 \times 1.008} \tag{3.13}$$

Most of the HCl is then converted to chlorine, so it is a good approximation to use this formula to estimate chlorine mass.

Chapter 4: Emissions inventory

We use the program defined in Chapter 2 and the emission calculations from Chapter 3 to create an inventory of rocket launch emissions between 2009 and 2018. We obtain CO_2 and water vapor profiles for all the launches, chlorine and alumina for launchers with solid propulsion systems.

4.1 Global emissions between 2009 and 2018

We first look at global emission between 2009 and 2018. This can help quantify emissions knowing which rockets are launched the most and the least, and which orbital missions have caused the most damage.

We plot the evolution of launches and emissions between 2009 and 2018 (Figure 10), then the total CO_2 and water vapor by launch vehicle (Figure 11), and by targeted orbit (Figure 12).

We differentiate orbit missions: Geostationary Orbit (GEO), Geotransfer Orbit (GTO), Low Earth Orbit (LEO), and Sun-Synchronous Orbit (SSO). GEO is a circular orbit at 35 786 km of altitude, and GTO is a transfer orbit in which satellites are injected to reach GEO by their own means through a circularization maneuver. The distinction between GTO and GEO is important because satellites launched to GTO are usually injected at low altitudes. LEO describes orbits at low altitude, between 200 and 2000 km. SSO is a special LEO, with an altitude around 800 km and a high inclination angle. They are called sun-synchronous because the point of the Earth it observes always has the same lighting.

Finally, since water vapor emission impacts depends on its location, we plot total water vapor emitted as a function of latitude (Figure 13).

Between 2009 and 2018, 140.5 kt of CO_2 were emitted in the atmosphere, 78.9 kt of water vapor and 5 kt of chlorine were emitted above the tropopause. We also calculate that 7.8 kt of alumina were emitted above the tropopause. In comparison, 1034 Tg of CO2 were emitted by aviation in 2018. [15]

The number of launches has increased by 66% since 2009. CO_2 emissions have increased in total by 73%, but they only started increasing faster than the number of launches in 2016 when the Falcon 9 launches started contributing to 25% of total launches. Rockets using kerosene for their main engine – Soyuz, Falcon 9, and Atlas – emitted 70% of total CO_2 in 2018.

However, water and chlorine emissions have decreased since 2009. The retirement of the Space Shuttle in 2011 and a decrease in Ariane 5 reduced their emissions by respectively 44 and 70% in 2013, before increasing again. The emissions in 2018 have declined by respectively 25 and 58% since 2009. The decrease in chlorine emissions is twice as important because only a few launchers emit chlorine. The Space Shuttle had thus a higher impact on chlorine emissions. Kerosene engines contributed to more than twice the emissions of rockets with cryogenic engines in 2018, while cryogenic engine emitted 70% of total water vapor in 2009.

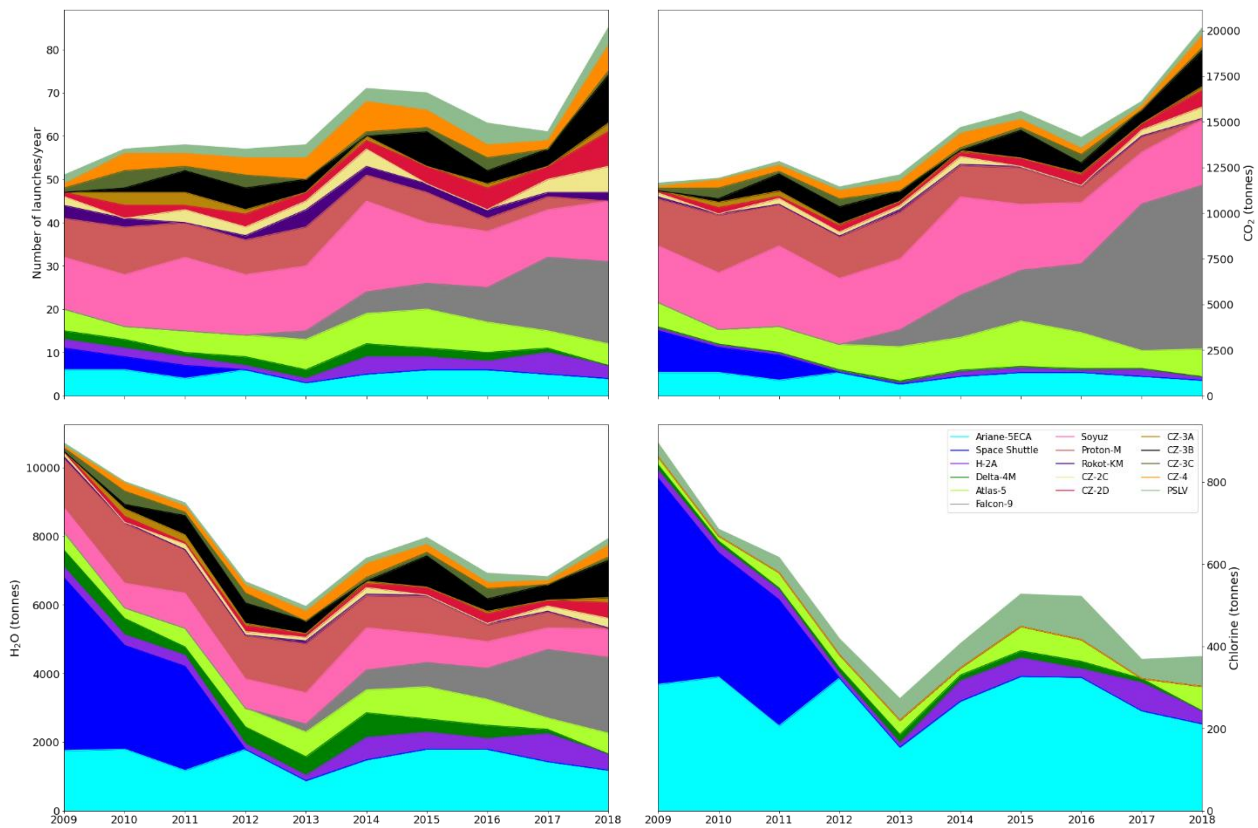


Figure 10 Evolution of number of launches, CO_2, water vapor and chlorine emissions in tonnes. Water vapor and chlorine emissions are only estimated above the tropopause.

Global CO_2 and water emissions are then considered in Figure 11. All launchers have similar profiles: the mass emitted at the first kilometer represents respectively 11 and 16% of total water and CO_2 emitted, and emissions at the stratopause are of the order of 2 to 3% that of take-off. In the upper atmosphere, emission values either remain constant, or water emissions increase while CO_2 decrease. This can happen when the upper stage starts burning, as it is more common to have cryogenic upper stages. The emission contributions and the missions of the main launchers are described in Table 1.

Rocket	Launches (%)	CO_2 emissions (%)	H_2O emissions (%)	Chlorine emissions (%)	Al_2O_3 emissions (%)	Missions to GTO	Missions to LEO	Missions to SSO
Soyuz	22.7	26.3	10.8	0	0	2	96	18
Proton	10.2	13	12.7	0	0	44	0	0
Falcon	9	19	8.4	0	0	26	20	8
Atlas	9	11	7.5	6.3	7	24	11	5
Ariane	8	7.7	19.1	52.9	54.1	48	0	1
Others	41	23	41.5	40.7	38.9	72	44	112

Table 1 Contribution of the 5 main launchers to CO_2, water vapor, chlorine and alumina emissions. The number of missions to GTO, LEO and SSO is also indicated.

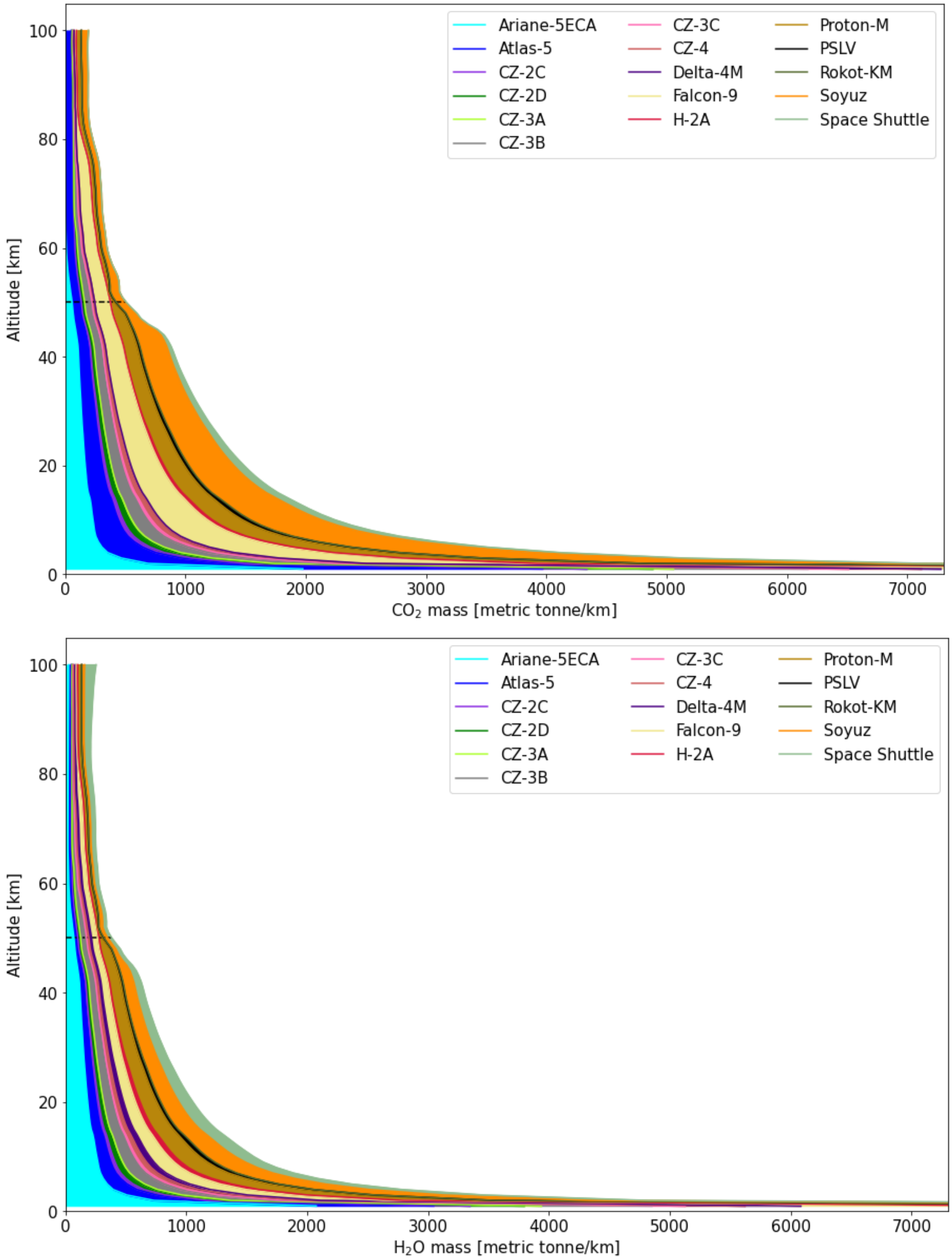
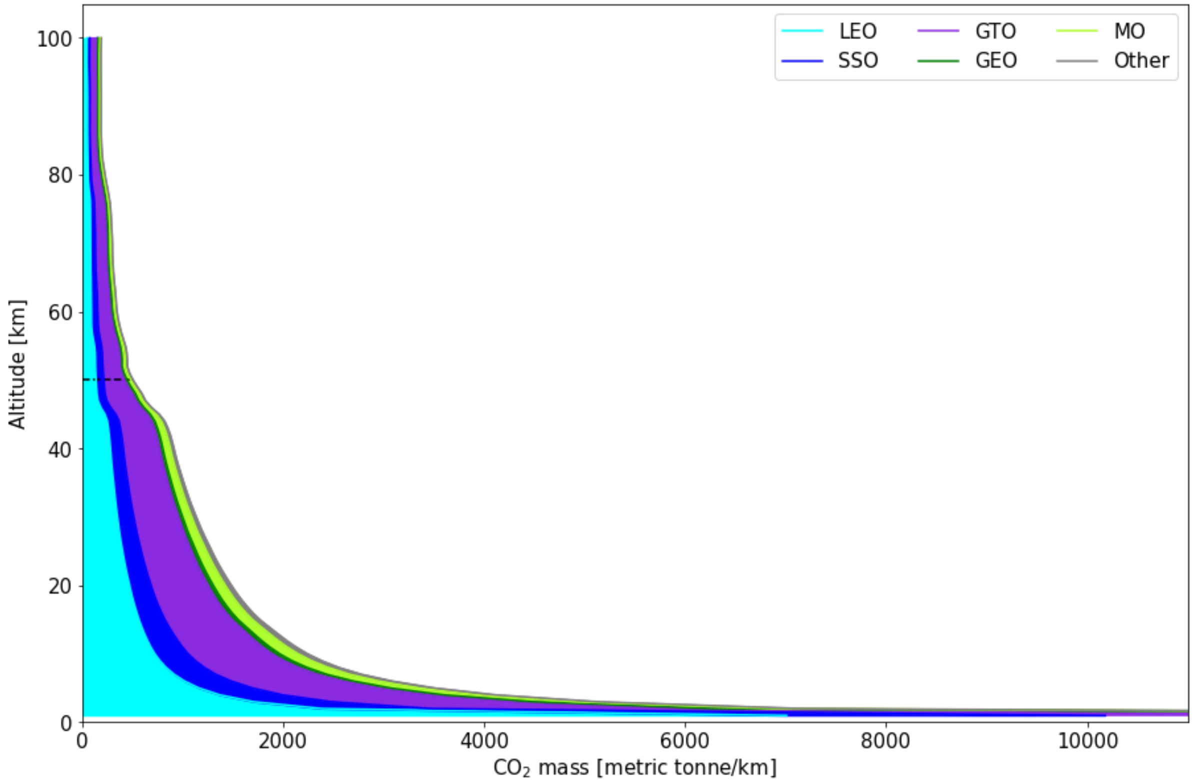


Figure 11 Global emissions of CO_2 and water vapor for the past decade, classified by launch vehicle. For readability purposes, we cut the emissions of the first km. The total CO_2 and H_2O mass emitted in the first km are respectively 20841 tonnes and 14533 tonnes.



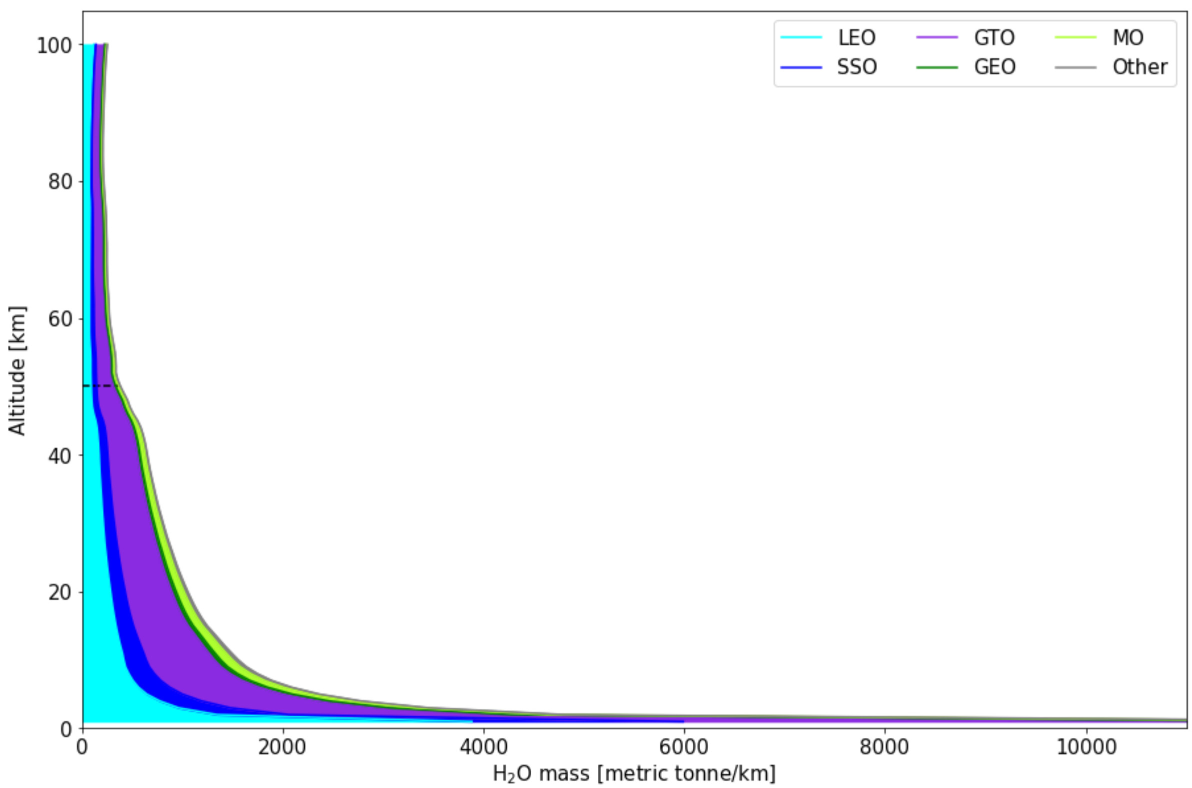


Figure 12 Global emissions of CO_2 and water vapor for the past decade, classified by targeted orbit. For readability purposes, we cut the emissions of the first km. The total CO_2 and H_2O mass emitted in the first km are respectively 20841 tonnes and 14533 tonnes.

The Soyuz is the greatest CO_2 contributor and the third greatest H_2O contributor, as it produces respectively 26.3% and 10.8% of the CO_2 and H_2O emitted by the launch vehicles that we simulate. It is also the rocket that has been launched the most frequently between 2009 and 2018: 22.7% of the launches that we simulated were from a Soyuz variant.

The Proton-M, the second launcher most simulated, represents 13% of CO_2 and H_2O emissions. It burns earth-storable propellants, alongside with the Rokot, the Chinese Long March (CZ) vehicles and partly the PSLV. All those rockets contribute equally to total CO_2 and H_2O emissions.

The other most common launchers are the Falcon 9, the Atlas V and the Ariane V. Kerosene-fueled rockets such as the Falcon 9 or the Atlas V contribute more to CO_2 emissions than water, although their shares in water emissions are respectively 8.4 and 7.5%. The Ariane V was launched almost as often, but because its main engines are cryogenic, its CO_2 emissions are smaller than the kerosene-fueled vehicles while its water emissions are greater. It is in fact, the first water emitter.

Most missions go to GTO (34%), LEO (27%) and SSO (22%). As a result, GTO missions represent the greatest emissions of CO_2 and H_2O, followed by LEO. GTO missions contribute more to H_2O production, with 42% of total emissions, because all the rockets with cryogenic main engines launch to GTO, especially the Ariane V. Finally, SSO contribution is around 14% for both products. This could suggest that emissions to SSO are smaller, however this will depend on the payload mass. Some rockets always launch to the same orbits: the Proton-M and the Ariane 5 only launched to GTO, while the Atlas V and the Falcon 9 launched to all orbits. Emissions have to be estimated with relatively to the mission of the launch.

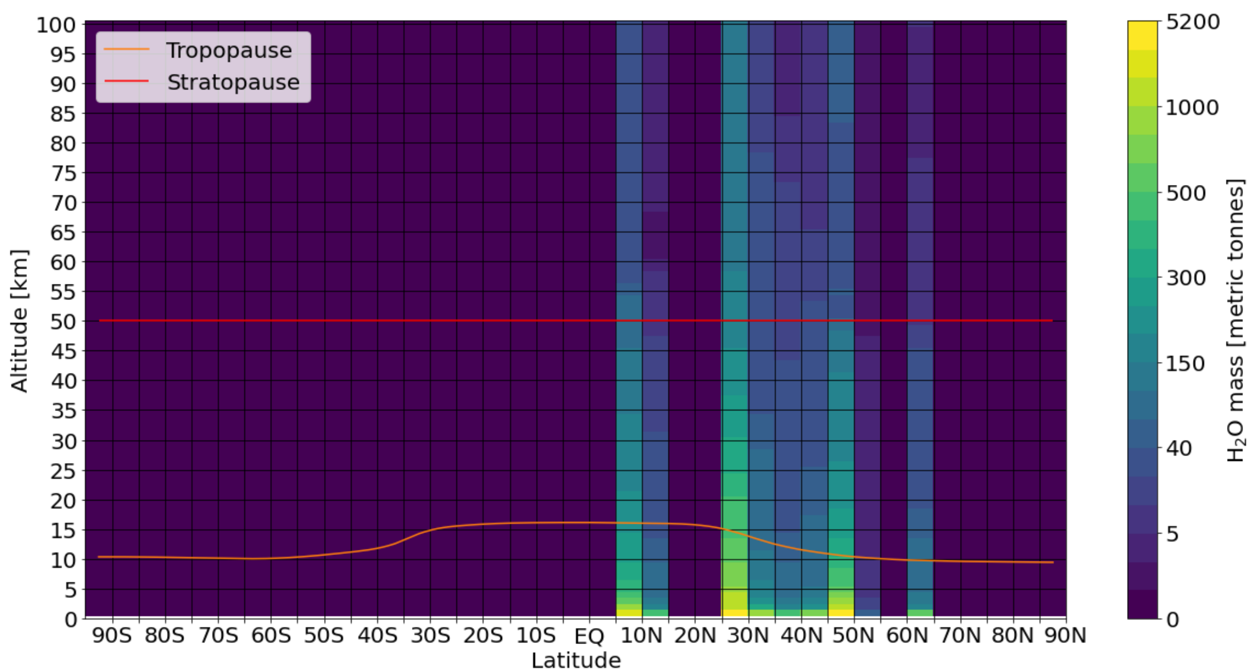


Figure 13 Total mass of water vapor (in metric tonnes) distributed within latitudes and altitudes.

4.2 Emission inventory for mission types

A launch vehicle is designed to accomplish a mission defined by the targeted orbit and the payload mass. The higher the targeted orbit is and the heavier the payload is, the more energy is required for the mission. We calculate the average exhaust mass per kilogram of payload that each vehicle has emitted between 2009 and 2018. Values for CO_2, water vapor and chlorine are displayed in Figures 14 to 16. Alumina results are similar to chlorine, as only solid propulsion engines can emit them. We try to group launch vehicles with similar propulsion systems. For example, Ariane-5, the Space Shuttle, H-IIA all use solid rocket boosters for lift-off and cryogenic engines as main engines.

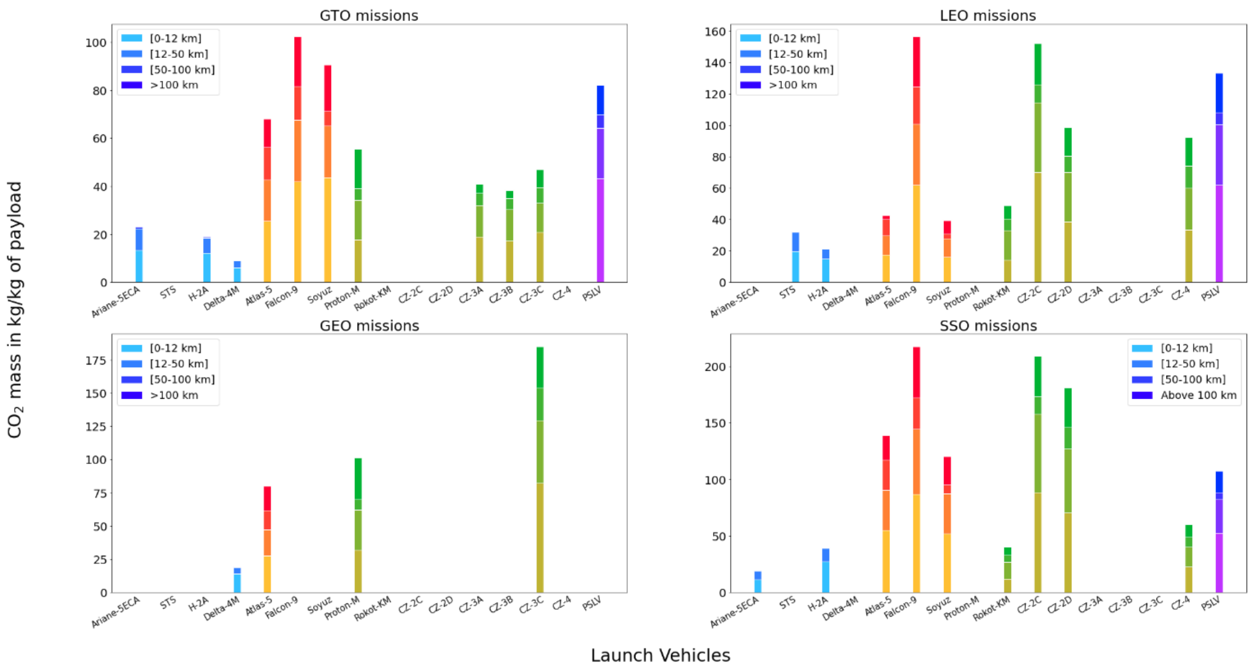


Figure 14 Average CO_2 rocket emissions for missions to LEO, SSO, GTO and GEO. STS is short for 'Space Transportation System', the other name for the Space Shuttle. Rockets are differentiated by the propellant used by their main engine: cryogenic in blue, RP-1 in red, earth-storable in green. The PSLV alternates propellants.

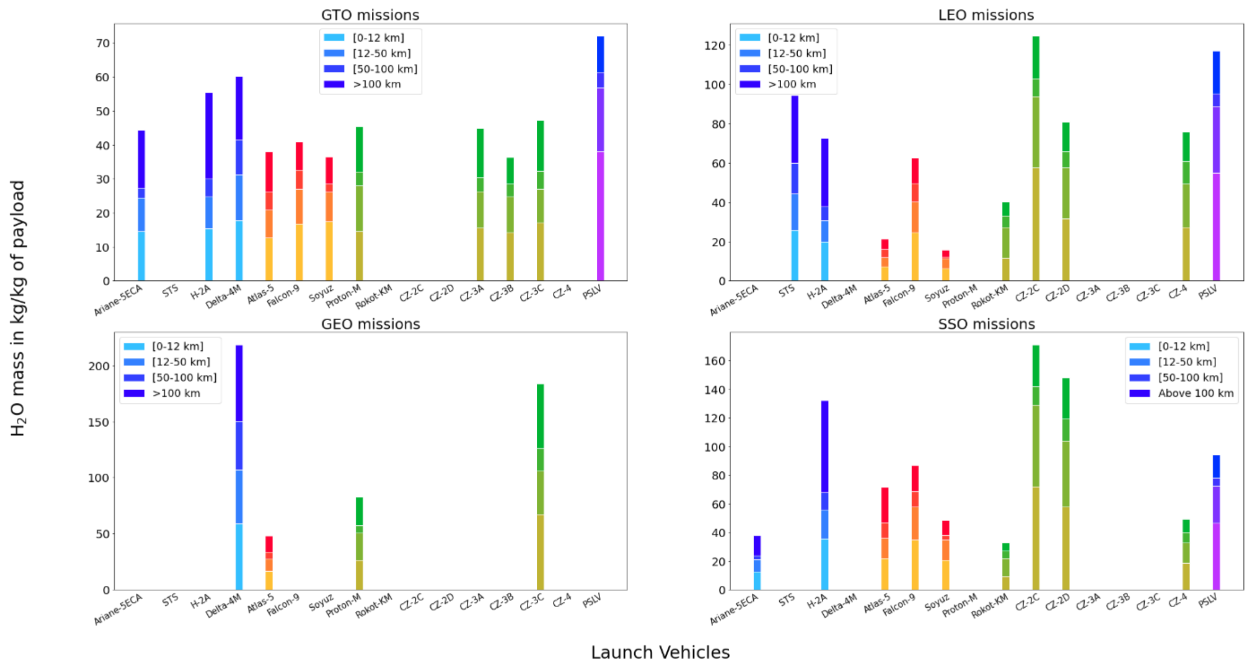


Figure 15 Average water vapor rocket emissions, similarly to Figure 14.

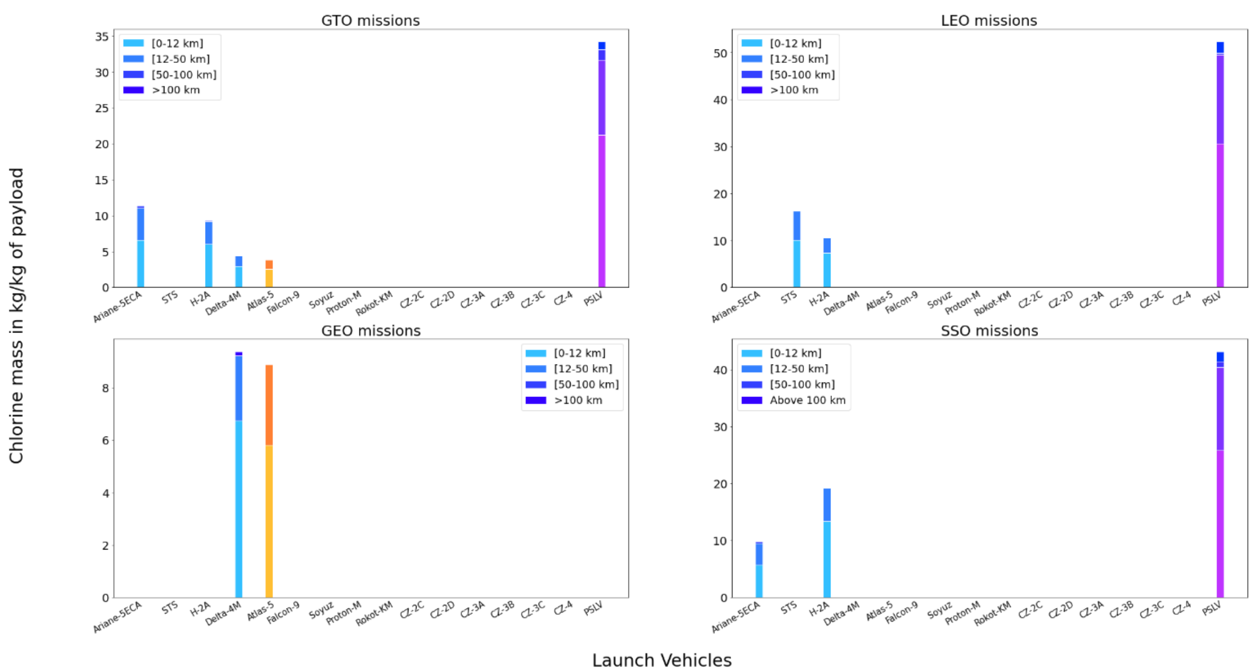


Figure 16 Average chlorine rocket performances. Only the rockets using solid propulsion systems are of interest.

4.3 Comparing launches with same purpose but different rockets

The inventory has previously presented emissions per kg of payload for different orbits to compare rocket emissions for similar missions. However, those values are averaged, and do not allow to compare launch vehicles used for similar purposes, such as launching the same family of payloads or launching crews to the International Space Station (ISS). Crewed missions have been occurring regularly over the decade, and are expected to increase, as crewed space exploration is developing. The United States used the Space Shuttle until it retired in 2011. Ever since, the Soyuz-FG has been the only vehicle to send astronauts to the ISS, until June 2020, when SpaceX became the first commercial company to send astronauts to space, with the reliable Falcon 9. Crewed missions to the ISS are characterized by a heavy payload sent to LEO (around 400 km of altitude), so the goal is to achieve sending a massive payload to a low orbit. However, the payload varies between the different launch vehicles: Soyuz FG missions send spacecrafts of around 7 t, whereas Falcon 9's Crew Dragon weighs fully loaded almost 13 t and the Shuttle is more than 16 t heavy. We plot the carbon dioxide and water vapor emission profiles of the three different missions.

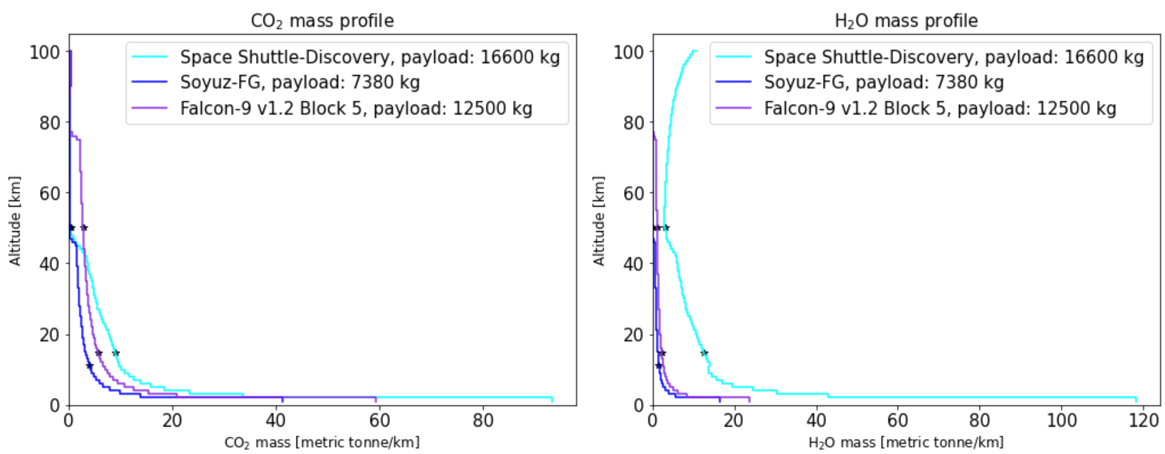


Figure 17 CO_2 and water vapor profile comparisons for three different crewed missions to the ISS

Chapter 5: Discussion

In this chapter, we analyze the results presented in the previous chapter. The aim is to have an overview of the global emissions and to compare the launch vehicles.

5.1 Global emissions between 2009 and 2018

Global CO_2 and water emissions are first considered. The trend in launch vehicles influence the emissions between 2009 and 2018. The Space Shuttle retirement in 2011 led to a decrease of almost 50% of water vapor emissions while the rise in Falcon 9 launches is making CO_2 emissions increase faster than the number of launches. The five rockets that were the most launched: Soyuz, Proton-M, Falcon-9, Atlas-V and Ariane 5 were verified to contribute the most to CO_2 water vapor and chlorine emissions. But their contributions vary with their main engines: kerosene engines contribute more to CO_2 emissions than water vapor, while cryogenic engines contribute more to water vapor emissions and earth-storable engine contributions are equivalent. The choice of propellant alone is however not enough to explain the differences in emissions.

The Ariane 5 emitted twice as much water than the Atlas V did, but also more than 80% its amount of CO_2. The only source of CO_2 for the Ariane 5 is its two SRM, the EAPs, that carry 240 t of propellant each and burn for the first 130 seconds. CO_2 emissions from the Ariane 5 could be reduced if the launcher relied more on its main engine to take-off. Falcon 9 CO_2 emissions are 1.5 times those of the Atlas V, which is interesting as the Atlas V burns SRBs for some missions. It seems that the Atlas V's ability to tailor the number of SRBs may help minimize its emissions, contrary to the other two rockets.

The Space Shuttle launched only 11 times in the ten years, and yet it emitted respectively 11% and 4.3% of the total water and CO_2. Those values can be explained by the two SRM that weighed twice as much as the EAPs, and the cryogenic main engine that carried four times more propellant than the Ariane 5's main engine.

Most of the other launch vehicles' simulations each represent between 3 and 5% of total launches. They burn earth-storable propellants or cryogenic propellants with SRM for lift-off. The H-IIA, that burn cryogenic propellant, produces respectively 1% and 4% of the total CO_2 and H_2O emissions. Earth-storable are a potentially interesting choice of propellant because they contribute as much to CO_2 and H_2O. For example, the CZ-2C was launched as often as the H-IIA. It contributes as much to CO_2 emissions but half as much to water emissions. However, the CZ-2C cannot be used to send a payload to GTO, while the H-IIA can.

The comparison of the different launchers illustrates already a trade-off between CO_2 and H_2O emissions. But the choice of launcher will depend on the mission. To go to a higher orbit a rocket needs to burn more propellant. However, the emissions do not follow the number of launches to each orbit. SSO missions contribute to 14% of water and CO_2 emissions, while it represents 22% of total launches. This could suggest that emissions to SSO are smaller, however this depends on the payload mass. We quantify this in the next subsection.

Before doing so, we discuss water vapor distribution across the different latitudes. All the emissions are displayed in the northern hemisphere. Some launches did occur in the southern hemisphere, as Rocket Lab launches its Electron in New Zealand. Since only four rockets were launched there between 2009 and 2018, it is a good approximation to maintain emissions to the southern hemisphere to 0. Most of the water is emitted near the Equator, near the northern Tropic but also at higher latitudes. Those emissions correspond to Centre Spatial Guyanais (Ariane-5), Cape Canaveral (most American rockets), and Baikonur Cosmodrome (most Russian rockets). The emissions are divided by ten between the troposphere and the stratosphere. At latitude 45 mass of water emitted increases at 85 km. Rockets that launch at the Baikonur Cosmodrome are the Soyuz and the Proton-M, the two main launchers between 2009 and 2018. This increase is probably caused more by the number of launches than the engine designs.

Water vapor emissions in the troposphere are of no interest, but the impact of water in the stratosphere and the upper atmosphere will depend on its latitude. A chemical transport model will be needed to quantify the impacts.

5.2 Emission inventory for mission types

We have shown that the greatest emitters were the rockets that launched the most, but that the choice of propellant and configuration influenced the values. The type of propellant can cause a tradeoff between water and CO_2 emission. 42% of water emissions were produced for missions to GTO, and that missions to SSO contributed less to CO_2 and water. However, those data do not enable to identify the emission performances of each launcher because they will depend on the missions they are designed for: the targeted orbit but also the payload mass. Chlorine emissions are also relevant to compare from one launcher to another, because of their impact on the stratosphere. We consider the three plots on figures 13, 14, 15.

Emissions for GEO are greater than for GTO. For example, in the troposphere a Delta-4M emits on average 18 kg of water per kg of payload for a mission to GTO and 56 kg of water per kg of payload for a mission to GEO. Because more energy is required to send a payload to GEO, the payload mass will be lighter. Since missions to GTO require more energy than to LEO and launcher payload capacities to GTO are generally smaller than to LEO, we expect emissions per payload mass for GTO to be greater than those for LEO or SSO. This however varies from one launcher to another. The PSLV changes configuration for different missions: missions to GTO use solid strap on boosters. This would entail larger emissions to GTO. And yet, on average, a PSLV rocket emits 133 kg of CO_2 for each kg of payload to LEO, 107 to SSO and only 82 to GTO. This means that lighter payloads are more frequently sent to lower orbits.

We next compare launchers for CO_2 emissions. The Falcon-9 is the greatest emitter with 156 kg /kg payload in LEO, 102 kg/kg payload in GTO and 216 kg/kg payload in SSO. Those results can be explained at the same time by the important amount of propellant it burns, but also by the wide range of payload it has sent this past decade, as those values are averaged While it can send very heavy payloads, it often launches payloads with mass largely smaller than its capability. Like the PSLV, it is specifically true for low orbits missions. The Soyuz is well suited for its LEO missions at it emits on average 39 kg/kg payload, which is even smaller than the Rokot-KM emissions. This is because the Soyuz-FG is used to launch crew missions to the ISS, therefore heavy payloads of more than 7 t, and the Rokot cannot launch payloads that exceed 2500 kg. The Atlas V's option to change the number of boosters enables sending payloads with mass varying between 900 kg and 6700 kg with emissions smaller than the Soyuz or the Falcon: on average 67 kg of CO_2 has been emitted per kg of payload.

The CZ-2C emissions are similar to those of the Falcon-9 for missions to SSO and LEO and are more than five times those of the H-2A. It seems that this launcher, which produce less CO_2 than the H-2A for a similar number of simulations emits more to send the same payload.

Finally, the rockets using cryogenic engines emit the least CO_2/kg of payload, with values between 9 kg/kg payload for the Delta 4 at GTO and 39 kg/kg payload for the H-2A at SSO. Even the Space Shuttle is in this range because it sends heavy payloads to LEO.

Water vapor emissions have different patterns. Rockets with cryogenic engines are water emitters, although launchers such as the PSLV and the CZ-2C are even larger emitters. This is probably because of their payload masses but also their propellant. The Ariane 5 is the rocket using cryogenic engines that minimizes water emissions the best, with emissions around 40 kg/kg payload.

Lastly, Figure 15 displays chlorine emissions, that originate from solid propellants. The PSLV is the only vehicle that we simulate with solid propulsion used for something else than strap-on boosters, hence the important gap between its emissions and that of other rockets. It also emits chlorine in the upper atmosphere. The emissions from cryogenic rockets using SRBs for LEO and SSO missions have similar trends than water vapor emissions.

However, GTO missions are the opposite: the Ariane 5 emits 11 kg/kg payload, while Delta-IV M only 4 kg/kg payload. This means that while the Ariane-5 relies more on its SRBs to lift-off and gain energy, the Delta-IV depends more on its main engine. SSO missions are achieved by the Atlas-V versions without any SRBs, hence the absence of chlorine emissions.

Alumina emissions can be derived from those of chlorines, since most SRBs use similar propellant constitution. Multiplying the values of chlorine emissions by 1.58 will give the average alumina mass per kg of payload. This number does not give many information as alumina impact will depend on atmospheric chemistry and combustion.

The choice of propellant will influence the emissions: solid propellants will produce chlorine and alumina, kerosene fuels tend to emit the most CO_2 and the least H_2O and cryogenic the opposite, while earth-storable propellants have similar amount of CO_2 and H_2O emissions, which can greatly vary from one launcher to another. But some send payloads with masses often smaller than their capability, when they could use the same propellant for heavier payload. The CZ series emit more CO_2 than kerosene-fueled rockets to send the same payload to SSO or LEO, because often send 300 kg payloads, while their payload capability to LEO varies between 1900 and 3600 kg. The Falcon-9 is cost effective because it reuses its first stage and can send a wide range of payloads, but it will also require more propellant to send a payload to a specific orbit. Getting the first stage back means using some propellant not to send the payload but to pilot the first stage: this propellant can be considered as economic savings but emission wastes.

Rockets such as the Ariane-5, the Soyuz or the Atlas-5 are more interesting. The Atlas-5 because it adapts its number of boosters to the payload, the Soyuz and the Ariane-5 because they are used to sending heavy payloads to specific orbits. The Ariane-5 seems however to reduce its water emissions by emitting more chlorine or alumina.

5.3 Comparing launches with same purpose but different rockets

The inventory has allowed us to compare the amount of emission to send the same propellant mass to orbit. Here we compare rockets that have a similar goal but different payload masses. The Space Shuttle is the largest emitter of water vapor. In the first kilometer, it produces more than fifty times the amount that the Falcon 9 and the Soyuz-FG do, and after reaching a minimum at the stratopause the quantity of water vapor increases again.

The Soyuz FG and the Falcon 9 both use kerosene as a propellant, the Soyuz FG emits less CO_2 and water vapor, as expected. However, the payload of the Falcon 9 is almost twice as heavy as that of the Soyuz FG, while Falcon-9 CO_2 emissions are 1.5 times those of the Soyuz-FG. The use of Falcon-9 would be in that case relevant for both cost and emission reasons.

The latitude of the launch is relevant. The Space Shuttle and the Falcon-9 are launched at Cape Canaveral (28.56°N) while the Soyuz FG is launch at Baikonur Cosmodrome (45.6°N). Plots show that it leads to a difference in water vapor emission within the troposphere. This is not enough however to make any conclusions on whether it is better to launch closer to the Equator or not.

5.4 The evolution of launch vehicles

Some launch vehicles are more suited than others to minimize the emissions of CO_2, water vapor or chlorine in the atmosphere. But the rockets are chosen for their performances and their costs. The cheapest launchers such as the Falcon-9 are not aligned with emission minimizations and they can on the contrary increase them.

Figure 10 shows the evolution of the number of launches per launch vehicle between 2009 and 2018. The first half of the decade displays a stable number of launches, except for the retirement of the Space Shuttle and the increase of Chinese launches.

The second half is more interesting. The historic American launchers Atlas V and Delta IV are being replaced by the Falcon 9, that launched more than once a month in 2018. Launches with rockets using SRBs and cryogenic engines are also decreasing, except for the Ariane-5, suggesting that less chlorine and alumina are being emitted while CO_2 emissions increase. The evolution of water vapor is harder to estimate because of the important of launchers with earth-storable propellants but the 70% contribution of kerosene fuel to H_2O emissions in 2018 and the increase in Falcon-9 launches suggest that water vapor emissions are going to increase again. Chinese launches keep rising, especially the CZ-2 and CZ-3B versions. Like the Falcon-9, the CZ-2, used for LEO and SSO missions, tends to not optimize its propellant contrary to the Atlas V.

The trend in the launches suggests that economic incentives will not be aligned with environmental harms. But this will also depend on the missions. The Falcon-9 for example, is going to increase crewed launches to the ISS, and will use its launch capability better.

Chapter 6: Conclusion

6.1 Summary

Launch vehicles are known to cause chemical impacts on the atmosphere with ozone depletion and positive radiative forcing as potential concerns. However, their impact is not well understood because emissions are not well quantified.

This work built the first comprehensive inventory of the stoichiometric emissions of launch vehicles between 2009 and 2018: CO_2, water vapor, chlorine and alumina. It used a program that provided launch trajectories and fuel burn profiles with publicly available data, before converting the propellant into emissions.

140.5 kt of CO_2 were emitted in the atmosphere, 78.9 kt of water vapor, 5 kt of chlorine and 7.8 kt of alumina were emitted above the tropopause. The increase in the number of launches has made CO_2 emissions grow by 73% between 2009 and 2018, while water and chlorine emissions have decreased by 25 and 58% since 2009 because of the retirement of the Space Shuttle. The rise in kerosene-fueled rockets launches, especially the Falcon-9, is making CO_2 emissions increase faster than the number of launches and suggest that water vapor emissions are going to increase.

Emissions vary from one launcher to another, and trade-offs are observed. CO_2 emissions are minimized with launchers that use cryogenic engines, although they often require solid boosters that emit CO_2, chlorine and alumina. Conversely, water vapor is minimized with launchers that use kerosene as a fuel, but that will increase CO_2 emissions. Earth-storable propellants did not show any trade-off for the emissions of interest, but they are known to be toxic and to emit NO_x.

Finally, emissions could be reduced by better tailoring the launch vehicle to the payload and the orbit: like the Atlas-V, by adding booster when needed, or like the Soyuz, by sending more often payloads close to their launch capability. However, rockets have not been designed to minimize their emissions but their cost and aim at maximizing their performances. Consequently, some payloads are sent by rockets with ill-adapted payload capability, such as the Falcon-9 or the CZ-2C.

The number of emissions is increasing with the number of launches. The trend in the launch vehicles suggests that economic incentives are going to be in opposition with emissions and, more broadly, environmental concerns. However, the appearance of new launchers such as the Ariane-6 or a better use of rockets such as the Falcon-9 launching heavier payloads may prove this wrong.

6.2 Future work

While this inventory was as comprehensive as possible, it could still improve, as some data is missing or have sources that could not be verified. We have only simulated 77 % of launches. Other launches should be simulated, either by using our method, or by assimilating each launch vehicle to one of our references.

The launch program could also be improved by a better understanding of how pitch angle is controlled during each launch.

Combustion modelling is also needed to calculate the other exhaust products, such as BC, or NO_x. A better understanding of rocket engines, rocket plume and aerosol chemistry would be important as well.

Once emissions are known, environmental impacts should be quantified. The use of a chemical transport model can help calculate the damage that water vapor could cause above the tropopause, to the ozone layer and to the temperature. Radiative forcing and ozone depletion induced by all the products could be calculated to quantify the damage of rockets. The environmental impact quantification could be taken further with a study on mesospheric clouds.

Appendix A: projecting below 100 km

In aeronautics and astronautics, the limit of the atmosphere is at the Karman line, which is approximately located at 100 km. It is considered the limit above which aeronautical flight can no longer be supported because the atmosphere is too thin. Beyond this line, spacecrafts can be set into orbit around the Earth. Propellants, in gaseous or solid form, are still burned, and can play a role in atmospheric emissions. We want to assess that propellant exhaust will fall back in the atmosphere and need to be included in emission assertions.

A.1 Propellant velocity

The first question that we want to answer is: will some of the exhaust products from rocket emissions escape Earth's gravity? And, for those that will not, will they stay in orbit around the Earth? We use the same method as Fourie et al., 2019 [16]. This paper evaluates the potential impacts of emissions from Hall Thrusters using mercury as their propellant, which is a product dangerous for human health. Their work treats only satellites already in orbit, and in LEO, but we can apply their method. We call $\boldsymbol{v_R}$ the velocity of the rocket, $\boldsymbol{v_{Ex}}$ the exhaust velocity of the products, and $\boldsymbol{v_{Ex/E}}$ the exhaust velocity with respect to the Earth. The relation between them is

$$\overrightarrow{v_{Ex/E}} = \overrightarrow{v_R} + \overrightarrow{v_{Ex}} \qquad (A.1)$$

Orbital velocity at an altitude h can be defined as

$$v_R(h) = \sqrt{2\mu\left(\frac{1}{R_E + h} - \frac{1}{(R_E + R_a) + (R_E + R_p)}\right)}; \qquad (A.2)$$

with R_E the Earth radius, R_a and R_p respectively the apogee and perigee of the orbit, and μ Earth's standard gravitational constant. For circular orbit, as the apogee and perigee are equal to the altitude, we have

$$v_R(h) = \sqrt{\frac{\mu}{R_E + h}} \qquad (A.3)$$

The exhaust velocity in vacuum can be approximated using the specific impulse. Indeed, by definition of thrust and specific impulse

$$v_{ex} = Isp - \frac{(P_e - P_a)}{\dot{m}} A_e \approx Isp \qquad (A.4)$$

In vacuum, the atmospheric pressure can be neglected. Equation (A.4) also shows that the exhaust velocity is smaller than specific impulse, so using specific impulses is satisfying for upper range. Chemical engines for upper stages have specific impulses between 200 s and 500 s, so those are the limits we will evaluate. Finally, the escape velocity at a specific altitude *h* is

$$v_e = \sqrt{\frac{2\mu}{R_E + h}} \qquad (A.5)$$

We now try to determine whether exhaust products can reach the escape velocity and thus leave the Earth's atmosphere. We calculate the magnitude of the exhaust velocity relatively to Earth and compare it to the escape velocity needed at the same altitude. In the simplest case, the rocket fires in the opposite direction of its velocity (angle of attack equal to zero). However, that is not necessarily the case, as thrust and rocket velocity direction are often not aligned. The exhaust velocity relatively of Earth is then

$$v_{Ex/E} = {v_R}^2 + {v_{Ex}}^2 - 2 v_R v_{Ex} \cos\alpha \qquad (A.7)$$

We do not know what values an angle of attack can reach, especially when the rocket is already above 100 km. An upper bound to be used can be an angle of 60°, although 30° is a more realistic value. Exhaust velocities relative to Earth for a launch vehicle in circular orbits are shown in Figure 19 and compared to the circular orbital velocity and the escape velocity.

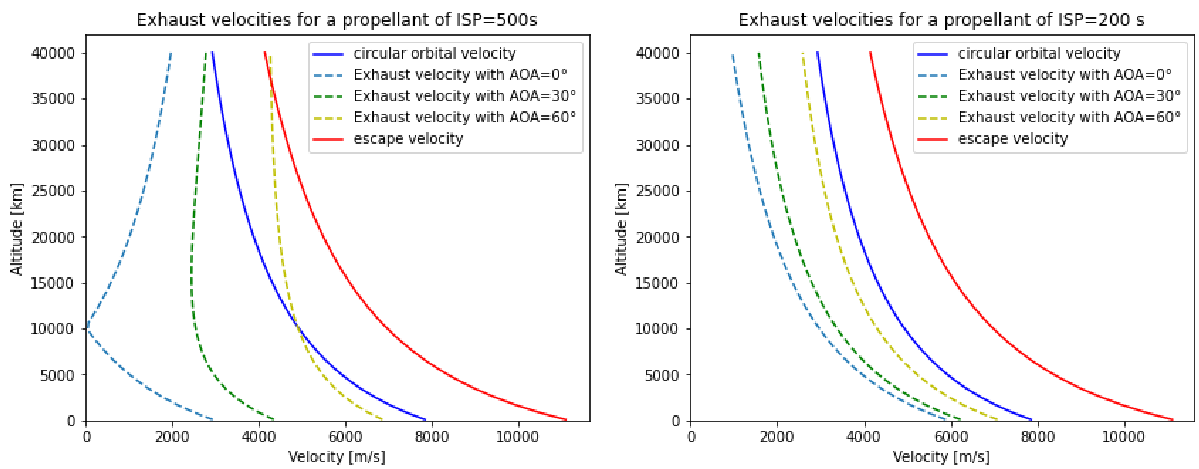


Figure 18 Exhaust velocity relatively to Earth for a vehicle in circular orbits between 0 and 40000 km for propellant with specific impulses of 200 and 500 s. AOA stands for angle of attack.

Orbital altitudes are between 100 km and 40 000 km. For a specific impulse of 200 s, neither circular nor escape velocity is ever reached. For a specific impulse of 500 s, escape velocity is reached for a 37 000 km high circular orbit, with an angle of 60 degrees, which is very unlikely to happen. In this case, uniquely, could the propellant reach a velocity higher than the circular orbital velocity at this altitude, which would propel it in another orbit. Now, a lot of orbits, whether temporary or not, are elliptical. Figure 20 illustrates the case of a GTO.

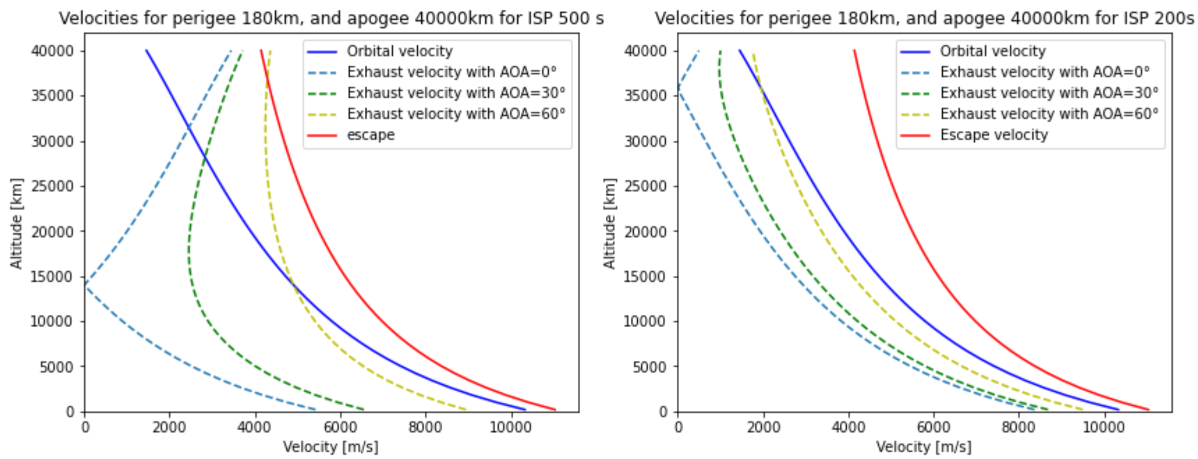


Figure 19 Exhaust velocity relatively to Earth as a function of altitude for a vehicle in GTO (perigee 180 km, apogee 40 000 km) for a propellant with specific impulses of 200 and 500 s.

Once again, for the low bound of specific impulse, escape velocity is never reached, and for the high bound, it happens in unlikely conditions. Indeed, an upper stage is rarely going to separate from the satellites at high altitudes but tends to leave the satellite in a lower altitude of the orbit.

The exhaust velocity is therefore unlikely to reach escape velocity when the rocket is in a specific orbit. However, a launch vehicle can travel along successive orbits. To change orbits, it does an impulsive maneuver that is considered instantaneous. For example, the Zenit 3F, when launching Elektro L-2 in Geosynchronous Orbit, has several intermediate orbits before reaching GEO [17]. We do not know the altitude at which each maneuver is realized, so we decide that it happens at the perigee of the future orbit. In this case, the orbital maneuver between orbit 167x554 km and 280x4306 km occurs at 280 km of altitude. We calculate the exhaust velocity for a specific impulse of 500 s and an angle of 60° at 280 km for both orbits. We observe that in each case the velocity is smaller than escape velocity. This happens for every maneuver (See table 2).

Orbital perigee (km)	Orbital apogee (km)	Altitude (km)	Exhaust velocity at this altitude (m/s)	Escape Velocity at this altitude (m/s)
167	554	280	6819	10948
280	4306	280	7467	10948
280	4306	338	7417	10900
338	35 911	338	8771	10900
338	35 911	35425	4318	4367
35 425	35 793	35425	4296	4367

Table 2 Velocities calculated during changes of orbits for a thrust angle of 60° and a specific impulse of 500 s

It seems that there should not be any exhaust loss during the launch to orbit. Now, once the payload is injected, some upper stages can lower their perigee to reach either a less congested area, or a re-entry altitude. In this case, we talk about retrograde burn, as thrust is fired in the direction of velocity to reduce the orbit. Baldwin and Lu, 2012 [18] indicate that the angle of thrust is not necessarily 180°. As de-orbit often seems to happen in SSO, we plot exhaust velocities in circular orbits for thrust angles changing to values between 30 and 180 °, and for two typical specific impulses (Figure 21). We observe that for high specific impulses, thrust angles greater than 120° for retrograde burn can lead the exhaust to reach escape velocity.

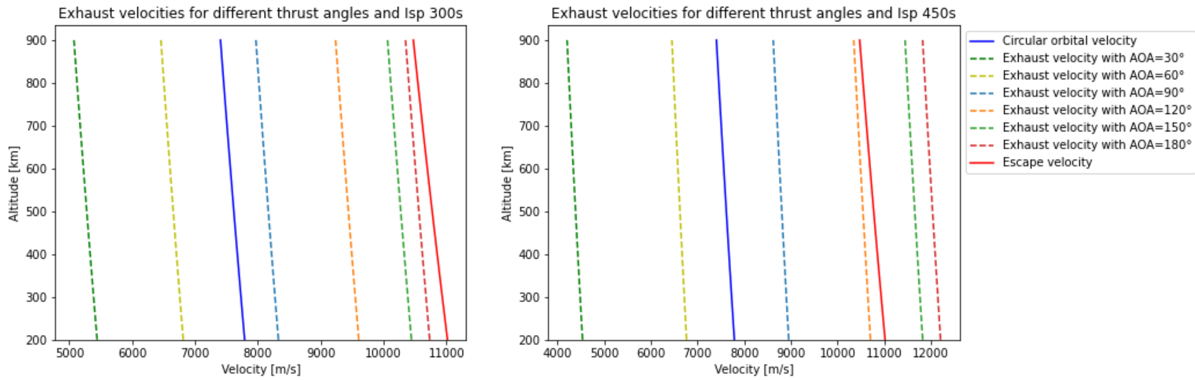


Figure 20 Exhaust velocity relatively to Earth in circular orbit with different thrust angles for retrograde burn

Now, we look at changes in orbits. We show two cases when from a circular orbit, the upper stage performs a burn that lowers the perigee and gets into an elliptical orbit with apogee being the altitude of the previous orbit. The first one is performed by Delta II's upper stage [19], and the second a Briz KM [20]. They have different specific impulses. Propellant velocity is plotted as a function of thrust angle. We observe that for Delta II's upper stage, escape velocity is never reached. However, for an angle greater than 160°, Briz KM will create some exhaust that will escape velocity.

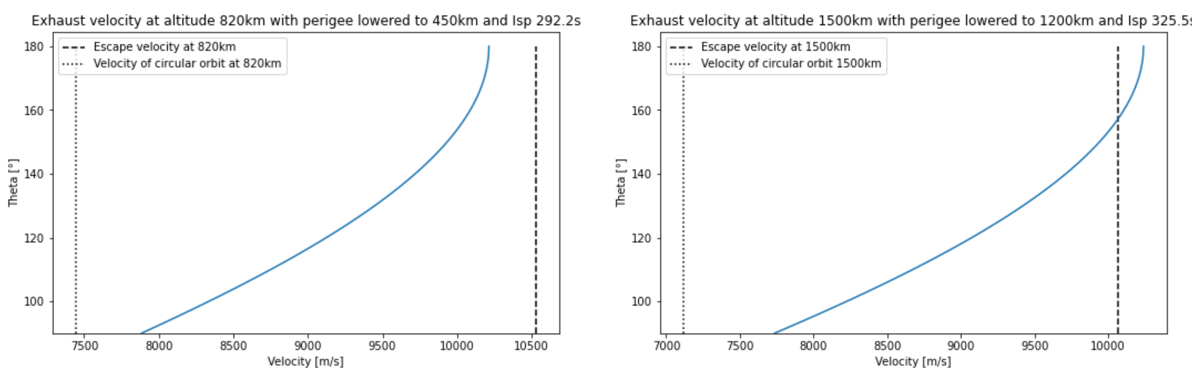


Figure 21 Exhaust velocities during perigee lowering as a function of AOA. Left: Delta II upper stage. Right: Briz KM.

Therefore, there are possible losses for deorbit. But in the general case, we can assume that the propellant will not reach escape velocity.

A.2 Lifetime in the upper atmosphere

We have established that propellant exhaust should stay within Earth's gravitational field. The remaining question is how long it would take to fall back below 100 km. Zinn et al. 1982 [21] studied the effects of rocket exhaust in the ionosphere. Considering cryogenic engines, they calculated that it took two hours for a water molecule to fall from 400 to 200 km of altitude, and six hours for a dihydrogen molecule. 1-

µm particles in either LEO or GTO with a perigee at 300 km were estimated by Ratcliff et al. 1993 [22] to stay in orbit for no more than a month. Although for gases nano sizes are more relevant, it can be an interesting order of magnitude for solid exhausts whose diameter can reach these sizes.

The region above 500 km is the exosphere, where mean free path is so big that collisions occur very rarely. The gas in the exosphere does not behave strictly as a continuum fluid. Neutral atoms and molecules follow ballistic trajectories in the Earth's gravitational field, with infrequent collisions, but can encounter some collisions that reduce their lifetimes. Richter et al. 1979 [23] estimated that a satellite particle in an orbit perigee of 3000 km and apogee of 250 900 km had a lifetime of 116 days. As a comparison, particles in nearly circular orbits at 520 km have a lifetime of a few seconds.

Below the exobase and above 100 km, eddy diffusion is dominated by molecular diffusion: vertical mixing is replaced by diffusive separation of species according to their molecular mass. Diffusion time is defined as the time for gravitational separation to cause a significant change in the density of a given constituent at a given level of the atmosphere. Kockarts [24] highlights that diffusion time below the exobase can be between a few hours and a day. It thus needs to be taken into consideration to model propellant expansion.

Bernhardt et al. included diffusion in their 1982 publication [25], with a three-dimensional diffusive model. In this example, they simulated a Saturn V-type launch vehicle which releases 1.67 x 1031 water molecules from 150 to 450 km altitude. The exhaust product falls back at the bottom of the thermosphere in a matter of hours. As noted in Boden et al [26], who look at Falcon 9 launches, the initial downward velocity of rocket exhaust molecules would result in their reaching lower altitudes before entering a diffusive expansion regime, when they start colliding with other particles. This collision affects the ionosphere, where electrons are recombined to form molecules. The ionosphere thus undergoes ion depletion for a few hours although it is of no interest for the scope of our problem. We can assert that the rocket exhaust products will not change form nor recombine by the time they fall back below 100 km.

Those different studies do not give a clear answer on how long exactly exhaust products would stay above 100 km. They are still good enough to assess that exhaust products may not stay in the upper atmosphere for more than a year, and that we can therefore estimate that all the exhaust mass will fall back down.

Appendix B: Solid propulsion

To calculate thrust, the exit pressure, the mass flow rate and the exhaust velocity are to be evaluated separately. Our expressions are derived from [12] and [27]. The exhaust velocity is defined as

$$Ve = Me\sqrt{\frac{\gamma RTc}{M}\frac{1}{\left(1+\frac{\gamma-1}{2}Me^{2}\right)}} \qquad (B.1)$$

where M_e is the exit Mach number, T_c the chamber temperature, γ the heat capacity ratio and M the propellant molar mass. While the three last values are parameters, we calculate the exit Mach number by using the expansion ratio, which is a function of A_t the throat area and A_e the exit area.

$$\frac{A_e}{A_t} = \left(\frac{\gamma+1}{2}\right)^{-\frac{\gamma+1}{2(\gamma-1)}}\frac{\left(1+\frac{\gamma-1}{2}{M_e}^{2}\right)^{\frac{\gamma+1}{2(\gamma-1)}}}{M_e} \qquad (B.2)$$

Mass flow rate is calculated at the nozzle. We approximate that mass flow rate at the chamber is negligible, therefore mass flow rate at the nozzle is mass flow rate at the gas generator

$$\dot{m} = A_b r\rho - \frac{d(Vc\rho c)}{dt} \approx A_b r\rho\,; \qquad (B.3)$$

with A_b the area of the propellant that is being burned, *r* the burning rate, and ρ the propellant density. The burning rate changes with chamber pressure and depends on two coefficients, the temperature coefficient *a* and the burning rate exponent *n* that are measured experimentally for each propellant. The most common empirical equation is

$$r = a{p_c}^{n} \qquad (B.4)$$

However, chamber pressure is not constant, as it varies with the burning area of the propellant

$${p_c}^{1-n} = \frac{A_b}{A_t}a\rho\sqrt{\frac{RT_c}{M\gamma}\left(\frac{\gamma+1}{2}\right)^{\frac{\gamma+1}{\gamma-1}}} \qquad (B.5)$$

Since the change in burning area also depends on the burning rate, A_b, *r* and p_c depend on one another. They also allow to calculate the exit pressure p_e that depends on chamber pressure

$$p_e = \frac{p_c}{\left(1+\frac{\gamma-1}{2}Me^{2}\right)^{\frac{\gamma}{\gamma-1}}} \qquad (B.6)$$

Those expressions rely on many parameters. Most of those inputs can be estimated as constant, but this means finding the right value of six constants that will change for each different engine. Thrust is sensitive to some of the constants, such as a and n.

The burning area of the propellant A_b is essential to calculate thrust. Having the initial burning area and being able to calculate its evolution is thus necessary. Several difficulties are raised:

- Uncertainty on the parameters: thrust and mass flow rate are highly sensitive to the burning rate coefficients especially. Those parameters are difficult to find in literature, and sources disagree on the values. For example, literature suggested at least two different sets of parameters for the Space Shuttle's SRB, which is better documented than most solid engines.

- Shape of the propellant: the propellant geometry can be complex and sometimes evolve along its length: the cross-section geometry is not necessarily constant.

- Inhibition: an inhibiter can be applied on the extremities of a propellant segment. It is used to influence thrust profile because it prevents the propellant from being consumed too fast. However, this inhibition can sometime be partial: only part of the cross section is inhibited and thrust is sensitive to those variations.

Those last two points would require a time consuming 3-dimension representation.

Appendix C: Simulations of reference

The Python program was first developed by comparing some simulations with references. The references come from YouTube launch webcast or manual profiles.

C.1 Falcon 9 v1.2

The Falcon 9 v1.2 has several particularities. The first one is first stage recovery. Between 2013 and 2016 it conducted sixteen test flights. The first success was in December 2015 and since 2017, the first stage landing was no longer an experiment but an inherent part of the launch. First stage is always recovered except when the biggest payloads are launch, as they require total use of propellant load. While we do not simulate the first stage landing, this tells us that the propellant of the first stage is not completely consumed for the launch itself. The first stage landing is indeed enabled through engine firing, contrary to previous technology. We find that approximately 6% of the first stage propellant is used for recovery. For our work, we classify this amount as structure mass and not propellant mass.

The Falcon 9 rocket is made of two stages: the first one is fired by the Merlin 9, and the second one by the Merlin 1. Both engines can throttle down. The first stage can often throttle down within the first 100 seconds of the launch to reduce stress on the vehicle when it is passing maximum dynamic pressure. It is done for GTO missions and Dragon missions, the latter requiring longer throttling. Several telemetry data seem to indicate that throttling around maximum dynamic pressure can happen for LEO missions. However, we do not know why some of them require throttling and some do not, and we do not know when and how long exactly this throttling occurs. This implies error in the simulation, and since the throttling a maximum dynamic pressure does not affect the rest of the acceleration profile, we decide not to simulate it.

The second cause of throttling can be found in the Falcon User's Guide. As described in 2.5.3, we decide to throttle only according to the axial acceleration limits, acknowledging that the rocket would probably throttle more at some moments, especially around MECO.

We simulate three launches: CRS-8 (mission to the ISS), Formosat (SSO mission), and SES-9 (GTO mission) for which we optimize the angle parameters. We obtained the altitude and speed profile from SpaceX's launch webcasts, that share telemetry data. We also derived those profiles using the finite difference method to compare height, speed, acceleration and vertical velocity. We observe that for the SSO mission, the second stage's first burn injects directly the payload into SSO, therefore at an altitude between 700 and 800 km, whereas most other launches insert their payload at an altitude between 170 and 250 km.

C.2 Ariane 5-ECA

Ariane 5-ECA is the most popular version of the Ariane that is used by the Europeans, before it gets replaced by the Ariane 6. The Ariane 5 consists in two stages in series. The first stage comprises the EPC, the core engine, and the EAPs, two solid rocket boosters used for lift-off. The EPC burns out at around 200 km of altitude, and the second stage starts operating, but stays at the same altitude for 500 seconds. As we do not simulate above an altitude of 100 km, we stop the simulation at EPC burn out. Both the EPC and upper stage use cryogenic propellant.

We have different sources of reference: Arianespace webcasts and the Ariane 5 User's Manual. We use the manual's load factor, speed and altitude profiles as reference, as the Arianespace launch webcasts only provide speed and altitude.

C.3 Soyuz-FG

The Soyuz is the rocket that flew the most between 2009 and 2018. It was developed by the Russians (Starsem) but is also launched from Guiana in partnership with Arianespace. It has multiple versions: Soyuz-U, Soyuz-FG, Soyuz 2.1a, Soyuz 2.1b, Soyuz 2.1v. All of them are very similar, except the Soyuz 2.1v that we do not simulate. For example, the main difference between Soyuz-FG and Soyuz 2.1 a is a conversion of flight conversion system from analogue to digital. The Soyuz-FG is used to launch missions to the ISS, especially crewed ones.

The Soyuz comprises three blocks: four liquid boosters, one core engine and a second stage. Depending on the mission, the Fregat upper stage can be added, with different variants. We only simulate until second stage cutoff. The liquid boosters and the core engine have throttling capabilities, however they only throttle before lift-off. The thrust values given by the Starsem and Arianespace manuals include verniers that overestimate speed profiles. As those verniers are used for attitude control, we decide to not simulate them.

C.4 CZ-2C, CZ-2D and CZ-4

The Changzheng (Long March) family was developed to launch all types of Chinese satellites since the mid-60s. The CZ-2C version is a two-stage rocket that was launched for the first time in 1975. It is mainly used for LEO missions. An upper stage can be added to send higher payload to orbit: either the SMA, which is a solid motor or the YZ1, same propellant as other stages. The YZ1 exists in different versions and is used as an upper stage for several Long March configurations.

We use the acceleration, speed and altitude profile from the Long March 2C User Manual as reference. It was mainly developed by the China Academy of Launch Vehicle Technology (CALT) while the following two vehicles were developed by the Shanghai Academy of Spaceflight Technology (SAST).

CZ-2D is a similar launch vehicle that is used for LEO and SSO missions and it is not clear when it is preferred to the CZ-2C. The only references available were plots from launch webcast that we use for optimization.

CZ-4B and CZ-4C are used for SSO missions. Similarly to CZ-2D, we use plots from launch webcasts as reference. The CZ-4B is a three-stage launcher based on the CZ-3 vehicle that is described below. We simulate all three stages.

All those launch vehicles use earth-storable liquid propellants (UDMH/N2O4) and they do not throttle. Verniers also burn on second stages for attitude control.

C.5 CZ-3A and CZ-3BE

The Long March 3 series are the launch vehicles that were developed for GTO missions. They have three stages, the first two being based on the CZ-2C series, with earth-storable propellants, and the upper stage a cryogenic stage. No engine throttles either. The CZ-3B and CZ-3BE versions also use liquid boosters for higher performances at lift-off. The upper stage is simulated only partially. It burns it two steps, with a coasting phase in between, so we simulate only the first burn.

We use as references plots from the *LM-3A Series Launch Vehicle User's Manual*, speed and altitude profiles for the optimization, and acceleration to validate our results. The manual has plots for the three launch vehicles.

C.6 Delta IV

The Delta launch vehicle was first developed in the 1950s and Delta IV is its latest version. The version that we are interested in is the Delta IV Medium Plus (Delta IV M+). It uses two liquid stages with liquid hydrogen and oxygen as propellant, and solid rocket boosters. The main engine, the Common Booster Core, is also used as boosters for the Delta IV Heavy. The solid rocket boosters are Graphite Epoxy Motors (the GEM 60) developed by ATK. The Delta IV M+ can send medium-class payloads to orbits and can accommodate different payloads by varying the number of SRBs and the type of payload fairing. Some versions also use the solid motor STAR 48B as an upper stage. The rocket uses two different launch sites: Cape Canaveral and Vandenberg AFB.

The Delta IV User Guide does not provide altitude or speed profiles. The launches are webcasted by the United Launch Alliance (ULA), but the telemetry data is very scarce. We obtained as a reference incomplete altitude, speed and acceleration values for the first stage phase, after the SRBs are being jettisoned. The CBC can throttle down to 55% of full thrust. But by comparing the references with our simulations, we inferred that in the case of Delta IV M+ launches it throttles very little, except before staging. The User Guide provides maximum axial acceleration values during first stage burn as a function of payload weight. From those, we set that axial acceleration cannot exceed 6 g.

C.7 Atlas V

The Atlas is another launch vehicle that has been used for decades by the US. The Atlas V is the latest of the series, developed to meet demands for government and commercial launches. Like the Delta-IV, it has several configurations to accommodate different payload and mission requirements, as it can launch to any orbit. It has two different payload fairing configuration, a 4-m diameter (Atlas V 400 series) and a 5-m diameter one (Atlas V 500 series). The main engine is for all configuration the RD-180, that uses RP-1 as fuel. The upper stage is the Centaur, propelled by liquid hydrogen and oxygen. SRBs can also be added for mission requirements, up to five for the 500 series. Until 2020, those SRBs were the AJ-60A, developed by the historic Aerojet. They were recently replaced by the GEM 63, provided by Northrop Grumman. For our scope, we are only interested in the AJ-60A. However, due to lack of data on their thrust profile, we decided to use data from the GEM-63, as we were told they were pretty similar.

We built our optimization on two configurations. The first one is the Atlas V 401, that has no booster and is the configuration that was the most launched. The Atlas V Launch Services User's Guide provides only acceleration profile and values of altitude at key events. We did not find any altitude profile from ULA webcasts, so we contented ourselves with the acceleration profile and the altitude value at main engine cutoff, which occurs over 100 km. The RD-180 can throttle down and the User's Guide has a throttling profile. This profile does not allow to obtain constant acceleration before MECO and payload fairing jettison, so we add a condition to maintain axial acceleration to values defined in the User's Guide. For example, we know that g-levels cannot reach 5 g before MECO.

The second configuration is one with SRBs. Once again, a launch profile was hard to find, but ULA shared telemetry for the Perseverance mission, in July 2020. We simulated this launch, with an Atlas V 541, that used four boosters. The altitude telemetry available is wrong, as it claims that MECO occurred at 700 km of altitude. We optimized the speed profile and with altitude at MECO provided by ULA on their website. We also use throttling profile and description from the User's Guide, that are similar but different from the Atlas V 401.

C.8 Proton M

Proton is Russia's primary heavy lift launch vehicle for uncrewed missions. The most recent is the Proton Briz M, that uses the Briz M as an upper stage, in addition to three stages. All engines run on earth-storable propellants. None of them throttle. We use as reference altitude and speed profiles from the Proton Launch System Mission Planner's Guide. The launch vehicle is used mostly for GTO missions.

C.9 PSLV family

The Polar Satellite Launch Vehicle (PSLV) is a rocket developed by India, originally to send some of its satellite to SSO, although it can also do missions to GTO. It exists in three configurations: PSLV CA, PSLV G and PSLV XL. The two latter use six strap-on SRB, with the one for PSLV XL being more powerful. Only four are ignited on the ground, while the remaining ones are lit in the air 25 seconds after liftoff. The particularity of the PSLV is that it alternates solid motors and liquid engines. The first and third stage are solid while the second and fourth stage are liquid (earth-storable propellants), although we do not need to simulate the last two stages. It is the first Indian launcher to use liquid engines.

We simulate the PSLV-XL and PSLV-CA, to have configurations with and without SRBs. We use as references plots that are available on webcasts shared by the ISRO. For the PSLV CA the altitude profile reference is different from other launchers as it is a function of downrange distance and not time. We adapted the optimization to take it into account.

C.10 H-II A family

The H-II A is the primary launcher used by the Japan Aerospace Exploration Agency (JAXA). It can accommodate for a lot of targeted orbits, although in practice missions are mostly to SSO and GTO. H-IIA has different variants. Two of them are relevant between 2009 and 2018: H-2A202 and H-2A204. H-A202 has two solid motors used as strap-on boosters while H-2A204 has four of them and higher payload capability to GTO. Its two main engines, the LE-7A and LE-5B, use liquid hydrogen and oxygen. The LE-5B has throttling capabilities that we do not simulate as the engine burns above 100 km.

The H-2A202 and H-2A204 boosters (SRB-A), while similar, are not the same model. The SRB-A used for the H-2A02 burns for a shorter time and has a higher maximum thrust. We were only able to find thrust profile for the booster used for the H-2A04 as they also burn for more recent Japanese launch vehicles such as the Epsilon. The H-IIA User's Manual provide typical altitude, speed and load factor for GTO missions for H-2A202. Since the load factor is not a function of flight path angle and since we can estimate liquid thrust from the main engine, we use the references to calculate thrust in vacuum for the SRB-A. We can them optimize the angles by following the method described in 2.3.

C.11 Rokot

Rokot is a three-stage launcher developed by Eurockot, a German-Russian company, for missions to LEO and particularly SSO. All the stages use earth-storable propellants. The third stage is the Breeze KM upper stage.

No engine throttles. We find an altitude reference from a webcast of the Sentinel 3B launch shared by the ESA. Like the PSLV CA, we only have altitude as a function of downrange distance, which requires to adapt the optimization.

C.12 Space Shuttle

This is certainly the launch vehicle with the most documentation available. It has been a real icon for thirty years, used for crewed missions to the ISS. Its retirement in 2011, for safety reasons, put American crewed launches on hold. The Space Shuttle has a very unusual architecture for a launcher. The orbiter, a shuttle, is at the same time a stage and the payload. Its main propulsion system encompasses three cryogenic engines called the SSMEs that receive the propellant from the external tank, on which the orbiter is placed. The orbiter also possesses an orbital maneuvering system (OMS) whose function is reaction control and orbit insertion, transfer, rendezvous and deorbit.

Two massive SRBs are used for lift-off: the Reusable Solid Rocket Motors (RSRM) provide 80% of thrust at lift-off. They are made of four segments with different geometry, while more recent solid motors have simpler designs. They burn jointly with the SSME for 123 s before being jettisoned. We continue the simulation until the SSME burn out and the external tank is jettisoned. The SSME throttle between 104% and 66%, we use a throttling profile available online. Optimization proceeds as described in Chapter 2.

Appendix D: Launch vehicle data

The inputs to create a rocket launch are presented in the tables below. The stages and propulsion systems are classified as described in 2.2. The number of engines is given into brackets. For example, Angara-A5 is made of three stages, and the first stage encompasses two different propulsion systems: four boosters P.0 and one main engine P.1.

“N/A” signifies that the propulsion system does not need that value, while “x” means that the value is missing. Upper stages such as Briz-M or rocket configurations are indicated into brackets.

Angara-A5(Briz-M)

Input [Unit]	Value				Reference
Fairing diameter [m]	4.35				[28]
Fairing length [m]	15.26				[28]
Fairing nose diameter [m]	0.128				[28]
Fairing nose length [m]	6.12				[28]
Fairing mass [kg]	2150.0				[29]
Fairing jettison time [s]	345.0				[30]
Latitude of the launch [°]	62.92				[28]
Payload mass [kg]	2042				[31] [32] [33]
Stages	Stage 0		Stage 1	Stage 2	
Length [m]	25.6		6.9	2.6	[29]
Diameter [m]	2.9		2.9	2.9	[29]
Propulsion System	P.0 (x4)	P.1 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	RP-1	RP-1	RP-1	UDMH	[29]
Oxidizer	O_2	O_2	O_2	N_2O_4	[29]
Propellant mass [tonnes]	132.7	132.7	35.8	19.8	[34] & [28]
Structure mass [tonnes]	10.0	10.0	4.8	2.5	[34] & [28]
Sea level thrust [kN]	1922.0	1922.0	294.0	19.62	[34] & [28]
Vacuum thrust [kN]	2090.0	2090.0	294.0	19.62	[34] & [28]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	N/A	
Beginning time T+[s]	0.0	0.0	330.0	x	[30]
Burn time [s]	325.0	212.0	395.0	x	[30]
Jettison time T+[s]	330.0	214.0	735.0	x	[30]

Antares-110

Input [Unit]	Value		Reference
Fairing diameter [m]	3.936		[35]
Fairing length [m]	9.851		[35]
Fairing nose diameter [m]	1.38		[35]
Fairing nose length [m]	3.2		[35]
Fairing mass [kg]	972.0		[35]
Fairing jettison time [s]	320.0		[36]
Latitude of the launch [°]	37.94		[35]
Payload mass [kg]	Between 3807 and 4127		[31] [32] [33]
Stages	Stage 0	Stage 1	
Length [m]	27.0	3.5	[35]
Diameter [m]	3.9	3.9	[35]
Propulsion System	P.0 (x1)	P.0 (x1)	
Fuel	RP-1	NH_4ClO_4	[35]
Oxidizer	O_2	Al	[35]
Propellant mass [tonnes]	242.4	12.834	[37]
Structure mass [tonnes]	18.8	1.224	[37]
Sea level thrust [kN]	3255.8	177.929	[38]
Vacuum thrust [kN]	3632.38	Thrust profile (Solid)	[37] & [39]
SRM nozzle exit area [m^2]	N/A	1.97	[39]
SRM specific impulse [s]	N/A	300.6	[39]
Beginning time T+[s]	0.0	335.0	[36]
Burn time [s]	233.0	128.0	[36]
Jettison time T+[s]	239.0	583.0	[36]

Antares-120

Input [Unit]	Value		Reference
Fairing diameter [m]	3.936		[35]
Fairing length [m]	9.851		[35]
Fairing nose diameter [m]	1.38		[35]
Fairing nose length [m]	3.2		[35]
Fairing mass [kg]	972.0		[35]
Fairing jettison time [s]	327.0		[36]
Latitude of the launch [°]	37.94		[35]
Payload mass [kg]	Between 4923 and 5644		[31] [32] [33]
Stages	Stage 0	Stage 1	
Length [m]	27.0	3.5	[35]
Diameter [m]	3.9	3.9	[35]
Propulsion System	P.0 (x1)	P.0 (x1)	
Fuel	RP-1	NH_4ClO_4	[37]
Oxidizer	O_2	Al	[37]
Propellant mass [tonnes]	242.4	12.834	[37]
Structure mass [tonnes]	18.8	1.224	[37]
Sea level thrust [kN]	3255.8	177.929	[38]
Vacuum thrust [kN]	3632.38	Thrust profile (Solid)	[37] & [39]
SRM nozzle exit area [m^2]	N/A	1.97	[39]
SRM specific impulse [s]	N/A	300.6	[39]
Beginning time T+[s]	0.0	335.0	[36]
Burn time [s]	233.0	128.0	[36]
Jettison time T+[s]	239.0	583.0	[36]

Antares-230

Input [Unit]	Value		Reference
Fairing diameter [m]	3.936		[35]
Fairing length [m]	9.851		[35]
Fairing nose diameter [m]	1.38		[35]
Fairing nose length [m]	3.2		[35]
Fairing mass [kg]	972.0		[35]
Fairing jettison time [s]	249.7		[40]
Latitude of the launch [°]	37.94		[35]
Payload mass [kg]	Between 6163 and 6173		[31] [32] [33]
Stages	Stage 0	Stage 1	
Length [m]	27.6	5.94	[35]
Diameter [m]	3.9	3.9	[35]
Propulsion System	P.0 (x1)	P.0 (x1)	
Fuel	RP-1	NH_4ClO_4	[37]
Oxidizer	O_2	Al	[37]
Propellant mass [tonnes]	239.3	24.9	[37]
Structure mass [tonnes]	4.44	1.587	[37]
Sea level thrust [kN]	3844.0	444.822	[41]
Vacuum thrust [kN]	4170.0	Thrust profile (Solid)	[41] & [39]
SRM nozzle exit area [m^2]	N/A	3.14	[39]
SRM specific impulse [s]	N/A	294.4	[39]
Beginning time T+[s]	0.0	262.0	[40]
Burn time [s]	209.0	182.0	[40]
Jettison time T+[s]	215.0	545.0	[40]

Ariane-5ECA

Input [Unit]	Value			Reference
Fairing diameter [m]	5.4			[42]
Fairing length [m]	17.0			[42]
Fairing nose diameter [m]	1.0			[42]
Fairing nose length [m]	7.0			[42]
Fairing mass [kg]	2400.0			[42]
Fairing jettison time [s]	208.0			[42]
Latitude of the launch [°]	5.23			[42]
Payload mass [kg]	Between 4241 and 10865			[43]
Stages	Stage 0		Stage 1	
Length [m]	35.8		4.8	[28]
Diameter [m]	5.4		5.4	[28]
Propulsion System	P.0 (x2)	P.1 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	H_2	H_2	[42]
Oxidizer	Al	O_2	O_2	[42]
Propellant mass [tonnes]	240.0	173.0	14.9	[42]
Structure mass [tonnes]	37.0	15.0	4.54	[42]
Sea level thrust [kN]	5520.0	960.0	67.0	[42]
Vacuum thrust [kN]	Thrust profile (Solid)	1390.0	67.0	[42] & [44]
SRM nozzle exit area [m^2]	7.0	N/A	N/A	[45]
SRM specific impulse [s]	275.0	N/A	N/A	[42]
Beginning time T+[s]	7.0	0.0	540.0	[42]
Burn time [s]	130.0	540.0	945.0	[42]
Jettison time T+[s]	137.0	540.0	1485.0	[42]

Ariane-5ES

Input [Unit]	Value			Reference
Fairing diameter [m]	5.4			[42]
Fairing length [m]	17.0			[42]
Fairing nose diameter [m]	1.0			[42]
Fairing nose length [m]	7.0			[42]
Fairing mass [kg]	2600.0			[46]
Fairing jettison time [s]	213.0			[46]
Latitude of the launch [°]	5.23			[42]
Payload mass [kg]	Between 3282 and 20060			[43]
Stages	Stage 0		Stage 1	
Length [m]	35.8		3.36	[28] [46]
Diameter [m]	5.4		5.4	[28]
Propulsion System	P.0 (x2)	P.1 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	H_2	H_2	[42]
Oxidizer	Al	O_2	O_2	[42]
Propellant mass [tonnes]	240.0	173.0	10.0	[42] [47]
Structure mass [tonnes]	37.0	15.0	1.3	[42] [47]
Sea level thrust [kN]	5520.0	960.0	29.0	[42] [47]
Vacuum thrust [kN]	Thrust profile (Solid)	1390.0	29.0	[44] [47]
SRM nozzle exit area [m^2]	7.0	N/A	N/A	[45]
SRM specific impulse [s]	275.0	N/A	N/A	[42]
Beginning time T+[s]	7.0	0.0	540.0	[42] [46]
Burn time [s]	130.0	528.0	1000.0	[42] [46]
Jettison time T+[s]	144.0	534.0	1540.0	[42] [46]

Ariane-5GS

Input [Unit]	Value			Reference
Fairing diameter [m]	5.4			[48]
Fairing length [m]	13.8			[48]
Fairing nose diameter [m]	1.0			[48]
Fairing nose length [m]	7.0			[48]
Fairing mass [kg]	2100.0			[48]
Fairing jettison time [s]	190.0			[48]
Latitude of the launch [°]	5.23			[48]
Payload mass [kg]	5954			[48]
Stages	Stage 0		Stage 1	
Length [m]	30.53		3.36	[48]
Diameter [m]	5.4		5.4	[48]
Propulsion System	P.0 (x2)	P.1 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	H_2	UDMH	[48]
Oxidizer	Al	O_2	N_2O_4	[48]
Propellant mass [tonnes]	240.0	158.0	10.0	[48]
Structure mass [tonnes]	31.0	13.6	2.7	[48]
Sea level thrust [kN]	5520.0	885.0	29.0	[48]
Vacuum thrust [kN]	Thrust profile (Solid)	1140.0	29.0	[48]
SRM nozzle exit area [m^2]	7.0	N/A	N/A	[45]
SRM specific impulse [s]	275.0	N/A	N/A	[42]
Beginning time T+[s]	7.0	0.0	589.0	[48]
Burn time [s]	128.0	576.0	1000.0	[48]
Jettison time T+[s]	140.0	582.0	1589.0	[48]

Atlas-5(401)

Input [Unit]	Value		Reference
Fairing diameter [m]	4.2		[49]
Fairing length [m]	12.9		[49]
Fairing nose diameter [m]	0.9		[49]
Fairing nose length [m]	6.755		[49]
Fairing mass [kg]	2305.0		[49]
Fairing jettison time [s]	268.0		[49]
Latitude of the launch [°]	28.56 or 34.73		[49]
Payload mass [kg]	Between 721 and 7492		[31] [32] [33]
Stages	Stage 0	Stage 1	
Length [m]	32.46	12.68	[49]
Diameter [m]	3.8	3.8	[49]
Propulsion System	P.0 (x1)	P.0 (x1)	
Fuel	RP-1	H_2	[49]
Oxidizer	O_2	O_2	[49]
Propellant mass [tonnes]	284.089	13.887	[49]
Structure mass [tonnes]	22.183	2.243	[49]
Sea level thrust [kN]	3827.0	99.2	[49]
Vacuum thrust [kN]	4152.0	99.2	[49]
SRM nozzle exit area [m^2]	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	
Beginning time T+[s]	0.0	260.0	[49]
Burn time [s]	240.0	686.0	[49]
Jettison time T+[s]	250.0	946.0	[49]

Atlas-5(411)

Input [Unit]	Value			Reference
Fairing diameter [m]	4.2			[49]
Fairing length [m]	12.9			[49]
Fairing nose diameter [m]	0.9			[49]
Fairing nose length [m]	6.755			[49]
Fairing mass [kg]	2305.0			[49]
Fairing jettison time [s]	268.0			[49]
Latitude of the launch [°]	34.73 or 28.56			[49]
Payload mass [kg]	Between 2100 and 6500			[31] [32] [33]
Stages	Stage 0		Stage 1	
Length [m]	32.46		12.68	[49]
Diameter [m]	3.8		3.8	[49]
Propulsion System	P.0 (x1)	P.1 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	RP-1	H_2	[49]
Oxidizer	Al	O_2	O_2	[49]
Propellant mass [tonnes]	44.2	284.089	13.887	[49]
Structure mass [tonnes]	5.1	22.183	2.243	[49]
Sea level thrust [kN]	1662.75	3827.0	99.2	[49]
Vacuum thrust [kN]	Thrust profile (Solid)	4152.0	99.2	[49] & [50]
SRM nozzle exit area [m^2]	1.75	N/A	N/A	[50]
SRM specific impulse [s]	279.0	N/A	N/A	[50]
Beginning time T+[s]	0.0	0.0	260.0	[49]
Burn time [s]	95.0	240.0	686.0	[49]
Jettison time T+[s]	110.0	250.0	946.0	[49]

Atlas-5(421)

Input [Unit]	Value			Reference
Fairing diameter [m]	4.2			[49]
Fairing length [m]	12.9			[49]
Fairing nose diameter [m]	0.9			[49]
Fairing nose length [m]	6.755			[49]
Fairing mass [kg]	2305.0			[49]
Fairing jettison time [s]	268.0			[49]
Latitude of the launch [°]	28.56			[49]
Payload mass [kg]	Between 5300 and 5987			[31] [32] [33]
Stages	Stage 0		Stage 1	
Length [m]	32.46		12.68	[49]
Diameter [m]	3.8		3.8	[49]
Propulsion System	P.0 (x2)	P.1 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	RP-1	H_2	[49]
Oxidizer	Al	O_2	O_2	[49]
Propellant mass [tonnes]	44.2	284.089	13.887	[49]
Structure mass [tonnes]	5.1	22.183	2.243	[49]
Sea level thrust [kN]	1662.75	3827.0	99.2	[49]
Vacuum thrust [kN]	Thrust profile (Solid)	4152.0	99.2	[49] & [50]
SRM nozzle exit area [m^2]	1.75	N/A	N/A	[50]
SRM specific impulse [s]	279.0	N/A	N/A	[50]
Beginning time T+[s]	0.0	0.0	260.0	[49]
Burn time [s]	95.0	240.0	686.0	[49]
Jettison time T+[s]	110.0	250.0	946.0	[49]

Atlas-5(431)

Input [Unit]	Value			Reference
Fairing diameter [m]	4.2			[49]
Fairing length [m]	12.9			[49]
Fairing nose diameter [m]	0.9			[49]
Fairing nose length [m]	6.755			[49]
Fairing mass [kg]	2305.0			[49]
Fairing jettison time [s]	268.0			[49]
Latitude of the launch [°]	28.56			[49]
Payload mass [kg]	Between 5663 and 6637			[31] [32] [33]
Stages	Stage 0		Stage 1	
Length [m]	32.46		12.68	[49]
Diameter [m]	3.8		3.8	[49]
Propulsion System	P.0 (x3)	P.1 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	RP-1	H_2	[49]
Oxidizer	Al	O_2	O_2	[49]
Propellant mass [tonnes]	44.2	284.089	13.887	[49]
Structure mass [tonnes]	5.1	22.183	2.243	[49]
Sea level thrust [kN]	1662.75	3827.0	99.2	[49]
Vacuum thrust [kN]	Thrust profile (Solid)	4152.0	99.2	[49] & [50]
SRM nozzle exit area [m^2]	1.75	N/A	N/A	[50]
SRM specific impulse [s]	279.0	N/A	N/A	[50]
Beginning time T+[s]	0.0	0.0	260.0	[49]
Burn time [s]	95.0	240.0	686.0	[49]
Jettison time T+[s]	110.0	250.0	946.0	[49]

Atlas-5(501)

Input [Unit]	Value		Reference
Fairing diameter [m]	5.4		[49]
Fairing length [m]	12.9		[49]
Fairing nose diameter [m]	1.4		[49]
Fairing nose length [m]	8.528		[49]
Fairing mass [kg]	3525.0		[49]
Fairing jettison time [s]	208.0		[49]
Latitude of the launch [°]	28.56 or 34.73		[49]
Payload mass [kg]	Between 3450 and 5000		[31] [32] [33]
Stages	Stage 0	Stage 1	
Length [m]	32.46	12.68	[49]
Diameter [m]	3.8	3.8	[49]
Propulsion System	P.0 (x1)	P.0 (x1)	
Fuel	RP-1	H_2	[49]
Oxidizer	O_2	O_2	[49]
Propellant mass [tonnes]	284.089	20.839	[49]
Structure mass [tonnes]	23.848	2.737	[49]
Sea level thrust [kN]	3827.0	99.2	[49]
Vacuum thrust [kN]	4152.0	99.2	[49]
SRM nozzle exit area [m^2]	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	
Beginning time T+[s]	0.0	268.0	[49]
Burn time [s]	264.0	686.0	[49]
Jettison time T+[s]	268.0	954.0	[49]

Atlas-5(531)

Input [Unit]	Value			Reference
Fairing diameter [m]	5.4			[49]
Fairing length [m]	12.9			[49]
Fairing nose diameter [m]	1.4			[49]
Fairing nose length [m]	8.528			[49]
Fairing mass [kg]	3525.0			[49]
Fairing jettison time [s]	208.0			[49]
Latitude of the launch [°]	28.56			[49]
Payload mass [kg]	Between 6170 and 6170			[31] [32] [33]
Stages	Stage 0		Stage 1	
Length [m]	32.46		12.68	[49]
Diameter [m]	3.8		3.8	[49]
Propulsion System	P.0 (x3)	P.1 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	RP-1	H_2	[49]
Oxidizer	Al	O_2	O_2	[49]
Propellant mass [tonnes]	44.2	284.089	20.839	[49]
Structure mass [tonnes]	5.1	23.848	2.737	[49]
Sea level thrust [kN]	1662.75	3827.0	99.2	[49]
Vacuum thrust [kN]	Thrust profile (Solid)	4152.0	99.2	[49] & [50]
SRM nozzle exit area [m^2]	1.75	N/A	N/A	[50]
SRM specific impulse [s]	279.0	N/A	N/A	[50]
Beginning time T+[s]	0.0	0.0	268.0	[49]
Burn time [s]	95.0	264.0	686.0	[49]
Jettison time T+[s]	110.0	268.0	954.0	[49]

Atlas-5(541)

Input [Unit]	Value			Reference
Fairing diameter [m]	5.4			[49]
Fairing length [m]	12.9			[49]
Fairing nose diameter [m]	1.4			[49]
Fairing nose length [m]	8.528			[49]
Fairing mass [kg]	3525.0			[49]
Fairing jettison time [s]	208.0			[49]
Latitude of the launch [°]	28.56 or 34.73			[49]
Payload mass [kg]	Between 3400 and 5192			[31] [32] [33]
Stages	Stage 0		Stage 1	
Length [m]	32.46		12.68	[49]
Diameter [m]	3.8		3.8	[49]
Propulsion System	P.0 (x4)	P.1 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	RP-1	H_2	[49]
Oxidizer	Al	O_2	O_2	[49]
Propellant mass [tonnes]	44.2	284.089	20.839	[49]
Structure mass [tonnes]	5.1	23.848	2.737	[49]
Sea level thrust [kN]	1662.75	3827.0	99.2	[49]
Vacuum thrust [kN]	Thrust profile (Solid)	4152.0	99.2	[49] & [50]
SRM nozzle exit area [m^2]	1.75	N/A	N/A	[50]
SRM specific impulse [s]	279.0	N/A	N/A	[50]
Beginning time T+[s]	0.0	0.0	268.0	[49]
Burn time [s]	95.0	264.0	686.0	[49]
Jettison time T+[s]	110.0	268.0	954.0	[49]

Atlas-5(551)

Input [Unit]	Value			Reference
Fairing diameter [m]	5.4			[49]
Fairing length [m]	12.9			[49]
Fairing nose diameter [m]	1.4			[49]
Fairing nose length [m]	8.528			[49]
Fairing mass [kg]	3525.0			[49]
Fairing jettison time [s]	208.0			[49]
Latitude of the launch [°]	28.56			[49]
Payload mass [kg]	Between 3625 and 6740			[31] [32] [33]
Stages	Stage 0		Stage 1	
Length [m]	32.46		12.68	[49]
Diameter [m]	3.8		3.8	[49]
Propulsion System	P.0 (x5)	P.1 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	RP-1	H_2	[49]
Oxidizer	Al	O_2	O_2	[49]
Propellant mass [tonnes]	44.2	284.089	20.839	[49]
Structure mass [tonnes]	5.1	23.848	2.737	[49]
Sea level thrust [kN]	1662.75	3827.0	99.2	[49]
Vacuum thrust [kN]	Thrust profile (Solid)	4152.0	99.2	[49] & [50]
SRM nozzle exit area [m^2]	1.75	N/A	N/A	[50]
SRM specific impulse [s]	279.0	N/A	N/A	[50]
Beginning time T+[s]	0.0	0.0	268.0	[49]
Burn time [s]	95.0	264.0	686.0	[49]
Jettison time T+[s]	110.0	268.0	954.0	[49]

CZ-2C

Input [Unit]	Value			Reference
Fairing diameter [m]	3.35			[51]
Fairing length [m]	8.368			[51]
Fairing nose diameter [m]	1.0			[51]
Fairing nose length [m]	3.387			[51]
Fairing mass [kg]	800.0			[52]
Fairing jettison time [s]	232.0			[51]
Latitude of the launch [°]	38.85 or 40.96 or 28.25			[51]
Payload mass [kg]	Between 300 and 2000			[53]
Stages	Stage 0	Stage 1		
Length [m]	25.72	7.757		[51]
Diameter [m]	3.35	3.35		[51]
Propulsion System	P.0 (x1)	P.0 (x1)	P.1 (x1)	
Fuel	UDMH	UDMH	UDMH	[51]
Oxidizer	N_2O_4	N_2O_4	N_2O_4	[51]
Propellant mass [tonnes]	162.706	46.427	8.24	[51]
Structure mass [tonnes]	8.56	6.0	0.0	Assumed from [54] and [55]
Sea level thrust [kN]	2961.6	741.4	47.2	[51]
Vacuum thrust [kN]	3695.0	741.4	47.2	[51] & S0 assumed from [56] and acceleration profile
SRM nozzle exit area [m^2]	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	
Beginning time T+[s]	0.0	122.0	122.0	[51]
Burn time [s]	120.0	183.0	444.0	[51]
Jettison time T+[s]	122.0	570.0	570.0	[51]

CZ-2C SMA

Input [Unit]	Value				Reference
Fairing diameter [m]	3.35				[51]
Fairing length [m]	8.368				[51]
Fairing nose diameter [m]	1.0				[51]
Fairing nose length [m]	3.387				[51]
Fairing mass [kg]	800.0				[52]
Fairing jettison time [s]	232.0				[51]
Latitude of the launch [°]	38.85 or 40.96				[51]
Payload mass [kg]	Between 1050 and 1540				[53]
Stages	Stage 0	Stage 1		Stage 2	
Length [m]	25.72	7.757		1.5	[51]
Diameter [m]	3.35	3.35		3.35	[51]
Propulsion System	P.0 (x1)	P.0 (x1)	P.1 (x1)	P.0 (x1)	
Fuel	UDMH	UDMH	UDMH	NH_4ClO_4	[51]
Oxidizer	N_2O_4	N_2O_4	N_2O_4	Al	[51]
Propellant mass [tonnes]	162.706	46.427	8.24	0.122	[51]
Structure mass [tonnes]	8.56	6.0	0.0	0.038	Assumed from [54] and [55]
Sea level thrust [kN]	2961.6	741.4	47.2	10.78	[51]
Vacuum thrust [kN]	3695.0	741.4	47.2	x	[51] & S0 assumed from [56] and acceleration profile
SRM nozzle exit area [m^2]	N/A	N/A	N/A	x	
SRM specific impulse [s]	N/A	N/A	N/A	x	
Beginning time T+[s]	0.0	122.0	122.0	x	[51]
Burn time [s]	120.0	183.0	444.0	x	[51]
Jettison time T+[s]	122.0	570.0	570.0	x	[51]

CZ-2C (YZ1S)

Input [Unit]	Value				Reference
Fairing diameter [m]	3.35				[51]
Fairing length [m]	8.368				[51]
Fairing nose diameter [m]	1.0				[51]
Fairing nose length [m]	3.387				[51]
Fairing mass [kg]	800.0				[52]
Fairing jettison time [s]	232.0				[51]
Latitude of the launch [°]	40.96				[51]
Payload mass [kg]	2000				[53]
Stages	Stage 0	Stage 1		Stage 2	
Length [m]	25.72	7.757		2.0	[51]
Diameter [m]	3.35	3.35		3.35	[51]
Propulsion System	P.0 (x1)	P.0 (x1)	P.1 (x1)	P.0 (x1)	
Fuel	UDMH	UDMH	UDMH	UDMH	[51]
Oxidizer	N_2O_4	N_2O_4	N_2O_4	N_2O_4	[51]
Propellant mass [tonnes]	162.706	46.427	8.24	6.0	[51] YZ1 from YZ2 (see CZ5/YZ2)
Structure mass [tonnes]	8.56	6.0	0.0	0.0	Assumed from [54] and [55]
Sea level thrust [kN]	2961.6	741.4	47.2	6.5	[51] [57]
Vacuum thrust [kN]	3695.0	741.4	47.2	6.5	[51] & S0 assumed from [56] and acceleration profile [57]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	N/A	
Beginning time T+[s]	0.0	122.0	122.0	x	[51]
Burn time [s]	120.0	183.0	444.0	x	[51]
Jettison time T+[s]	122.0	570.0	570.0	x	[51]

YuanZheng 1S (YZ-1S) is an upgrade of YZ-1. However, due to the lack of data, we kept that of YZ-1 (which is already poor). According to the China Aerospace Science and Technology Corporation (CASC), Yuanzheng-1S differs from Yuanzheng-1 in several ways: the engine of Yuanzheng-1S allows for suborbital ignition as opposed to orbital ignition, and also has a reduced flight time of 10 minutes as compared to its parent's 6-7 hours. The satellites are also positioned differently, with the ability for the first satellite to separate after the first ignition, as compared to a later separation in Yuanzheng-1.

CZ-2D

Input [Unit]	Value			Reference
Fairing diameter [m]	3.35			[58]
Fairing length [m]	6.983			[58]
Fairing nose diameter [m]	1.0			[58]
Fairing nose length [m]	3.387			[58]
Fairing mass [kg]	750.0			[52]
Fairing jettison time [s]	214.0			[59]
Latitude of the launch [°]	40.96 or 38.85			[51]
Payload mass [kg]	Between 223 and 3363			[53]
Stages	Stage 0	Stage 1		
Length [m]	27.91	10.9		[58]
Diameter [m]	3.35	3.35		[58]
Propulsion System	P.0 (x1)	P.0 (x1)	P.1 (x1)	
Fuel	UDMH	UDMH	UDMH	[58]
Oxidizer	N_2O_4	N_2O_4	N_2O_4	[58]
Propellant mass [tonnes]	182.0	42.762	9.938	[58]
Structure mass [tonnes]	9.6	5.9	0.0	[58]
Sea level thrust [kN]	2961.6	742.04	47.1	
Vacuum thrust [kN]	3295.0	742.04	47.1	[58] & 1 stage estimated from CZ-2 and [56] and acceleration profiles
SRM nozzle exit area [m^2]	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	
Beginning time T+[s]	0.0	154.0	154.0	[59]
Burn time [s]	153.0	179.0	598.0	[59]
Jettison time T+[s]	154.0	767.0	767.0	[59]

CZ-2F/G

Input [Unit]	Value				Reference
Fairing diameter [m]	3.6				Image: [60]
Fairing length [m]	19.11				[61] [62]
Fairing nose diameter [m]	1.7				Image: [60]
Fairing nose length [m]	4.0				Image: [60]
Fairing mass [kg]	3000.0				[63]
Fairing jettison time [s]	215.0				[28]
Latitude of the launch [°]	40.96				[28]
Payload mass [kg]	Between 8082 and 8082				[53]
Stages	Stage 0		Stage 1		
Length [m]	23.7		15.5		[28]
Diameter [m]	3.35		3.35		[28]
Propulsion System	P.0 (x4)	P.1 (x1)	P.0 (x1)	P.1 (x1)	
Fuel	UDMH	UDMH	UDMH	UDMH	[28]
Oxidizer	N_2O_4	N_2O_4	N_2O_4	N_2O_4	[28]
Propellant mass [tonnes]	35.75	187.0	64.5	21.5	[28]
Structure mass [tonnes]	3.2	9.5	5.5	0.0	[28]
Sea level thrust [kN]	740.4	2961.6	742.0	47.2	[28]
Vacuum thrust [kN]	815.859	3265.398	742.0	47.2	[56]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	N/A	
Beginning time T+[s]	0.0	0.0	167.0	167.0	[64]
Burn time [s]	128.0	166.0	462.0	521.0	[64]
Jettison time T+[s]	128.0	166.0	521.0	521.0	[64]

CZ-2F/T

Input [Unit]	Value				Reference
Fairing diameter [m]	3.8				Image: [60]
Fairing length [m]	12.78				[61] [62]
Fairing nose diameter [m]	1.7				Image: [60]
Fairing nose length [m]	4.0				Image: [60]
Fairing mass [kg]	3000.0				[63]
Fairing jettison time [s]	215.0				[28]
Latitude of the launch [°]	40.96				[28]
Payload mass [kg]	Between 8506 and 9000				[53]
Stages	Stage 0		Stage 1		
Length [m]	23.7		15.5		[28]
Diameter [m]	3.35		3.35		[28]
Propulsion System	P.0 (x4)	P.1 (x1)	P.0 (x1)	P.1 (x1)	
Fuel	UDMH	UDMH	UDMH	UDMH	[28]
Oxidizer	N_2O_4	N_2O_4	N_2O_4	N_2O_4	[28]
Propellant mass [tonnes]	35.75	187.0	64.5	21.5	[28]
Structure mass [tonnes]	3.2	9.5	5.5	0.0	[28]
Sea level thrust [kN]	740.4	2961.6	742.0	47.2	[28]
Vacuum thrust [kN]	815.859	3265.398	742.0	47.2	[56]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	N/A	
Beginning time T+[s]	0.0	0.0	167.0	167.0	[28]
Burn time [s]	128.0	166.0	462.0	521.0	[28]
Jettison time T+[s]	128.0	166.0	521.0	521.0	[28]

CZ-3A

Input [Unit]	Value				Reference
Fairing diameter [m]	3.35				[64]
Fairing length [m]	8.887				[64]
Fairing nose diameter [m]	1.828				[64]
Fairing nose length [m]	4.787				[64]
Fairing mass [kg]	800.0				[65]
Fairing jettison time [s]	237.0				[64]
Latitude of the launch [°]	28.25				[64]
Payload mass [kg]	Between 1380 and 4200				[53]
Stages	Stage 0	Stage 1		Stage 2	
Length [m]	23.272	11.276		12.375	[64]
Diameter [m]	3.35	3.35		3.35	[64]
Propulsion System	P.0 (x1)	P.0 (x1)	P.1 (x1)	P.0 (x1)	
Fuel	UDMH	UDMH	UDMH	H_2	[64]
Oxidizer	N_2O_4	N_2O_4	N_2O_4	O_2	[64]
Propellant mass [tonnes]	171.8	30.644	1.956	13.65	[64]
Structure mass [tonnes]	10.0	4.0	0.0	3.0	[28]
Sea level thrust [kN]	2961.6	742.0	47.1	167.17	[64]
Vacuum thrust [kN]	3300.0	742.0	47.1	167.17	[64]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	N/A	
Beginning time T+[s]	0.0	149.0	149.0	265.0	[64]
Burn time [s]	149.0	111.0	117.0	353.0	[64]
Jettison time T+[s]	149.0	267.0	267.0	619.0	[64]

CZ-3B/G3Z

Input [Unit]	Value					Reference
Fairing diameter [m]	3.7					[64]
Fairing length [m]	8.887					[64]
Fairing nose diameter [m]	1.585					[64]
Fairing nose length [m]	3.172					[64]
Fairing mass [kg]	1800.0					[65]
Fairing jettison time [s]	215.0					[64]
Latitude of the launch [°]	28.25					[64]
Payload mass [kg]	Between 3780 and 3780					[53]
Stages	Stage 0		Stage 1		Stage 2	
Length [m]	23.272		12.92		12.375	[64]
Diameter [m]	3.35		3.35		3.35	[64]
Propulsion System	P.0 (x4)	P.1 (x1)	P.0 (x1)	P.1 (x1)	P.0 (x1)	
Fuel	UDMH	UDMH	UDMH	UDMH	H_2	[64]
Oxidizer	N_2O_4	N_2O_4	N_2O_4	N_2O_4	O_2	[64]
Propellant mass [tonnes]	37.7	171.8	46.4	3.0	9.1	[64]
Structure mass [tonnes]	3.0	12.12	3.848	0.0	2.742	Estimated from [66]
Sea level thrust [kN]	740.4	2961.6	742.0	47.1	167.17	[64]
Vacuum thrust [kN]	816.0	3300.0	742.0	47.1	167.17	[64] & S1 estimated from CZ2C table and [56] and acceleration profiles
SRM nozzle exit area [m^2]	N/A	N/A	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	N/A	N/A	
Beginning time T+[s]	0.0	0.0	147.0	147.0	332.0	[64]
Burn time [s]	128.0	145.0	179.0	184.0	270.0	[64]
Jettison time T+[s]	129.0	147.0	332.0	332.0	602.0	[64]

CZ-3B/G2

Input [Unit]	Value					Reference
Fairing diameter [m]	4.0					[64]
Fairing length [m]	9.777					[64]
Fairing nose diameter [m]	1.178					[64]
Fairing nose length [m]	3.777					[64]
Fairing mass [kg]	1800.0					[65]
Fairing jettison time [s]	235.0					[64]
Latitude of the launch [°]	28.25					[64]
Payload mass [kg]	Between 3800 and 5380					[53]
Stages	Stage 0		Stage 1		Stage 2	
Length [m]	24.76		12.92		12.375	[64]
Diameter [m]	3.35		3.35		3.35	[64]
Propulsion System	P.0 (x4)	P.1 (x1)	P.0 (x1)	P.1 (x1)	P.0 (x1)	
Fuel	UDMH	UDMH	UDMH	UDMH	H_2	[64]
Oxidizer	N_2O_4	N_2O_4	N_2O_4	N_2O_4	O_2	[64]
Propellant mass [tonnes]	41.1	186.2	46.4	3.0	9.1	[64]
Structure mass [tonnes]	3.0	12.12	3.848	0.0	2.742	Estimated from [66]
Sea level thrust [kN]	740.4	2961.6	742.0	47.1	167.17	[64]
Vacuum thrust [kN]	816.0	3265.0	742.0	47.1	167.17	[64]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	N/A	N/A	
Beginning time T+[s]	0.0	0.0	159.0	159.0	346.0	[64]
Burn time [s]	140.0	158.0	181.0	186.0	284.0	[64]
Jettison time T+[s]	141.0	159.0	346.0	346.0	630.0	[64]

CZ-3B/G2(YZ1)

Input [Unit]	Value						Reference
Fairing diameter [m]	4.0						[64]
Fairing length [m]	9.777						[64]
Fairing nose diameter [m]	1.178						[64]
Fairing nose length [m]	3.777						[64]
Fairing mass [kg]	1800.0						[65]
Fairing jettison time [s]	235.0						[64]
Latitude of the launch [°]	28.25						[64]
Payload mass [kg]	Between 2028 and 4600						[53]
Stages	Stage 0		Stage 1		Stage 2	Stage 3	
Length [m]	24.76		12.92		12.375	2.0	[64]
Diameter [m]	3.35		3.35		3.35	3.35	[64]
Propulsion System	P.0 (x4)	P.1 (x1)	P.0 (x1)	P.1 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	UDMH	UDMH	UDMH	UDMH	H_2	UDMH	[64]
Oxidizer	N_2O_4	N_2O_4	N_2O_4	N_2O_4	O_2	N_2O_4	[64]
Propellant mass [tonnes]	41.1	186.2	46.4	3.0	9.1	6.0	[64] YZ1 from YZ2 (see CZ5/YZ2)
Structure mass [tonnes]	3.0	12.12	3.848	0.0	2.742	0.0	Estimated from [66]
Sea level thrust [kN]	740.4	2961.6	742.0	47.1	167.17	6.5	[64] [57]
Vacuum thrust [kN]	816.0	3265.0	742.0	47.1	167.17	6.5	[64] [57] S1 estimated from CZ2C table and [56] and acceleration profiles
SRM nozzle exit area [m^2]	N/A	N/A	N/A	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	N/A	N/A	N/A	
Beginning time T+[s]	0.0	0.0	159.0	159.0	346.0	x	[64]
Burn time [s]	140.0	158.0	181.0	186.0	284.0	x	[64]
Jettison time T+[s]	141.0	159.0	346.0	346.0	630.0	x	[64]

CZ-3B

Input [Unit]	Value					Reference
Fairing diameter [m]	3.7					[64]
Fairing length [m]	8.887					[64]
Fairing nose diameter [m]	1.585					[64]
Fairing nose length [m]	3.172					[64]
Fairing mass [kg]	1800.0					[65]
Fairing jettison time [s]	215.0					[64]
Latitude of the launch [°]	28.25					[64]
Payload mass [kg]	Between 4100 and 4320					[53]
Stages	Stage 0		Stage 1		Stage 2	
Length [m]	23.272		12.92		12.375	[64]
Diameter [m]	3.35		3.35		3.35	[64]
Propulsion System	P.0 (x4)	P.1 (x1)	P.0 (x1)	P.1 (x1)	P.0 (x1)	
Fuel	UDMH	UDMH	UDMH	UDMH	H_2	[64]
Oxidizer	N_2O_4	N_2O_4	N_2O_4	N_2O_4	O_2	[64]
Propellant mass [tonnes]	37.7	171.8	46.4	3.0	9.1	[64]
Structure mass [tonnes]	3.0	12.12	3.848	0.0	2.742	Estimated from [66]
Sea level thrust [kN]	740.4	2961.6	742.0	47.1	167.17	[64]
Vacuum thrust [kN]	816.0	3300.0	742.0	47.1	167.17	[64] S1 estimated from CZ2C table and [56] and acceleration profiles
SRM nozzle exit area [m^2]	N/A	N/A	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	N/A	N/A	
Beginning time T+[s]	0.0	0.0	147.0	147.0	332.0	[64]
Burn time [s]	128.0	145.0	179.0	184.0	270.0	[64]
Jettison time T+[s]	129.0	147.0	332.0	332.0	602.0	[64]

CZ-3C

Input [Unit]	Value					Reference
Fairing diameter [m]	3.7					[64]
Fairing length [m]	8.887					[64]
Fairing nose diameter [m]	1.585					[64]
Fairing nose length [m]	3.172					[64]
Fairing mass [kg]	1800.0					[65]
Fairing jettison time [s]	259.0					[64]
Latitude of the launch [°]	28.25					[64]
Payload mass [kg]	Between 2500 and 4600					[53]
Stages	Stage 0		Stage 1		Stage 2	
Length [m]	23.272		12.92		12.375	[64]
Diameter [m]	3.35		3.35		3.35	[64]
Propulsion System	P.0 (x2)	P.1 (x1)	P.0 (x1)	P.1 (x1)	P.0 (x1)	
Fuel	UDMH	UDMH	UDMH	UDMH	H_2	[64]
Oxidizer	N_2O_4	N_2O_4	N_2O_4	N_2O_4	O_2	[64]
Propellant mass [tonnes]	37.7	171.8	46.4	3.0	9.1	[64]
Structure mass [tonnes]	3.0	12.12	3.848	0.0	2.742	Estimated from [66]
Sea level thrust [kN]	740.4	2961.6	742.0	47.1	167.17	[64]
Vacuum thrust [kN]	816.0	3300.0	742.0	47.1	167.17	[64] S1 estimated from CZ2C table and [56] and acceleration profiles
SRM nozzle exit area [m^2]	N/A	N/A	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	N/A	N/A	
Beginning time T+[s]	0.0	0.0	147.0	147.0	332.0	[64]
Burn time [s]	128.0	145.0	179.0	184.0	270.0	[64]
Jettison time T+[s]	129.0	147.0	332.0	332.0	602.0	[64]

CZ-3C/G2

Input [Unit]	Value					Reference
Fairing diameter [m]	3.7					[64]
Fairing length [m]	8.887					[64]
Fairing nose diameter [m]	1.585					[64]
Fairing nose length [m]	3.172					[64]
Fairing mass [kg]	1800.0					[65]
Fairing jettison time [s]	235.0					[64]
Latitude of the launch [°]	28.25					[64]
Payload mass [kg]	Between 2450 and 4600					[53]
Stages	Stage 0		Stage 1		Stage 2	
Length [m]	23.272		12.92		12.375	[64]
Diameter [m]	3.35		3.35		3.35	[64]
Propulsion System	P.0 (x2)	P.1 (x1)	P.0 (x1)	P.1 (x1)	P.0 (x1)	
Fuel	UDMH	UDMH	UDMH	UDMH	H_2	[64]
Oxidizer	N_2O_4	N_2O_4	N_2O_4	N_2O_4	O_2	[64]
Propellant mass [tonnes]	41.1	186.2	46.4	3.0	9.1	[64]
Structure mass [tonnes]	3.0	12.12	3.848	0.0	2.742	Estimated from [66]
Sea level thrust [kN]	740.4	2961.6	742.0	47.1	167.17	[64]
Vacuum thrust [kN]	816.0	3265.0	742.0	47.1	167.17	[64]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	N/A	N/A	
Beginning time T+[s]	0.0	0.0	159.0	159.0	346.0	[64]
Burn time [s]	140.0	158.0	181.0	186.0	284.0	[64]
Jettison time T+[s]	141.0	159.0	346.0	346.0	630.0	[64]

CZ-3C/G2(YZ1)

Input [Unit]	Value						Reference
Fairing diameter [m]	3.7						[64]
Fairing length [m]	8.887						[64]
Fairing nose diameter [m]	1.585						[64]
Fairing nose length [m]	3.172						[64]
Fairing mass [kg]	1800.0						[65]
Fairing jettison time [s]	235.0						[64]
Latitude of the launch [°]	28.25						[64]
Payload mass [kg]	Between 850 and 1014						[53]
Stages	Stage 0		Stage 1		Stage 2	Stage 3	
Length [m]	23.272		12.92		12.375	2.0	[64]
Diameter [m]	3.35		3.35		3.35	3.35	[64]
Propulsion System	P.0 (x2)	P.1 (x1)	P.0 (x1)	P.1 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	UDMH	UDMH	UDMH	UDMH	H_2	UDMH	[64]
Oxidizer	N_2O_4	N_2O_4	N_2O_4	N_2O_4	O_2	N_2O_4	[64]
Propellant mass [tonnes]	41.1	186.2	46.4	3.0	9.1	6.0	[64] YZ1 from YZ2 (see CZ5/YZ2)
Structure mass [tonnes]	3.0	12.12	3.848	0.0	2.742	0.0	Estimated from [66]
Sea level thrust [kN]	740.4	2961.6	742.0	47.1	167.17	6.5	[64] [57]
Vacuum thrust [kN]	816.0	3265.0	742.0	47.1	167.17	6.5	[64] [57]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	N/A	N/A	N/A	
Beginning time T+[s]	0.0	0.0	159.0	159.0	346.0	x	[64]
Burn time [s]	140.0	158.0	181.0	186.0	284.0	x	[64]
Jettison time T+[s]	141.0	159.0	346.0	346.0	630.0	x	[64]

CZ-4B

Input [Unit]	Value				Reference
Fairing diameter [m]	3.35				[28]
Fairing length [m]	8.48				[28]
Fairing nose diameter [m]	1.138				[28]
Fairing nose length [m]	3.0				[28]
Fairing mass [kg]	1350.0				[67]
Fairing jettison time [s]	140.0				[68]
Latitude of the launch [°]	38.85 or 40.96				[64]
Payload mass [kg]	Between 1350 and 2847				[53]
Stages	Stage 0	Stage 1		Stage 2	
Length [m]	24.66	10.41		4.0	[28]
Diameter [m]	3.35	3.35		3.35	[28]
Propulsion System	P.0 (x1)	P.0 (x1)	P.1 (x1)	P.0 (x1)	
Fuel	UDMH	UDMH	UDMH	UDMH	[28]
Oxidizer	N_2O_4	N_2O_4	N_2O_4	N_2O_4	[28]
Propellant mass [tonnes]	183.2	33.29	2.277	14.2	[28]
Structure mass [tonnes]	9.5	4.0	0.0	1.0	[28]
Sea level thrust [kN]	2961.6	742.04	47.1	100.0	[28]
Vacuum thrust [kN]	3295.0	742.04	47.1	100.0	[28] S1 based of CZ2C table and Space Launch Report [56] and estimated through acceleration profiles
SRM nozzle exit area [m^2]	N/A	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	N/A	
Beginning time T+[s]	0.0	171.0	171.0	309.0	[68]
Burn time [s]	170.0	127.0	137.0	941.0	[68]
Jettison time T+[s]	171.0	309.0	309.0	1250.0	[68]

CZ-4C

Input [Unit]	Value				Reference
Fairing diameter [m]	3.35				[28]
Fairing length [m]	8.48				[28]
Fairing nose diameter [m]	1.138				[28]
Fairing nose length [m]	3.0				[28]
Fairing mass [kg]	1350.0				[67]
Fairing jettison time [s]	140.0				[68]
Latitude of the launch [°]	38.85 or 40.96 or 28.25				[64]
Payload mass [kg]	Between 448 and 2950				[53]
Stages	Stage 0	Stage 1		Stage 2	
Length [m]	24.66	10.41		4.0	[28]
Diameter [m]	3.35	3.35		3.35	[28]
Propulsion System	P.0 (x1)	P.0 (x1)	P.1 (x1)	P.0 (x1)	
Fuel	UDMH	UDMH	UDMH	UDMH	[28]
Oxidizer	N_2O_4	N_2O_4	N_2O_4	N_2O_4	[28]
Propellant mass [tonnes]	183.2	33.29	2.277	14.2	[28]
Structure mass [tonnes]	9.5	4.0	0.0	1.0	[28]
Sea level thrust [kN]	2961.6	742.04	47.1	100.0	[28]
Vacuum thrust [kN]	3295.0	742.04	47.1	100.0	[28] S1 based of CZ2C table and Space Launch Report [56] and estimated through acceleration profiles
SRM nozzle exit area [m^2]	N/A	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	N/A	
Beginning time T+[s]	0.0	171.0	171.0	309.0	[68]
Burn time [s]	170.0	127.0	137.0	941.0	[68]
Jettison time T+[s]	171.0	309.0	309.0	1250.0	[68]

CZ-5

Input [Unit]	Value			Reference
Fairing diameter [m]	5.2			[69] [70]
Fairing length [m]	12.5			[69] [70]
Fairing nose diameter [m]	0.88			[69] [70]
Fairing nose length [m]	5.2			[69] [70]
Fairing mass [kg]	3500.0			[71]
Fairing jettison time [s]	295.0			[69]
Latitude of the launch [°]	19.62			[69]
Payload mass [kg]	7600			[53]
Stages	Stage 0		Stage 1	
Length [m]	31.02		12.0	[69]
Diameter [m]	5.0		5.0	[69]
Propulsion System	P.0 (x4)	P.1 (x1)	P.0 (x1)	
Fuel	H_2	RP-1	H_2	[69]
Oxidizer	O_2	O_2	O_2	[69]
Propellant mass [tonnes]	135.0	158.0	22.9	[69]
Structure mass [tonnes]	12.0	17.8	3.1	[69]
Sea level thrust [kN]	2400.0	1020.0	88.26	[69]
Vacuum thrust [kN]	2680.0	1400.0	88.26	[69]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	
Beginning time T+[s]	0.0	0.0	520.0	[69]
Burn time [s]	155.0	520.0	780.0	[69]
Jettison time T+[s]	155.0	520.0	1300.0	[69]

CZ-5(YZ2)

Input [Unit]	Value				Reference
Fairing diameter [m]	5.2				[69] [70]
Fairing length [m]	12.5				[69] [70]
Fairing nose diameter [m]	0.88				[69] [70]
Fairing nose length [m]	5.2				[69] [70]
Fairing mass [kg]	3500.0				[71]
Fairing jettison time [s]	295.0				[69]
Latitude of the launch [°]	19.62				[69]
Payload mass [kg]	4000				[53]
Stages	Stage 0		Stage 1	Stage 2	
Length [m]	31.02		12.0	2.0	[69]
Diameter [m]	5.0		5.0	5.0	[69]
Propulsion System	P.0 (x4)	P.1 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	H_2	RP-1	H_2	UDMH	[69]
Oxidizer	O_2	O_2	O_2	N_2O_4	[69]
Propellant mass [tonnes]	135.0	158.0	22.9	12.0	[69] YZ2 estimated from liftoff mass and payload capability in [72]
Structure mass [tonnes]	12.0	17.8	3.1	0.0	[69]
Sea level thrust [kN]	2400.0	1020.0	88.26	13.0	[69] [72]
Vacuum thrust [kN]	2680.0	1400.0	88.26	13.0	[69] [72]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	N/A	
Beginning time T+[s]	0.0	0.0	520.0	1300.0	[69]
Burn time [s]	155.0	520.0	780.0	1105.0	[69]
Jettison time T+[s]	155.0	520.0	1300.0	2405.0	[69]

CZ-6

Input [Unit]	Value			Reference
Fairing diameter [m]	2.6			[73]
Fairing length [m]	5.7			[73]
Fairing nose diameter [m]	0.89			[74]
Fairing nose length [m]	2.3			[74]
Fairing mass [kg]	1500.0			[75]
Fairing jettison time [s]	295.0			[69]
Latitude of the launch [°]	38.85			[76]
Payload mass [kg]	Between 624 and 1080			[53]
Stages	Stage 0	Stage 1	Stage 2	
Length [m]	15.0	7.3	1.8	[73]
Diameter [m]	3.35	3.35	3.35	[73]
Propulsion System	P.0 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	RP-1	RP-1	UDMH	[77]
Oxidizer	O_2	O_2	N_2O_4	[77]
Propellant mass [tonnes]	76.0	15.15	1.33	[77]
Structure mass [tonnes]	7.53	1.49	0.0	[77]
Sea level thrust [kN]	1180.0	175.0	6.5	[73]
Vacuum thrust [kN]	1340.0	175.0	6.5	[73]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	
Beginning time T+[s]	0.0	155.0	x	[77]
Burn time [s]	155.0	290.0	x	[77]
Jettison time T+[s]	155.0	445.0	x	[77]

CZ-7

Input [Unit]	Value			Reference
Fairing diameter [m]	4.2			[78]
Fairing length [m]	12.4			[78]
Fairing nose diameter [m]	1.3			[79]
Fairing nose length [m]	3.0			[79]
Fairing mass [kg]	2500.0			[80]
Fairing jettison time [s]	211.0			[81]
Latitude of the launch [°]	19.62			[82]
Payload mass [kg]	12914.5			[53]
Stages	Stage 0		Stage 1	
Length [m]	26.3		8.0	[77]
Diameter [m]	3.35		3.35	[77]
Propulsion System	P.0 (x1)	P.1 (x4)	P.0 (x1)	
Fuel	RP-1	RP-1	RP-1	[77]
Oxidizer	O_2	O_2	O_2	[77]
Propellant mass [tonnes]	174.0	75.5	65.0	[77]
Structure mass [tonnes]	12.5	6.0	5.5	[77]
Sea level thrust [kN]	2400.0	1200.0	720.0	[77]
Vacuum thrust [kN]	2680.0	1340.0	720.0	[77]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	
Beginning time T+[s]	0.0	0.0	188.0	[81]
Burn time [s]	178.0	175.0	390.0	[81]
Jettison time T+[s]	188.0	175.0	603.0	[81]

CZ-7(YZ1A)

Input [Unit]	Value				Reference
Fairing diameter [m]	4.2				[78]
Fairing length [m]	12.4				[78]
Fairing nose diameter [m]	1.3				[79]
Fairing nose length [m]	3.0				[79]
Fairing mass [kg]	2500.0				[80]
Fairing jettison time [s]	211.0				[81]
Latitude of the launch [°]	19.62				[82]
Payload mass [kg]	3418				[53]
Stages	Stage 0		Stage 1	Stage 2	
Length [m]	26.3		8.0	2.0	[77]
Diameter [m]	3.35		3.35	3.35	[77]
Propulsion System	P.0 (x1)	P.1 (x4)	P.0 (x1)	P.0 (x1)	
Fuel	RP-1	RP-1	RP-1	UDMH	[77]
Oxidizer	O_2	O_2	O_2	N_2O_4	[77]
Propellant mass [tonnes]	174.0	75.5	65.0	6.0	[77] YZ1 from YZ2 (see CZ-5/YZ2)
Structure mass [tonnes]	12.5	6.0	5.5	0.0	[77]
Sea level thrust [kN]	2400.0	1200.0	720.0	6.5	[77] [57]
Vacuum thrust [kN]	2680.0	1340.0	720.0	6.5	[77] [57]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	N/A	
Beginning time T+[s]	0.0	0.0	188.0	x	[81]
Burn time [s]	178.0	175.0	390.0	x	[81]
Jettison time T+[s]	188.0	175.0	603.0	x	[81]

Delta-2(7320-10C)

Input [Unit]	Value			Reference
Fairing diameter [m]	3.0			[83]
Fairing length [m]	8.9			[83]
Fairing nose diameter [m]	0.1			[83]
Fairing nose length [m]	3.9			[83]
Fairing mass [kg]	1040.0			[28]
Fairing jettison time [s]	293.0			[83]
Latitude of the launch [°]	34.73			[83]
Payload mass [kg]	Between 454 and 1440			[31] [32] [33]
Stages	Stage 0		Stage 1	
Length [m]	26.0		6.0	[28]
Diameter [m]	2.4		2.4	[28]
Propulsion System	P.0 (x3)	P.1 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	H_2	Aerozine 50	[28]
Oxidizer	Al	O_2	N_2O_4	[28]
Propellant mass [tonnes]	11.767	96.12	6.004	[28] & [39]
Structure mass [tonnes]	1.237	5.68	0.95	[28] & [39]
Sea level thrust [kN]	446.0	889.644	43.65	[28] & [84]
Vacuum thrust [kN]	Thrust profile (Solid)	1085.0	43.65	[28] & [39] & [84]
SRM nozzle exit area [m^2]	0.52	N/A	N/A	[39]
SRM specific impulse [s]	274.0	N/A	N/A	[39]
Beginning time T+[s]	0.0	0.0	278.0	[83]
Burn time [s]	64.0	264.0	388.0	[83]
Jettison time T+[s]	99.0	272.0	666.0	[83]

Delta-2(7420-10C)

Input [Unit]	Value			Reference
Fairing diameter [m]	3.0			[83]
Fairing length [m]	8.9			[83]
Fairing nose diameter [m]	0.1			[83]
Fairing nose length [m]	3.9			[83]
Fairing mass [kg]	1040.0			[28]
Fairing jettison time [s]	293.0			[83]
Latitude of the launch [°]	34.73			[83]
Payload mass [kg]	Between 1400 and 1900			[31] [32] [33]
Stages	Stage 0		Stage 1	
Length [m]	26.0		6.0	[28]
Diameter [m]	2.4		2.4	[28]
Propulsion System	P.0 (x4)	P.1 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	H_2	Aerozine 50	[28]
Oxidizer	Al	O_2	N_2O_4	[28]
Propellant mass [tonnes]	11.767	96.12	6.004	[28] & [39]
Structure mass [tonnes]	1.237	5.68	0.95	[28] & [39]
Sea level thrust [kN]	446.0	889.644	43.65	[28] & [84]
Vacuum thrust [kN]	Thrust profile (Solid)	1085.0	43.65	[28] & [39] & [84]
SRM nozzle exit area [m^2]	0.52	N/A	N/A	[39]
SRM specific impulse [s]	274.0	N/A	N/A	[39]
Beginning time T+[s]	0.0	0.0	278.0	[83]
Burn time [s]	64.0	264.0	388.0	[83]
Jettison time T+[s]	99.0	272.0	666.0	[83]

Delta-2(7920-10C)

Input [Unit]	Value				Reference
Fairing diameter [m]	3.0				[83]
Fairing length [m]	8.9				[83]
Fairing nose diameter [m]	0.1				[83]
Fairing nose length [m]	3.9				[83]
Fairing mass [kg]	1040.0				[28]
Fairing jettison time [s]	283.0				[83]
Latitude of the launch [°]	34.73 or 28.56				[83]
Payload mass [kg]	Between 1988 and 2800				[31] [32] [33]
Stages	Stage 0			Stage 1	
Length [m]	26.0			6.0	[28]
Diameter [m]	2.4			2.4	[28]
Propulsion System	P.0 (x6)	P.1 (x3)	P.2 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	NH_4ClO_4	H_2	Aerozine 50	[28]
Oxidizer	Al	Al	O_2	N_2O_4	[28]
Propellant mass [tonnes]	11.767	11.775	96.12	6.004	[28] & [39]
Structure mass [tonnes]	1.237	1.356	5.68	0.95	[28] & [39]
Sea level thrust [kN]	446.0	446.0	889.644	43.65	[28] & [84]
Vacuum thrust [kN]	Thrust profile (Solid)	Thrust profile (Solid)	1085.0	43.65	[28] & [39] & [84]
SRM nozzle exit area [m^2]	0.52	0.76	N/A	N/A	[39]
SRM specific impulse [s]	274.0	283.4	N/A	N/A	[39]
Beginning time T+[s]	0.0	66.0	0.0	278.0	[83]
Burn time [s]	64.0	63.0	264.0	388.0	[83]
Jettison time T+[s]	86.0	132.0	272.0	666.0	[83]

Delta-2(7920H-10C)

Input [Unit]	Value				Reference
Fairing diameter [m]	3.0				[83]
Fairing length [m]	8.9				[83]
Fairing nose diameter [m]	0.1				[83]
Fairing nose length [m]	3.9				[83]
Fairing mass [kg]	1040.0				[28]
Fairing jettison time [s]	303.0				[83]
Latitude of the launch [°]	28.56				[83]
Payload mass [kg]	614				[31] [32] [33]
Stages	Stage 0			Stage 1	
Length [m]	26.0			6.0	[28]
Diameter [m]	2.4			2.4	[28]
Propulsion System	P.0 (x6)	P.1 (x3)	P.2 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	NH_4ClO_4	H_2	Aerozine 50	[28]
Oxidizer	Al	Al	O_2	N_2O_4	[28]
Propellant mass [tonnes]	16.865	16.865	96.12	6.004	[28] & [39]
Structure mass [tonnes]	2.0	2.204	5.68	0.95	[28] & [39]
Sea level thrust [kN]	711.766	711.766	889.644	43.65	[28] & [84]
Vacuum thrust [kN]	Thrust profile (Solid)	Thrust profile (Solid)	1085.0	43.65	[28] & [39] & [84]
SRM nozzle exit area [m^2]	0.69	1.23	N/A	N/A	[39]
SRM specific impulse [s]	279.6	290.7	N/A	N/A	[39]
Beginning time T+[s]	0.0	66.0	0.0	278.0	[83]
Burn time [s]	63.0	63.0	264.0	388.0	[83]
Jettison time T+[s]	66.0	132.0	272.0	666.0	[83]

Delta-2(7925)

Input [Unit]	Value					Reference
Fairing diameter [m]	3.0					[83]
Fairing length [m]	8.9					[83]
Fairing nose diameter [m]	0.1					[83]
Fairing nose length [m]	3.9					[83]
Fairing mass [kg]	1040.0					[28]
Fairing jettison time [s]	303.0					[83]
Latitude of the launch [°]	28.56					[83]
Payload mass [kg]	Between 2032 and 2032					[31] [32] [33]
Stages	Stage 0			Stage 1	Stage 2	
Length [m]	26.0			6.0	2.0	[28]
Diameter [m]	2.4			2.4	2.4	[28]
Propulsion System	P.0 (x6)	P.1 (x3)	P.2 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	NH_4ClO_4	H_2	Aerozine 50	NH_4ClO_4	[28]
Oxidizer	Al	Al	O_2	N_2O_4	Al	[28]
Propellant mass [tonnes]	11.767	11.775	96.12	6.004	2.01	[28] & [39]
Structure mass [tonnes]	1.237	1.356	5.68	0.95	0.206	[28] & [39]
Sea level thrust [kN]	446.0	446.0	889.644	43.65	x	[28] & [84]
Vacuum thrust [kN]	Thrust profile (Solid)	Thrust profile (Solid)	1085.0	43.65	x	[28] & [39] & [84]
SRM nozzle exit area [m^2]	0.52	0.76	N/A	N/A	x	[39]
SRM specific impulse [s]	274.0	283.4	N/A	N/A	x	[39]
Beginning time T+[s]	0.0	66.0	0.0	278.0	x	[83]
Burn time [s]	64.0	63.0	264.0	388.0	x	[83]
Jettison time T+[s]	86.0	132.0	272.0	666.0	x	[83]

Delta-2(7925-10L)

Input [Unit]	Value					Reference
Fairing diameter [m]	3.0					[83]
Fairing length [m]	9.2					[83]
Fairing nose diameter [m]	0.2					[83]
Fairing nose length [m]	2.2					[83]
Fairing mass [kg]	1000.0					[28]
Fairing jettison time [s]	303.0					[83]
Latitude of the launch [°]	28.56					[83]
Payload mass [kg]	1039					[31] [32] [33]
Stages	Stage 0			Stage 1	Stage 2	
Length [m]	26.0			6.0	2.0	[28]
Diameter [m]	2.4			2.4	2.4	[28]
Propulsion System	P.0 (x6)	P.1 (x3)	P.2 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	NH_4ClO_4	H_2	Aerozine 50	NH_4ClO_4	[28]
Oxidizer	Al	Al	O_2	N_2O_4	Al	[28]
Propellant mass [tonnes]	11.767	11.775	96.12	6.004	2.01	[28] & [39]
Structure mass [tonnes]	1.237	1.356	5.68	0.95	0.206	[28] & [39]
Sea level thrust [kN]	446.0	446.0	889.644	43.65	x	[28] & [84]
Vacuum thrust [kN]	Thrust profile (Solid)	Thrust profile (Solid)	1085.0	43.65	x	[28] & [39] & [84]
SRM nozzle exit area [m^2]	0.52	0.76	N/A	N/A	x	[39]
SRM specific impulse [s]	274.0	283.4	N/A	N/A	x	[39]
Beginning time T+[s]	0.0	66.0	0.0	278.0	x	[83]
Burn time [s]	64.0	63.0	264.0	388.0	x	[83]
Jettison time T+[s]	86.0	132.0	272.0	666.0	x	[83]

Delta-4H

Input [Unit]	Value			Reference
Fairing diameter [m]	5.0			[85]
Fairing length [m]	11.3			[85]
Fairing nose diameter [m]	1.026			[85]
Fairing nose length [m]	6.023			[85]
Fairing mass [kg]	2935.0			[28]
Fairing jettison time [s]	275.0			[85]
Latitude of the launch [°]	28.56 or 34.73			[85]
Payload mass [kg]	Between 1700 and 12000			[31] [32] [33]
Stages	Stage 0		Stage 1	
Length [m]	40.8		12.0	[85]
Diameter [m]	5.0		5.0	[85]
Propulsion System	P.0 (x2)	P.1 (x1)	P.0 (x1)	
Fuel	H_2	H_2	H_2	[85]
Oxidizer	O_2	O_2	O_2	[85]
Propellant mass [tonnes]	204.0	204.0	20.4	[85] & [28]
Structure mass [tonnes]	28.0	28.0	3.49	[28]
Sea level thrust [kN]	3137.0	3137.0	110.0	[85]
Vacuum thrust [kN]	3560.0	3560.0	110.0	[85] & [86] & [39]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	[39]
SRM specific impulse [s]	N/A	N/A	N/A	[39]
Beginning time T+[s]	0.0	0.0	247.0	[85]
Burn time [s]	242.0	328.0	721.0	[85]
Jettison time T+[s]	245.0	334.0	4528.0	[85]

Delta-4H (upg.)

Input [Unit]	Value				Reference
Fairing diameter [m]	5.0				[85]
Fairing length [m]	11.3				[85]
Fairing nose diameter [m]	1.026				[85]
Fairing nose length [m]	6.023				[85]
Fairing mass [kg]	2935.0				[28]
Fairing jettison time [s]	275.0				[85]
Latitude of the launch [°]	28.56				[85]
Payload mass [kg]	Between 685 and 5400				[31] [32] [33]
Stages	Stage 0		Stage 1	Stage 2	
Length [m]	40.8		12.0	2.0	[85]
Diameter [m]	5.0		5.0	2.4	[85]
Propulsion System	P.0 (x2)	P.1 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	H_2	H_2	H_2	NH_4ClO_4	[85]
Oxidizer	O_2	O_2	O_2	Al	[85]
Propellant mass [tonnes]	204.0	204.0	20.4	2.01	[85] & [39] & [28]
Structure mass [tonnes]	28.0	28.0	3.49	0.206	[28] & [39]
Sea level thrust [kN]	3137.0	3137.0	110.0	x	[85]
Vacuum thrust [kN]	3560.0	3560.0	110.0	x	[85] & [86] & [39]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	x	[39]
SRM specific impulse [s]	N/A	N/A	N/A	x	[39]
Beginning time T+[s]	0.0	0.0	247.0	x	[85]
Burn time [s]	242.0	328.0	721.0	x	[85]
Jettison time T+[s]	245.0	334.0	4528.0	x	[85]

Delta-4M+(4,2)

Input [Unit]	Value			Reference
Fairing diameter [m]	4.0			[85]
Fairing length [m]	11.3			[85]
Fairing nose diameter [m]	0.94			[85]
Fairing nose length [m]	4.82			[85]
Fairing mass [kg]	1677.0			[28]
Fairing jettison time [s]	275.0			[85] & [87]
Latitude of the launch [°]	28.56 or 28.52			[85]
Payload mass [kg]	Between 1470 and 3133			[31] [32] [33]
Stages	Stage 0		Stage 1	
Length [m]	40.8		12.0	[85]
Diameter [m]	5.0		5.0	[85]
Propulsion System	P.0 (x2)	P.1 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	H_2	H_2	[85]
Oxidizer	Al	O_2	O_2	[85]
Propellant mass [tonnes]	29.7	199.6	15.308	[85] & [28]
Structure mass [tonnes]	3.4	27.0	2.85	[28]
Sea level thrust [kN]	1118.476	3137.0	110.0	[85]
Vacuum thrust [kN]	Thrust profile (Solid)	3560.0	110.0	[85] & [86] & [39]
SRM nozzle exit area [m^2]	0.94	N/A	N/A	[39]
SRM specific impulse [s]	274.2099898063201	N/A	N/A	[39]
Beginning time T+[s]	0.0	0.0	265.0	[85] & [87]
Burn time [s]	96.0	245.0	623.0	[85] & [87]
Jettison time T+[s]	100.0	251.0	888.0	[85] & [87]

Delta-4M+(4,2) (upg.)

Input [Unit]	Value				Reference
Fairing diameter [m]	4.0				[85]
Fairing length [m]	11.3				[85]
Fairing nose diameter [m]	0.94				[85]
Fairing nose length [m]	4.82				[85]
Fairing mass [kg]	1677.0				[28]
Fairing jettison time [s]	275.0				[85] & [87]
Latitude of the launch [°]	28.56				[85]
Payload mass [kg]	1400				[31] [32] [33]
Stages	Stage 0		Stage 1	Stage 2	
Length [m]	40.8		12.0	2.0	[85]
Diameter [m]	5.0		5.0	2.4	[85]
Propulsion System	P.0 (x2)	P.1 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	H_2	H_2	NH_4ClO_4	[85]
Oxidizer	Al	O_2	O_2	Al	[85]
Propellant mass [tonnes]	29.7	199.6	15.308	2.01	[85] & [39] & [28]
Structure mass [tonnes]	3.4	27.0	2.85	0.206	[28] & [39]
Sea level thrust [kN]	1118.476	3137.0	110.0	x	[85]
Vacuum thrust [kN]	Thrust profile (Solid)	3560.0	110.0	x	[85] & [86] & [39]
SRM nozzle exit area [m^2]	0.94	N/A	N/A	x	[39]
SRM specific impulse [s]	274.2099898063201	N/A	N/A	x	[39]
Beginning time T+[s]	0.0	0.0	265.0	x	[85] & [87]
Burn time [s]	96.0	245.0	623.0	x	[85] & [87]
Jettison time T+[s]	100.0	251.0	888.0	x	[85] & [87]

Delta-4M+(5,2)

Input [Unit]	Value			Reference
Fairing diameter [m]	5.0			[85]
Fairing length [m]	11.2			[85]
Fairing nose diameter [m]	1.026			[85]
Fairing nose length [m]	6.023			[85]
Fairing mass [kg]	2935.0			[28]
Fairing jettison time [s]	275.0			[85]
Latitude of the launch [°]	34.73			[85]
Payload mass [kg]	x			
Stages	Stage 0		Stage 1	
Length [m]	40.8		12.0	[85]
Diameter [m]	5.0		5.0	[85]
Propulsion System	P.0 (x2)	P.1 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	H_2	H_2	[85]
Oxidizer	Al	O_2	O_2	[85]
Propellant mass [tonnes]	29.7	204.0	20.4	[85] & [28]
Structure mass [tonnes]	3.4	28.0	3.49	[28]
Sea level thrust [kN]	1118.476	3137.0	110.0	[85]
Vacuum thrust [kN]	Thrust profile (Solid)	3560.0	110.0	[85] & [86] & [39]
SRM nozzle exit area [m^2]	0.94	N/A	N/A	[39]
SRM specific impulse [s]	274.2099898063201	N/A	N/A	[39]
Beginning time T+[s]	0.0	0.0	265.0	[85]
Burn time [s]	96.0	246.0	691.0	[85]
Jettison time T+[s]	100.0	252.0	956.0	[85]

Delta-4M+(5,2) (upg.)

Input [Unit]	Value				Reference
Fairing diameter [m]	5.0				[85]
Fairing length [m]	11.2				[85]
Fairing nose diameter [m]	1.026				[85]
Fairing nose length [m]	6.023				[85]
Fairing mass [kg]	2935.0				[28]
Fairing jettison time [s]	275.0				[85]
Latitude of the launch [°]	34.73				[85]
Payload mass [kg]	x				
Stages	Stage 0		Stage 1	Stage 2	
Length [m]	40.8		12.0	2.0	[85]
Diameter [m]	5.0		5.0	2.4	[85]
Propulsion System	P.0 (x2)	P.1 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	H_2	H_2	NH_4ClO_4	[85]
Oxidizer	Al	O_2	O_2	Al	[85]
Propellant mass [tonnes]	29.7	204.0	20.4	2.01	[85] & [39] & [28]
Structure mass [tonnes]	3.4	28.0	3.49	0.206	[28] & [39]
Sea level thrust [kN]	1118.476	3137.0	110.0	x	[85]
Vacuum thrust [kN]	Thrust profile (Solid)	3560.0	110.0	x	[85] & [86] & [39]
SRM nozzle exit area [m^2]	0.94	N/A	N/A	x	[39]
SRM specific impulse [s]	274.2099898063201	N/A	N/A	x	[39]
Beginning time T+[s]	0.0	0.0	265.0	x	[85]
Burn time [s]	96.0	246.0	691.0	x	[85]
Jettison time T+[s]	100.0	252.0	956.0	x	[85]

Delta-4M+(5,4)

Input [Unit]	Value			Reference
Fairing diameter [m]	5.0			[85]
Fairing length [m]	11.3			[85]
Fairing nose diameter [m]	1.026			[85]
Fairing nose length [m]	6.023			[85]
Fairing mass [kg]	2935.0			[28]
Fairing jettison time [s]	204.0			[85]
Latitude of the launch [°]	28.56			[85]
Payload mass [kg]	Between 5987 and 6000			[31] [32] [33]
Stages	Stage 0		Stage 1	
Length [m]	40.8		12.0	[85]
Diameter [m]	5.0		5.0	[85]
Propulsion System	P.0 (x4)	P.1 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	H_2	H_2	[85]
Oxidizer	Al	O_2	O_2	[85]
Propellant mass [tonnes]	29.7	204.0	20.4	[85] & [28]
Structure mass [tonnes]	3.4	28.0	3.49	[28]
Sea level thrust [kN]	1118.476	3137.0	110.0	[85]
Vacuum thrust [kN]	Thrust profile (Solid)	3560.0	110.0	[85] & [86] & [39]
SRM nozzle exit area [m^2]	0.94	N/A	N/A	[39]
SRM specific impulse [s]	274.2099898063201	N/A	N/A	[39]
Beginning time T+[s]	0.0	0.0	265.0	[85]
Burn time [s]	96.0	246.0	753.0	[85]
Jettison time T+[s]	100.0	252.0	5868.0	[85]

Delta-4M+(5,4) (upg.)

Input [Unit]	Value				Reference
Fairing diameter [m]	5.0				[85]
Fairing length [m]	11.3				[85]
Fairing nose diameter [m]	1.026				[85]
Fairing nose length [m]	6.023				[85]
Fairing mass [kg]	2935.0				[28]
Fairing jettison time [s]	204.0				[85]
Latitude of the launch [°]	28.56				[85]
Payload mass [kg]	Between 5987 and 5987				[31] [32] [33]
Stages	Stage 0		Stage 1	Stage 2	
Length [m]	40.8		12.0	2.0	[85]
Diameter [m]	5.0		5.0	2.4	[85]
Propulsion System	P.0 (x4)	P.1 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	H_2	H_2	NH_4ClO_4	[85]
Oxidizer	Al	O_2	O_2	Al	[85]
Propellant mass [tonnes]	29.7	204.0	20.4	2.01	[85] & [39] & [28]
Structure mass [tonnes]	3.4	28.0	3.49	0.206	[28] & [39]
Sea level thrust [kN]	1118.476	3137.0	110.0	x	[85]
Vacuum thrust [kN]	Thrust profile (Solid)	3560.0	110.0	x	[85] & [86] & [39]
SRM nozzle exit area [m^2]	0.94	N/A	N/A	x	[39]
SRM specific impulse [s]	274.2099898063201	N/A	N/A	x	[39]
Beginning time T+[s]	0.0	0.0	265.0	x	[85]
Burn time [s]	96.0	246.0	753.0	x	[85]
Jettison time T+[s]	100.0	252.0	5868.0	x	[85]

Dnepr

Input [Unit]	Value			Reference
Fairing diameter [m]	3.0			[28]
Fairing length [m]	5.25			[28]
Fairing nose diameter [m]	0.41			[28]
Fairing nose length [m]	3.17			[28]
Fairing mass [kg]	500.0			[88]
Fairing jettison time [s]	200.0			[89]
Latitude of the launch [°]	45.6 or 50.8			[90]
Payload mass [kg]	Between 450 and 2190			[31] [32] [33]
Stages	Stage 0	Stage 1	Stage 2	
Length [m]	22.0	5.7	1.0	[28]
Diameter [m]	3.0	3.0	3.0	[90]
Propulsion System	P.0 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	UDMH	UDMH	UDMH	[90]
Oxidizer	N_2O_4	N_2O_4	N_2O_4	[90]
Propellant mass [tonnes]	147.9	36.74	1.91	[90]
Structure mass [tonnes]	13.62	3.37	2.356	[90]
Sea level thrust [kN]	4166.0	760.0	18.633	[91]
Vacuum thrust [kN]	4522.827	760.0	18.633	[90]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	
Beginning time T+[s]	0.0	98.0	278.0	[89]
Burn time [s]	98.0	174.0	709.0	[89]
Jettison time T+[s]	98.0	278.0	987.0	[89]

Electron

Input [Unit]	Value		Reference
Fairing diameter [m]	1.2		[92]
Fairing length [m]	2.5		[92]
Fairing nose diameter [m]	0.278		[92]
Fairing nose length [m]	1.59		[92]
Fairing mass [kg]	44.0		[92]
Fairing jettison time [s]	187.0		[93]
Latitude of the launch [°]	-39.26		[92]
Payload mass [kg]	10		[31] [32] [33]
Stages	Stage 0	Stage 1	
Length [m]	12.1	2.4	[92]
Diameter [m]	1.2	1.2	[92]
Propulsion System	P.0 (x1)	P.0 (x1)	
Fuel	RP-1	RP-1	[92]
Oxidizer	O_2	O_2	[92]
Propellant mass [tonnes]	9.25	2.05	[94]
Structure mass [tonnes]	0.95	0.25	[94]
Sea level thrust [kN]	155.936	22.618	[95]
Vacuum thrust [kN]	187.522	22.618	[95]
SRM nozzle exit area [m^2]	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	
Beginning time T+[s]	0.0	157.0	[93]
Burn time [s]	151.0	389.0	[93]
Jettison time T+[s]	154.0	550.0	[93]

Electron (Curie)

Input [Unit]	Value			Reference
Fairing diameter [m]	1.2			[92]
Fairing length [m]	2.5			[92]
Fairing nose diameter [m]	0.278			[92]
Fairing nose length [m]	1.59			[92]
Fairing mass [kg]	44.0			[92]
Fairing jettison time [s]	187.0			[93]
Latitude of the launch [°]	-39.26			[92]
Payload mass [kg]	Between 19 and 78			[31] [32] [33]
Stages	Stage 0	Stage 1	Stage 2	
Length [m]	12.1	2.4	0.0	[92]
Diameter [m]	1.2	1.2	1.2	[92]
Propulsion System	P.0 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	RP-1	RP-1	RP-1	[92]
Oxidizer	O_2	O_2	O_2	[92]
Propellant mass [tonnes]	9.25	2.05	x	[94]
Structure mass [tonnes]	0.95	0.25	x	[94]
Sea level thrust [kN]	155.936	22.618	x	[95]
Vacuum thrust [kN]	187.522	22.618	x	[95]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	
Beginning time T+[s]	0.0	157.0	3000.0	[93]
Burn time [s]	151.0	389.0	90.0	[93]
Jettison time T+[s]	154.0	550.0	3360.0	[93]

Epsilon(CLPS)

Input [Unit]	Value				Reference
Fairing diameter [m]	2.6				[96]
Fairing length [m]	8.8				[96]
Fairing nose diameter [m]	0.204				[96]
Fairing nose length [m]	2.358				[96]
Fairing mass [kg]	1000.0				[96]
Fairing jettison time [s]	150.0				[97]
Latitude of the launch [°]	31.25				[96]
Payload mass [kg]	348				[31] [32] [33]
Stages	Stage 0	Stage 1	Stage 2	Stage 3	
Length [m]	11.7	4.3	2.3	1.2	[96]
Diameter [m]	2.6	2.6	2.6	2.6	[96]
Propulsion System	P.0 (x1)	P.0 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	NH_4ClO_4	NH_4ClO_4	UDMH	[96]
Oxidizer	Al	Al	Al	N_2O_4	[96]
Propellant mass [tonnes]	66.3	15.0	2.5	0.1	[96]
Structure mass [tonnes]	8.7	2.0	0.8	0.155	[96] [98]
Sea level thrust [kN]	2271.0	372.0	99.8	0.4	[96]
Vacuum thrust [kN]	Thrust profile (Solid)	Thrust profile (Solid)	x	0.4	[96] & [99]
SRM nozzle exit area [m^2]	3.14	3.14	x	N/A	[99]
SRM specific impulse [s]	283.0	299.9	x	N/A	[99]
Beginning time T+[s]	0.0	165.0	402.0	x	[97]
Burn time [s]	109.0	128.0	89.0	x	[97]
Jettison time T+[s]	161.0	398.0	807.0	x	[97]

Epsilon-Enhanced

Input [Unit]	Value			Reference
Fairing diameter [m]	2.6			[96]
Fairing length [m]	8.8			[96]
Fairing nose diameter [m]	0.204			[96]
Fairing nose length [m]	2.358			[96]
Fairing mass [kg]	1000.0			[96]
Fairing jettison time [s]	150.0			[97]
Latitude of the launch [°]	31.25			[96]
Payload mass [kg]	365			[31] [32] [33]
Stages	Stage 0	Stage 1	Stage 2	
Length [m]	11.7	4.3	2.3	[96]
Diameter [m]	2.6	2.6	2.6	[96]
Propulsion System	P.0 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	NH_4ClO_4	NH_4ClO_4	[96]
Oxidizer	Al	Al	Al	[96]
Propellant mass [tonnes]	66.3	15.0	2.5	[96]
Structure mass [tonnes]	8.7	2.0	0.8	[96]
Sea level thrust [kN]	2271.0	372.0	99.8	[96]
Vacuum thrust [kN]	Thrust profile (Solid)	Thrust profile (Solid)	x	[96] & [99]
SRM nozzle exit area [m^2]	3.14	3.14	x	[99]
SRM specific impulse [s]	283.0	299.9	x	[99]
Beginning time T+[s]	0.0	165.0	402.0	[97]
Burn time [s]	109.0	128.0	89.0	[97]
Jettison time T+[s]	161.0	398.0	807.0	[97]

Epsilon-Enhanced(CLPS)

Input [Unit]	Value				Reference
Fairing diameter [m]	2.6				[96]
Fairing length [m]	8.8				[96]
Fairing nose diameter [m]	0.204				[96]
Fairing nose length [m]	2.358				[96]
Fairing mass [kg]	1000.0				[96]
Fairing jettison time [s]	150.0				[97]
Latitude of the launch [°]	31.25				[96]
Payload mass [kg]	570				[31] [32] [33]
Stages	Stage 0	Stage 1	Stage 2	Stage 3	
Length [m]	11.7	4.3	2.3	1.2	[96]
Diameter [m]	2.6	2.6	2.6	2.6	[96]
Propulsion System	P.0 (x1)	P.0 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	NH_4ClO_4	NH_4ClO_4	UDMH	[96]
Oxidizer	Al	Al	Al	N_2O_4	[96]
Propellant mass [tonnes]	66.3	15.0	2.5	0.1	[96]
Structure mass [tonnes]	8.7	2.0	0.8	0.155	[96] [98]
Sea level thrust [kN]	2271.0	372.0	99.8	0.4	[96]
Vacuum thrust [kN]	Thrust profile (Solid)	Thrust profile (Solid)	x	0.4	[96] & [99]
SRM nozzle exit area [m^2]	3.14	3.14	x	N/A	[99]
SRM specific impulse [s]	283.0	299.9	x	N/A	[99]
Beginning time T+[s]	0.0	165.0	402.0	x	[97]
Burn time [s]	109.0	128.0	89.0	x	[97]
Jettison time T+[s]	161.0	398.0	807.0	x	[97]

Falcon-1

Input [Unit]	Value		Reference
Fairing diameter [m]	1.52		[100]
Fairing length [m]	3.5		[100]
Fairing nose diameter [m]	0.35		[100]
Fairing nose length [m]	2.0		[100]
Fairing mass [kg]	145.0		[100]
Fairing jettison time [s]	197.0		[100]
Latitude of the launch [°]	9.05		[100]
Payload mass [kg]	200		[31] [32] [33]
Stages	Stage 0	Stage 1	
Length [m]	15.0	2.7	[101]
Diameter [m]	1.54	1.54	[100]
Propulsion System	P.0 (x1)	P.0 (x1)	
Fuel	RP-1	RP-1	[100]
Oxidizer	O_2	O_2	[100]
Propellant mass [tonnes]	21.491	4.037	[100]
Structure mass [tonnes]	1.361	0.544	[100]
Sea level thrust [kN]	347.0	30.7	[100]
Vacuum thrust [kN]	409.106	30.7	[100] & [101]
SRM nozzle exit area [m^2]	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	
Beginning time T+[s]	0.0	174.0	[100]
Burn time [s]	170.0	368.0	[100]
Jettison time T+[s]	172.0	560.0	[100]

Falcon-9 v1.0

Input [Unit]	Value		Reference
Fairing diameter [m]	5.2		[102]
Fairing length [m]	13.9		[102]
Fairing nose diameter [m]	1.3		[103]
Fairing nose length [m]	5.9		[103]
Fairing mass [kg]	2000.0		[102]
Fairing jettison time [s]	199.0		[103]
Latitude of the launch [°]	28.56		[102]
Payload mass [kg]	Between 4225 and 6668		[31] [32] [33]
Stages	Stage 0	Stage 1	
Length [m]	29.0	10.5	[103]
Diameter [m]	3.66	3.66	[103]
Propulsion System	P.0 (x1)	P.0 (x1)	
Fuel	RP-1	RP-1	[102]
Oxidizer	O_2	O_2	[102]
Propellant mass [tonnes]	254.2	54.65	[103]
Structure mass [tonnes]	15.3	2.15	[103]
Sea level thrust [kN]	3803.3	520.4	[103]
Vacuum thrust [kN]	4344.0	520.4	[102] & [103]
SRM nozzle exit area [m^2]	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	
Beginning time T+[s]	0.0	176.0	[103]
Burn time [s]	175.0	346.0	[103]
Jettison time T+[s]	175.0	522.0	[103]

Falcon-9 v1.1

Input [Unit]	Value		Reference
Fairing diameter [m]	5.2		[13]
Fairing length [m]	13.0		[13]
Fairing nose diameter [m]	1.45		[13]
Fairing nose length [m]	4.8		[13]
Fairing mass [kg]	1900.0		[104]
Fairing jettison time [s]	199.0		[103]
Latitude of the launch [°]	28.56 or 34.73		[102]
Payload mass [kg]	Between 553 and 7930		[31] [32] [33]
Stages	Stage 0	Stage 1	
Length [m]	41.5	14.0	[103]
Diameter [m]	3.66	3.66	[103]
Propulsion System	P.0 (x1)	P.0 (x1)	
Fuel	RP-1	RP-1	[102]
Oxidizer	O_2	O_2	[102]
Propellant mass [tonnes]	393.96	88.95	[103]
Structure mass [tonnes]	19.74	3.1	[103]
Sea level thrust [kN]	5889.1	805.1	[103]
Vacuum thrust [kN]	6672.0	805.1	[102] & [103]
SRM nozzle exit area [m^2]	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	
Beginning time T+[s]	0.0	185.0	[103]
Burn time [s]	185.0	376.0	[103]
Jettison time T+[s]	185.0	561.0	[103]

Falcon-9 v1.1(ex)

Input [Unit]	Value		Reference
Fairing diameter [m]	5.2		[13]
Fairing length [m]	13.0		[13]
Fairing nose diameter [m]	1.45		[13]
Fairing nose length [m]	4.8		[13]
Fairing mass [kg]	1900.0		[104]
Fairing jettison time [s]	199.0		[103]
Latitude of the launch [°]	34.73 or 28.56		[102]
Payload mass [kg]	Between 600 and 7829		[31] [32] [33]
Stages	Stage 0	Stage 1	
Length [m]	41.5	14.0	[103]
Diameter [m]	3.66	3.66	[103]
Propulsion System	P.0 (x1)	P.0 (x1)	
Fuel	RP-1	RP-1	[102]
Oxidizer	O_2	O_2	[102]
Propellant mass [tonnes]	393.96	88.95	[103]
Structure mass [tonnes]	19.74	3.1	[103]
Sea level thrust [kN]	5889.1	805.1	[103]
Vacuum thrust [kN]	6672.0	805.1	[102] & [103]
SRM nozzle exit area [m^2]	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	
Beginning time T+[s]	0.0	185.0	[103]
Burn time [s]	185.0	376.0	[103]
Jettison time T+[s]	185.0	561.0	[103]

« Ex » stands for expandable : the first stage that is flying is reused from another flight.

Falcon-9 v1.2

Input [Unit]	Value		Reference
Fairing diameter [m]	3.7		[13]
Fairing length [m]	13.9		[13]
Fairing nose diameter [m]	2.2		[13]
Fairing nose length [m]	5.3		[13]
Fairing mass [kg]	5613.0		[104]
Fairing jettison time [s]	1000.0		[105] (CRS-8) [106] (Formosat)
Latitude of the launch [°]	28.56 or 34.73 or 28.52		[13]
Payload mass [kg]	Between 350 and 8749		[31] [32] [33]
Stages	Stage 0	Stage 1	
Length [m]	47.7	8.15	[103]
Diameter [m]	3.66	3.66	[13]
Propulsion System	P.0 (x1)	P.0 (x1)	
Fuel	RP-1	RP-1	[13]
Oxidizer	O_2	O_2	[13]
Propellant mass [tonnes]	386.34	105.35	[104] & [107]
Structure mass [tonnes]	46.86	6.15	[104] & [107]
Sea level thrust [kN]	6806.0	934.0	[102]
Vacuum thrust [kN]	7426.0	934.0	[102]
SRM nozzle exit area [m^2]	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	
Beginning time T+[s]	0.0	165.0	[105] (CRS-8) [106] (Formosat)
Burn time [s]	154.0	431.0	[105] (CRS-8) [106] (Formosat)
Jettison time T+[s]	155.0	597.0	[105] (CRS-8) [106] (Formosat)

The Falcon 9 v1.2 version has been using three different variants in the years of interest: Block 3 (Full Thrust FT), Block 4 and Block 5. Block 5 performances have since increased, but we do not take it into account here except for the crewed Dragon mission of 2020.
Block 4 is an intermediate version between Block 3 and 5. At first, only the second stage was modified to Block 4 standards, flying on top of a "Block 3" first stage. The only important difference with Block 3 seems to be an increase in engine thrust, so that is what we assessed. [108]

Falcon-9 v1.2(ex)

Input [Unit]	Value		Reference
Fairing diameter [m]	3.7		[13]
Fairing length [m]	13.9		[13]
Fairing nose diameter [m]	2.2		[13]
Fairing nose length [m]	5.3		[13]
Fairing mass [kg]	5613.0		[104]
Fairing jettison time [s]	1000.0		[105] (CRS-8) [106] (Formosat)
Latitude of the launch [°]	28.52 or 34.73 or 28.56		[13]
Payload mass [kg]	Between 2141 and 8600		[31] [32] [33]
Stages	Stage 0	Stage 1	
Length [m]	47.7	8.15	[103]
Diameter [m]	3.66	3.66	[13]
Propulsion System	P.0 (x1)	P.0 (x1)	
Fuel	RP-1	RP-1	[13]
Oxidizer	O_2	O_2	[13]
Propellant mass [tonnes]	386.34	105.35	[104] & [107]
Structure mass [tonnes]	46.86	6.15	[104] & [107]
Sea level thrust [kN]	6806.0	934.0	[102]
Vacuum thrust [kN]	7426.0	934.0	[102]
SRM nozzle exit area [m^2]	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	
Beginning time T+[s]	0.0	165.0	[105] (CRS-8) [106] (Formosat)
Burn time [s]	154.0	431.0	[105] (CRS-8) [106] (Formosat)
Jettison time T+[s]	155.0	597.0	[105] (CRS-8) [106] (Formosat)

Falcon-9 v1.2 Block 5

Input [Unit]	Value		Reference
Fairing diameter [m]	3.7		[13]
Fairing length [m]	13.9		[13]
Fairing nose diameter [m]	2.2		[13]
Fairing nose length [m]	5.3		[13]
Fairing mass [kg]	5613.0		[104]
Fairing jettison time [s]	1000.0		[105]
Latitude of the launch [°]	28.56		[13]
Payload mass [kg]	12500		[31] [32] [33]
Stages	Stage 0	Stage 1	
Length [m]	47.7	8.15	[103]
Diameter [m]	3.66	3.66	[13]
Propulsion System	P.0 (x1)	P.0 (x1)	
Fuel	RP-1	RP-1	[13]
Oxidizer	O_2	O_2	[13]
Propellant mass [tonnes]	395.0	105.35	[102]
Structure mass [tonnes]	46.86	6.46	[102]
Sea level thrust [kN]	7606.0	980.0	[102]
Vacuum thrust [kN]	8296.0	980.0	[102]
SRM nozzle exit area [m^2]	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	
Beginning time T+[s]	0.0	165.0	[105]
Burn time [s]	154.0	431.0	[105]
Jettison time T+[s]	155.0	597.0	[105]

Falcon-Heavy

Input [Unit]	Value			Reference
Fairing diameter [m]	5.2			[13]
Fairing length [m]	13.2			[13]
Fairing nose diameter [m]	1.45			[13]
Fairing nose length [m]	4.8			[13]
Fairing mass [kg]	1900.0			[104]
Fairing jettison time [s]	229.0			[109]
Latitude of the launch [°]	28.52			[13]
Payload mass [kg]	1300			[31] [32] [33]
Stages	Stage 0		Stage 1	
Length [m]	47.7		8.15	[103]
Diameter [m]	3.66		3.66	[13]
Propulsion System	P.0 (x2)	P.1 (x1)	P.0 (x1)	
Fuel	RP-1	RP-1	RP-1	[13]
Oxidizer	O_2	O_2	O_2	[13]
Propellant mass [tonnes]	395.0	395.0	105.35	[102]
Structure mass [tonnes]	46.86	46.86	6.46	[102]
Sea level thrust [kN]	7606.0	7606.0	980.0	[102]
Vacuum thrust [kN]	8296.0	8296.0	980.0	[102]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	
Beginning time T+[s]	0.0	0.0	187.0	[109]
Burn time [s]	149.0	184.0	324.0	[109]
Jettison time T+[s]	153.0	187.0	1732.0	[109]

GSLV Mk.2

Input [Unit]	Value				Reference
Fairing diameter [m]	3.4				[28]
Fairing length [m]	7.84				[28]
Fairing nose diameter [m]	1.7				[28]
Fairing nose length [m]	3.6				[28]
Fairing mass [kg]	1250.0				[28]
Fairing jettison time [s]	226.0				[110]
Latitude of the launch [°]	13.72				[111]
Payload mass [kg]	Between 1982 and 2250				[31] [32] [33]
Stages	Stage 0		Stage 1	Stage 2	
Length [m]	21.25		11.57	8.47	[110]
Diameter [m]	3.4		3.4	3.4	[110]
Propulsion System	P.0 (x4)	P.1 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	UDMH	NH_4ClO_4	UDMH	H_2	[110]
Oxidizer	N_2O_4	Al	N_2O_4	O_2	[110]
Propellant mass [tonnes]	42.7	138.11	39.48	12.84	[110]
Structure mass [tonnes]	5.0	23.0	5.1	2.5	[110]
Sea level thrust [kN]	x	x	x	x	
Vacuum thrust [kN]	759.4	Thrust profile (Solid)	846.8	73.55	[110] & [112]
SRM nozzle exit area [m^2]	N/A	4.52	N/A	N/A	[113]
SRM specific impulse [s]	N/A	270.0	N/A	N/A	[113]
Beginning time T+[s]	0.0	0.0	150.0	287.0	[110]
Burn time [s]	149.0	150.0	132.0	765.0	[110]
Jettison time T+[s]	151.0	151.0	286.0	1067.0	[110]

GSLV Mk.3

Input [Unit]	Value			Reference
Fairing diameter [m]	5.0			[28]
Fairing length [m]	8.63			[28]
Fairing nose diameter [m]	1.87			[28]
Fairing nose length [m]	4.37			[28]
Fairing mass [kg]	1250.0			[28]
Fairing jettison time [s]	225.0			[114]
Latitude of the launch [°]	13.72			[111]
Payload mass [kg]	3136			[31] [32] [33]
Stages	Stage 0		Stage 1	
Length [m]	21.39		13.545	[114]
Diameter [m]	4.0		4.0	[114]
Propulsion System	P.0 (x2)	P.1 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	UDMH	H_2	[114]
Oxidizer	Al	N_2O_4	O_2	[114]
Propellant mass [tonnes]	205.0	116.0	28.0	[114]
Structure mass [tonnes]	31.0	9.0	5.0	[114]
Sea level thrust [kN]	x	x	x	
Vacuum thrust [kN]	Thrust profile (Solid)	1598.0	186.0	[115]& [112]
SRM nozzle exit area [m^2]	13.85	N/A	N/A	[113]
SRM specific impulse [s]	275.0	N/A	N/A	[113]
Beginning time T+[s]	0.0	114.0	322.0	[114]
Burn time [s]	139.0	203.0	643.0	[114]
Jettison time T+[s]	140.0	320.0	980.0	[114]

GSLV Mk.3 (HTVE)

Input [Unit]	Value			Reference
Fairing diameter [m]	5.0			[28]
Fairing length [m]	8.63			[28]
Fairing nose diameter [m]	1.87			[28]
Fairing nose length [m]	4.37			[28]
Fairing mass [kg]	1250.0			[28]
Fairing jettison time [s]	225.0			[114]
Latitude of the launch [°]	13.72			[111]
Payload mass [kg]	3400			[31] [32] [33]
Stages	Stage 0		Stage 1	
Length [m]	21.39		13.545	[114]
Diameter [m]	4.0		4.0	[114]
Propulsion System	P.0 (x2)	P.1 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	UDMH	H_2	[114]
Oxidizer	Al	N_2O_4	O_2	[114]
Propellant mass [tonnes]	205.0	116.0	28.0	[114]
Structure mass [tonnes]	31.0	9.0	5.0	[114]
Sea level thrust [kN]	4541.935	x	186.0	
Vacuum thrust [kN]	Thrust profile (Solid)	1693.98	186.0	From [115] & [116] P.0 : [112]
SRM nozzle exit area [m^2]	13.85	N/A	N/A	[113]
SRM specific impulse [s]	275.0	N/A	N/A	[113]
Beginning time T+[s]	0.0	114.0	322.0	[114]
Burn time [s]	139.0	203.0	643.0	[114]
Jettison time T+[s]	140.0	320.0	980.0	[114]

HTVE stands for " High Thrust Vikas Engine ", an upgrade of the Vikas engine.

H-2A-202

Input [Unit]	Value			Reference
Fairing diameter [m]	5.1			[117]
Fairing length [m]	7.414			[117]
Fairing nose diameter [m]	1.6			[117]
Fairing nose length [m]	5.0			[117]
Fairing mass [kg]	1500.0			[118]
Fairing jettison time [s]	245.0			[117]
Latitude of the launch [°]	30.4			[117]
Payload mass [kg]	Between 710 and 4100			[31] [32] [33]
Stages	Stage 0		Stage 1	
Length [m]	37.2		0.0	[118]
Diameter [m]	4.0		4.0	[117]
Propulsion System	P.0 (x2)	P.1 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	H_2	H_2	[118]
Oxidizer	Al	O_2	O_2	[118]
Propellant mass [tonnes]	65.0	100.0	16.6	[117]
Structure mass [tonnes]	10.5	14.0	3.4	[118]
Sea level thrust [kN]	2346.357	854.0	x	[119]
Vacuum thrust [kN]	Thrust profile (Solid)	1098.0	x	[117] (Estimation for thrust profile of SRB-A)
SRM nozzle exit area [m^2]	3.8	N/A	N/A	[120]
SRM specific impulse [s]	283.0	N/A	N/A	[117]
Beginning time T+[s]	0.0	0.0	x	[117]
Burn time [s]	99.0	396.0	x	[117]
Jettison time T+[s]	110.0	404.0	x	[117]

H-2A-204

Input [Unit]	Value			Reference
Fairing diameter [m]	5.1			[117]
Fairing length [m]	7.414			[117]
Fairing nose diameter [m]	1.6			[117]
Fairing nose length [m]	5.0			[117]
Fairing mass [kg]	1500.0			[118]
Fairing jettison time [s]	245.0			[117]
Latitude of the launch [°]	30.4			[117]
Payload mass [kg]	Between 4700 and 4900			[31] [32] [33]
Stages	Stage 0		Stage 1	
Length [m]	37.2		0.0	[118]
Diameter [m]	4.0		4.0	[117]
Propulsion System	P.0 (x4)	P.1 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	H_2	H_2	[118]
Oxidizer	Al	O_2	O_2	[118]
Propellant mass [tonnes]	66.0	100.0	16.6	[117]
Structure mass [tonnes]	10.5	14.0	3.4	[118]
Sea level thrust [kN]	2059.75	854.0	x	[119]
Vacuum thrust [kN]	Thrust profile (Solid)	1098.0	x	[117] & [99]
SRM nozzle exit area [m^2]	3.14	N/A	N/A	Estimated from [121]
SRM specific impulse [s]	283.0	N/A	N/A	[117]
Beginning time T+[s]	0.0	0.0	x	[117]
Burn time [s]	117.0	396.0	x	[117]
Jettison time T+[s]	128.0	404.0	x	[117]

H-2B-304

Input [Unit]	Value			Reference
Fairing diameter [m]	4.07			[117]
Fairing length [m]	7.414			[117]
Fairing nose diameter [m]	1.233			[117]
Fairing nose length [m]	4.69			[117]
Fairing mass [kg]	1500.0			[118]
Fairing jettison time [s]	245.0			[117]
Latitude of the launch [°]	30.4			[117]
Payload mass [kg]	Between 15000 and 16500			[31] [32] [33]
Stages	Stage 0		Stage 1	
Length [m]	38.2		11.0	[122]
Diameter [m]	5.2		5.2	[122]
Propulsion System	P.0 (x4)	P.1 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	H_2	H_2	[118]
Oxidizer	Al	O_2	O_2	[118]
Propellant mass [tonnes]	66.0	177.8	16.6	[122]
Structure mass [tonnes]	10.5	24.2	3.4	[123]
Sea level thrust [kN]	2059.75	1677.0	x	[119]
Vacuum thrust [kN]	Thrust profile (Solid)	2196.0	x	[122] & [99]
SRM nozzle exit area [m^2]	3.14	N/A	N/A	Estimated from [121]
SRM specific impulse [s]	283.0	N/A	N/A	[117]
Beginning time T+[s]	0.0	0.0	x	[117]
Burn time [s]	117.0	440.0	x	[117]
Jettison time T+[s]	128.0	440.0	x	[117]

Kosmos-3M

Input [Unit]	Value				Reference
Fairing diameter [m]	2.4				[28]
Fairing length [m]	5.72				[28]
Fairing nose diameter [m]	0.209				[28]
Fairing nose length [m]	2.704				[28]
Fairing mass [kg]	348.0				[28]
Fairing jettison time [s]	147.0				[28]
Latitude of the launch [°]	62.92 or 28.25				[28]
Payload mass [kg]	Between 820 and 980				[31] [32] [33]
Stages	Stage 0	Stage 1			
Length [m]	22.4	6.0			[28]
Diameter [m]	2.4	2.4			[28]
Propulsion System	P.0 (x1)	P.0 (x1)	P.1 (x1)	P.2 (x1)	
Fuel	UDMH	UDMH	UDMH	UDMH	[28]
Oxidizer	N_2O_4	N_2O_4	N_2O_4	N_2O_4	[28]
Propellant mass [tonnes]	81.9	18.7	0.0	0.0	[28]
Structure mass [tonnes]	5.3	1.435	0.0	0.0	[28]
Sea level thrust [kN]	1485.0	157.0	6.0	6.0	[28]
Vacuum thrust [kN]	1745.0	157.0	6.0	6.0	[28]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	N/A	
Beginning time T+[s]	0.0	135.0	131.0	1728.0	[28]
Burn time [s]	130.0	342.0	357.0	15.0	[28]
Jettison time T+[s]	132.0	1763.0	1763.0	1763.0	[28]

Minotaur-1

Input [Unit]	Value				Reference
Fairing diameter [m]	1.27				[28]
Fairing length [m]	4.42				[28]
Fairing nose diameter [m]	0.787				[28]
Fairing nose length [m]	3.22				[28]
Fairing mass [kg]	194.0				[28]
Fairing jettison time [s]	123.0				[124]
Latitude of the launch [°]	37.94 or 34.73				[28]
Payload mass [kg]	Between 335 and 431				[31] [32] [33]
Stages	Stage 0	Stage 1	Stage 2	Stage 3	
Length [m]	7.49	4.12	3.58	1.34	[28]
Diameter [m]	1.3	1.3	1.3	1.3	[28]
Propulsion System	P.0 (x1)	P.0 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	NH_4ClO_4	NH_4ClO_4	NH_4ClO_4	[28]
Oxidizer	Al	Al	Al	Al	[28]
Propellant mass [tonnes]	10.785	6.237	3.915	0.771	[28]
Structure mass [tonnes]	2.248	0.691	0.416	0.126	[28]
Sea level thrust [kN]	792.0	268.0	154.0	32.0	From [125]
Vacuum thrust [kN]	x	x	Thrust profile (Solid)	Thrust profile (Solid)	[125]
SRM nozzle exit area [m^2]	x	x	0.58	0.22	[125]
SRM specific impulse [s]	x	x	290.65	286.97	[125]
Beginning time T+[s]	0.0	62.0	130.0	635.0	[124]
Burn time [s]	62.0	66.0	74.0	128.0	[124]
Jettison time T+[s]	62.0	128.0	623.0	763.0	[124]

Minotaur-4

Input [Unit]	Value				Reference
Fairing diameter [m]	2.34				[126]
Fairing length [m]	6.38				[126]
Fairing nose diameter [m]	0.488				[126] (image)
Fairing nose length [m]	2.984				[126] (image)
Fairing mass [kg]	400.0				[126]
Fairing jettison time [s]	156.0				
Latitude of the launch [°]	34.73				[28]
Payload mass [kg]	1031				[31] [32] [33]
Stages	Stage 0	Stage 1	Stage 2	Stage 3	
Length [m]	8.39	7.88	2.33	0.97	[126]
Diameter [m]	2.34	2.34	2.34	2.34	[126]
Propulsion System	P.0 (x1)	P.0 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	NH_4ClO_4	NH_4ClO_4	NH_4ClO_4	[126]
Oxidizer	Al	Al	Al	Al	[126]
Propellant mass [tonnes]	45.37	24.49	7.07	0.771	[126]
Structure mass [tonnes]	3.62	3.18	0.64	0.126	[126]
Sea level thrust [kN]	2049.454	2224.0	1223.0	32.0	[126]
Vacuum thrust [kN]	x	x	x	Thrust profile (Solid)	[125]
SRM nozzle exit area [m^2]	x	x	x	0.22	[125]
SRM specific impulse [s]	x	x	x	286.97	[125]
Beginning time T+[s]	0.0	58.0	133.0	646.0	[126]
Burn time [s]	58.0	60.0	74.0	69.0	[126]
Jettison time T+[s]	58.0	133.0	695.0	715.0	[126]

Minotaur-4(HAPS)

Input [Unit]	Value					Reference
Fairing diameter [m]	2.34					[126]
Fairing length [m]	6.38					[126]
Fairing nose diameter [m]	0.488					[126] (image)
Fairing nose length [m]	2.984					[126] (image)
Fairing mass [kg]	400.0					[126]
Fairing jettison time [s]	130.0					[127]
Latitude of the launch [°]	57.44					[28]
Payload mass [kg]	550					[31] [32] [33]
Stages	Stage 0	Stage 1	Stage 2	Stage 3	Stage 4	
Length [m]	8.39	7.88	2.33	0.97	0.71	[126]
Diameter [m]	2.34	2.34	2.34	2.34	2.34	[126]
Propulsion System	P.0 (x1)	P.0 (x1)	P.0 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	NH_4ClO_4	NH_4ClO_4	NH_4ClO_4	NH_4ClO_4	[126]
Oxidizer	Al	Al	Al	Al	Al	[126]
Propellant mass [tonnes]	45.37	24.49	7.07	0.771	0.059	[126]
Structure mass [tonnes]	3.62	3.18	0.64	0.126	0.022	[126]
Sea level thrust [kN]	2049.454	2224.0	1223.0	32.0	x	[126]
Vacuum thrust [kN]	x	x	x	Thrust profile (Solid)	x	[125]
SRM nozzle exit area [m^2]	x	x	x	0.22	x	[125]
SRM specific impulse [s]	x	x	x	286.97	x	[125]
Beginning time T+[s]	0.0	58.0	133.0	646.0	x	[127]
Burn time [s]	58.0	60.0	74.0	69.0	x	[127]
Jettison time T+[s]	58.0	133.0	695.0	715.0	x	[127]

Minotaur-4(Orion 38)

Input [Unit]	Value					Reference
Fairing diameter [m]	2.34					[126]
Fairing length [m]	6.38					[126]
Fairing nose diameter [m]	0.488					[126] (image)
Fairing nose length [m]	2.984					[126] (image)
Fairing mass [kg]	400.0					[126]
Fairing jettison time [s]	130.0					[127]
Latitude of the launch [°]	28.56					[28]
Payload mass [kg]	140					[31] [32] [33]
Stages	Stage 0	Stage 1	Stage 2	Stage 3	Stage 4	
Length [m]	8.39	7.88	2.33	0.97	1.34	[126]
Diameter [m]	2.34	2.34	2.34	2.34	1.2	[126]
Propulsion System	P.0 (x1)	P.0 (x1)	P.0 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	NH_4ClO_4	NH_4ClO_4	NH_4ClO_4	NH_4ClO_4	[126]
Oxidizer	Al	Al	Al	Al	Al	[126]
Propellant mass [tonnes]	45.37	24.49	7.07	0.771	0.77	[126]
Structure mass [tonnes]	3.62	3.18	0.64	0.126	0.093	[126]
Sea level thrust [kN]	2049.454	2224.0	1223.0	32.0	32.23	[126]
Vacuum thrust [kN]	x	x	x	Thrust profile (Solid)	Thrust profile (Solid)	[125]
SRM nozzle exit area [m^2]	x	x	x	0.22	0.22	[125]
SRM specific impulse [s]	x	x	x	286.97	286.97	[125]
Beginning time T+[s]	0.0	58.0	133.0	646.0	x	[127]
Burn time [s]	58.0	60.0	74.0	69.0	x	[127]
Jettison time T+[s]	58.0	133.0	695.0	715.0	x	[127]

Minotaur-4+

Input [Unit]	Value				Reference
Fairing diameter [m]	2.34				[126]
Fairing length [m]	6.38				[126]
Fairing nose diameter [m]	0.488				[126] (image)
Fairing nose length [m]	2.984				[126] (image)
Fairing mass [kg]	400.0				[126]
Fairing jettison time [s]	130.0				[127]
Latitude of the launch [°]	57.44				[28]
Payload mass [kg]	450				[31] [32] [33]
Stages	Stage 0	Stage 1	Stage 2	Stage 3	
Length [m]	8.39	7.88	2.33	1.245	[126]
Diameter [m]	2.34	2.34	2.34	2.34	[126]
Propulsion System	P.0 (x1)	P.0 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	NH_4ClO_4	NH_4ClO_4	NH_4ClO_4	[126]
Oxidizer	Al	Al	Al	Al	[126]
Propellant mass [tonnes]	45.37	24.49	7.07	2.014	[126]
Structure mass [tonnes]	3.62	3.18	0.64	0.157	[126]
Sea level thrust [kN]	2049.454	2224.0	1223.0	68.63	[126]
Vacuum thrust [kN]	x	x	x	Thrust profile (Solid)	[125]
SRM nozzle exit area [m^2]	x	x	x	0.32	[125]
SRM specific impulse [s]	x	x	x	292.1	[125]
Beginning time T+[s]	0.0	58.0	133.0	1379.0	[127] [126]
Burn time [s]	58.0	60.0	74.0	82.0	[127] [126]
Jettison time T+[s]	58.0	133.0	695.0	1661.0	[127] [126]

Minotaur-5

Input [Unit]	Value					Reference
Fairing diameter [m]	2.34					[126]
Fairing length [m]	6.38					[126]
Fairing nose diameter [m]	0.488					[126] (image)
Fairing nose length [m]	2.984					[126] (image)
Fairing mass [kg]	400.0					[126]
Fairing jettison time [s]	130.0					[127]
Latitude of the launch [°]	37.94					[28]
Payload mass [kg]	383					[31] [32] [33]
Stages	Stage 0	Stage 1	Stage 2	Stage 3	Stage 4	
Length [m]	8.39	7.88	2.33	1.245	0.0	[126]
Diameter [m]	2.34	2.34	2.34	2.34	2.34	[126]
Propulsion System	P.0 (x1)	P.0 (x1)	P.0 (x1)	P.0 (x1)	P.0 (x1)	[126]
Fuel	NH_4ClO_4	NH_4ClO_4	NH_4ClO_4	NH_4ClO_4	NH_4ClO_4	[126]
Oxidizer	Al	Al	Al	Al	Al	[126]
Propellant mass [tonnes]	45.37	24.49	7.07	2.014	1.068	[126]
Structure mass [tonnes]	3.62	3.18	0.64	0.157	0.082	[126]
Sea level thrust [kN]	2049.454	2224.0	1223.0	68.63	48.13	[126]
Vacuum thrust [kN]	x	x	x	Thrust profile (Solid)	Thrust profile (Solid)	[125]
SRM nozzle exit area [m^2]	x	x	x	0.32	0.32	[125]
SRM specific impulse [s]	x	x	x	292.1	289.8	[125]
Beginning time T+[s]	0.0	58.0	133.0	1379.0	1661.0	[126] [127]
Burn time [s]	58.0	60.0	74.0	82.0	63.0	[126] [127]
Jettison time T+[s]	58.0	133.0	695.0	1661.0	1724.0	[126] [127]

Minotaur-C-XL-3110

Input [Unit]	Value				Reference
Fairing diameter [m]	1.6				[28]
Fairing length [m]	5.5				[28]
Fairing nose diameter [m]	0.762				[28]
Fairing nose length [m]	1.61				[28]
Fairing mass [kg]	360.0				[28]
Fairing jettison time [s]	178.0				[128]
Latitude of the launch [°]	34.73				[28]
Payload mass [kg]	Between 407 and 538				[31] [32] [33]
Stages	Stage 0	Stage 1	Stage 2	Stage 3	
Length [m]	12.0	9.88	3.09	1.34	[39] & [129]
Diameter [m]	1.2	1.2	1.2	1.2	[28]
Propulsion System	P.0 (x1)	P.0 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	NH_4ClO_4	NH_4ClO_4	NH_4ClO_4	[39]
Oxidizer	Al	Al	Al	Al	[39]
Propellant mass [tonnes]	48.949	15.034	3.915	0.77	[39]
Structure mass [tonnes]	4.126	1.121	0.366	0.093	[39]
Sea level thrust [kN]	1685.79	614.8	160.55	32.23	[39]
Vacuum thrust [kN]	Thrust profile (Solid)	Thrust profile (Solid)	Thrust profile (Solid)	Thrust profile (Solid)	[39]
SRM nozzle exit area [m^2]	1.81	1.15	0.58	0.22	[39]
SRM specific impulse [s]	280.0	284.61	290.65	286.97	[39]
Beginning time T+[s]	0.0	85.0	172.0	598.0	[128]
Burn time [s]	85.0	85.0	79.0	72.0	[128]
Jettison time T+[s]	85.0	170.0	315.0	670.0	[128]

Minotaur-C-XL-3210

Input [Unit]	Value				Reference
Fairing diameter [m]	2.34				[28]
Fairing length [m]	7.0				[28]
Fairing nose diameter [m]	0.441				[28]
Fairing nose length [m]	2.9				[28]
Fairing mass [kg]	400.0				[28]
Fairing jettison time [s]	178.0				[128]
Latitude of the launch [°]	34.73				[28]
Payload mass [kg]	676				[31] [32] [33]
Stages	Stage 0	Stage 1	Stage 2	Stage 3	
Length [m]	12.0	9.88	3.09	1.34	[39] & [129]
Diameter [m]	1.2	1.2	1.2	1.2	[28]
Propulsion System	P.0 (x1)	P.0 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	NH_4ClO_4	NH_4ClO_4	NH_4ClO_4	[39]
Oxidizer	Al	Al	Al	Al	[39]
Propellant mass [tonnes]	48.949	15.034	3.915	0.77	[39]
Structure mass [tonnes]	4.126	1.121	0.366	0.093	[39]
Sea level thrust [kN]	1685.79	614.8	160.55	32.23	[39]
Vacuum thrust [kN]	Thrust profile (Solid)	Thrust profile (Solid)	Thrust profile (Solid)	Thrust profile (Solid)	[39]
SRM nozzle exit area [m^2]	1.81	1.15	0.58	0.22	[39]
SRM specific impulse [s]	280.0	284.61	290.65	286.97	[39]
Beginning time T+[s]	0.0	85.0	172.0	598.0	[128]
Burn time [s]	85.0	85.0	79.0	72.0	[128]
Jettison time T+[s]	85.0	170.0	315.0	670.0	[128]

Molniya-M (Blok-2BL)

Input [Unit]	Value				Reference
Fairing diameter [m]	2.7				[28]
Fairing length [m]	7.8				[28]
Fairing nose diameter [m]	0.24				[28]
Fairing nose length [m]	3.01				[28]
Fairing mass [kg]	1000.0				[28]
Fairing jettison time [s]	187.0				[130]
Latitude of the launch [°]	62.92				[130]
Payload mass [kg]	2400				[31] [32] [33]
Stages	Stage 0		Stage 1	Stage 2	
Length [m]	27.1		6.7	2.6	[130]
Diameter [m]	2.6		2.6	2.6	[130]
Propulsion System	P.0 (x1)	P.1 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	RP-1	RP-1	RP-1	RP-1	[130]
Oxidizer	O_2	O_2	O_2	O_2	[130]
Propellant mass [tonnes]	158.4	92.95	22.59	4.35	[130]
Structure mass [tonnes]	15.2	6.55	2.71	1.05	[130]
Sea level thrust [kN]	3255.808	745.305	298.024	66.68	[131]
Vacuum thrust [kN]	3966.984	941.438	298.024	66.68	[131]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	N/A	
Beginning time T+[s]	0.0	0.0	280.0	520.0	[130]
Burn time [s]	118.0	280.0	240.0	192.0	[130]
Jettison time T+[s]	118.0	280.0	520.0	712.0	[130]

Naro-1

Input [Unit]	Value		Reference
Fairing diameter [m]	2.0		[132]
Fairing length [m]	5.0		[132]
Fairing nose diameter [m]	0.35		[133]
Fairing nose length [m]	3.43		[133]
Fairing mass [kg]	200.0		[132]
Fairing jettison time [s]	216.0		[134]
Latitude of the launch [°]	34.43		[135]
Payload mass [kg]	Between 99 and 100		[31] [32] [33]
Stages	Stage 0	Stage 1	
Length [m]	25.8	2.7	[132]
Diameter [m]	2.9	2.9	[132]
Propulsion System	P.0 (x1)	P.0 (x1)	
Fuel	RP-1	NH_4ClO_4	[132]
Oxidizer	O_2	Al	[132]
Propellant mass [tonnes]	127.5	1.3	[132]
Structure mass [tonnes]	10.5	0.2	[132]
Sea level thrust [kN]	1667.0	78.448	[132]
Vacuum thrust [kN]	1741.5	x	[132]
SRM nozzle exit area [m^2]	N/A	x	
SRM specific impulse [s]	N/A	x	
Beginning time T+[s]	0.0	395.0	[132] & [134]
Burn time [s]	228.0	58.0	[132] & [134]
Jettison time T+[s]	228.0	540.0	[132] & [134]

Pegasus-XL

Input [Unit]	Value			Reference
Fairing diameter [m]	4.42			[28]
Fairing length [m]	4.42			[28]
Fairing nose diameter [m]	0.72			[28]
Fairing nose length [m]	1.36			[28]
Fairing mass [kg]	170.0			[28]
Fairing jettison time [s]	132.0			[136]
Latitude of the launch [°]	8.72 or 34.73 or 28.56			[137]
Payload mass [kg]	Between 181 and 350			[31] [32] [33]
Stages	Stage 0	Stage 1	Stage 2	
Length [m]	10.3	3.11	1.34	[137]
Diameter [m]	1.28	1.28	1.28	[137]
Propulsion System	P.0 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	NH_4ClO_4	NH_4ClO_4	[137]
Oxidizer	Al	Al	Al	[137]
Propellant mass [tonnes]	15.032	3.923	0.77	[137]
Structure mass [tonnes]	1.386	0.416	0.108	[137]
Sea level thrust [kN]	462.6	160.5	32.231	From [125]
Vacuum thrust [kN]	Thrust profile (Solid)	Thrust profile (Solid)	Thrust profile (Solid)	[125]
SRM nozzle exit area [m^2]	1.59	0.58	0.22	[125]
SRM specific impulse [s]	292.78	290.65	286.97	[125]
Beginning time T+[s]	5.0	94.0	408.0	[136]
Burn time [s]	77.0	74.0	68.0	[136]
Jettison time T+[s]	93.0	397.0	776.0	[136]

Proton-K(Blok-DM-2)

Input [Unit]	Value					Reference
Fairing diameter [m]	4.1					[28]
Fairing length [m]	6.28					[28]
Fairing nose diameter [m]	0.262					[28]
Fairing nose length [m]	4.417					[28]
Fairing mass [kg]	1500.0					[138]
Fairing jettison time [s]	344.0					[139]
Latitude of the launch [°]	45.6					[139]
Payload mass [kg]	Between 2320 and 2600					[31] [32] [33]
Stages	Stage 0	Stage 1	Stage 2		Stage 3	
Length [m]	21.2	17.0	6.9		6.28	[28]
Diameter [m]	4.1	4.1	4.1		4.1	[28]
Propulsion System	P.0 (x1)	P.0 (x1)	P.0 (x1)	P.1 (x1)	P.0 (x1)	
Fuel	UDMH	UDMH	UDMH	UDMH	RP-1	[139]
Oxidizer	N_2O_4	N_2O_4	N_2O_4	N_2O_4	O_2	[139]
Propellant mass [tonnes]	419.41	156.113	46.562	0.0	15.05	[139]
Structure mass [tonnes]	31.0	11.75	4.15	0.0	3.13	[139]
Sea level thrust [kN]	9500.0	2300.0	583.0	31.0	83.5	[139]
Vacuum thrust [kN]	10493.0	2300.0	583.0	31.0	83.5	[139]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	N/A	N/A	
Beginning time T+[s]	0.0	117.0	336.0	330.0	x	[139]
Burn time [s]	117.0	216.0	231.0	247.0	x	[139]
Jettison time T+[s]	121.0	333.0	582.0	582.0	x	[139]

Proton-M(Blok-DM-03)

Input [Unit]	Value					Reference
Fairing diameter [m]	4.1					[28]
Fairing length [m]	6.28					[28]
Fairing nose diameter [m]	0.262					[28]
Fairing nose length [m]	4.417					[28]
Fairing mass [kg]	1500.0					[138]
Fairing jettison time [s]	344.0					[140]
Latitude of the launch [°]	45.6					[140]
Payload mass [kg]	Between 2100 and 4245					[31] [32] [33]
Stages	Stage 0	Stage 1	Stage 2		Stage 3	
Length [m]	21.2	17.0	6.75		6.28	[141]
Diameter [m]	4.1	4.1	4.1		4.1	[141]
Propulsion System	P.0 (x1)	P.0 (x1)	P.0 (x1)	P.1 (x1)	P.0 (x1)	
Fuel	UDMH	UDMH	UDMH	UDMH	RP-1	[141]
Oxidizer	N_2O_4	N_2O_4	N_2O_4	N_2O_4	O_2	[141]
Propellant mass [tonnes]	428.3	157.3	43.854	2.708	18.7	[141] [142]
Structure mass [tonnes]	30.6	11.0	3.5	0.0	2.9	[141] [142]
Sea level thrust [kN]	10000.0	2400.0	583.0	31.0	x	[141]
Vacuum thrust [kN]	11000.0	2400.0	583.0	31.0	x	[141]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	N/A	N/A	
Beginning time T+[s]	0.0	124.0	338.0	338.0	x	[140]
Burn time [s]	119.0	203.0	239.0	251.0	x	[140]
Jettison time T+[s]	124.0	327.0	589.0	589.0	x	[140]

Proton-M(Blok-DM-2)

Input [Unit]	Value					Reference
Fairing diameter [m]	4.1					[28]
Fairing length [m]	6.28					[28]
Fairing nose diameter [m]	0.262					[28]
Fairing nose length [m]	4.417					[28]
Fairing mass [kg]	1500.0					[138]
Fairing jettison time [s]	344.0					[140]
Latitude of the launch [°]	45.6					[140]
Payload mass [kg]	Between 4245 and 4245					[31] [32] [33]
Stages	Stage 0	Stage 1	Stage 2		Stage 3	
Length [m]	21.2	17.0	6.75		6.28	[141]
Diameter [m]	4.1	4.1	4.1		4.1	[141]
Propulsion System	P.0 (x1)	P.0 (x1)	P.0 (x1)	P.1 (x1)	P.0 (x1)	
Fuel	UDMH	UDMH	UDMH	UDMH	RP-1	[141]
Oxidizer	N_2O_4	N_2O_4	N_2O_4	N_2O_4	O_2	[141]
Propellant mass [tonnes]	428.3	157.3	43.854	2.708	15.05	[141] & estimated from [139]
Structure mass [tonnes]	30.6	11.0	3.5	0.0	3.13	[141] & [139]
Sea level thrust [kN]	10000.0	2400.0	583.0	31.0	83.5	[141] & [139]
Vacuum thrust [kN]	11000.0	2400.0	583.0	31.0	83.5	[141] & [139]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	N/A	N/A	
Beginning time T+[s]	0.0	124.0	338.0	338.0	x	[140]
Burn time [s]	119.0	203.0	239.0	251.0	x	[140]
Jettison time T+[s]	124.0	327.0	589.0	589.0	x	[140]

Proton-M(Briz-M)

Input [Unit]	Value					Reference
Fairing diameter [m]	4.35					[28]
Fairing length [m]	15.2					[28]
Fairing nose diameter [m]	0.3					[28]
Fairing nose length [m]	6.1					[28]
Fairing mass [kg]	3000.0					[143]
Fairing jettison time [s]	349.0					[140]
Latitude of the launch [°]	45.6					[140]
Payload mass [kg]	Between 2056 and 6871					[31] [32] [33]
Stages	Stage 0	Stage 1	Stage 2		Stage 3	
Length [m]	21.2	17.0	6.75		2.65	[141]
Diameter [m]	4.1	4.1	4.1		4.1	[141]
Propulsion System	P.0 (x1)	P.0 (x1)	P.0 (x1)	P.1 (x1)	P.0 (x1)	
Fuel	UDMH	UDMH	UDMH	UDMH	UDMH	[141]
Oxidizer	N_2O_4	N_2O_4	N_2O_4	N_2O_4	N_2O_4	[141]
Propellant mass [tonnes]	428.3	157.3	43.854	2.708	19.8	[141]
Structure mass [tonnes]	30.6	11.0	3.5	0.0	2.5	[141]
Sea level thrust [kN]	10000.0	2400.0	583.0	31.0	x	[141]
Vacuum thrust [kN]	11000.0	2400.0	583.0	31.0	x	[141]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	N/A	N/A	
Beginning time T+[s]	0.0	124.0	338.0	338.0	x	[140]
Burn time [s]	119.0	203.0	239.0	251.0	x	[140]
Jettison time T+[s]	124.0	327.0	589.0	589.0	x	[140]

PSLV-CA

Input [Unit]	Value				Reference
Fairing diameter [m]	3.2				[28]
Fairing length [m]	8.3				[28]
Fairing nose diameter [m]	1.3				[28]
Fairing nose length [m]	2.9				[28]
Fairing mass [kg]	110.0				[28]
Fairing jettison time [s]	188.0				[144] [145]
Latitude of the launch [°]	13.72				[111]
Payload mass [kg]	Between 350 and 1047				[31] [32] [33]
Stages	Stage 0	Stage 1	Stage 2	Stage 3	
Length [m]	20.0	12.8	0.0	0.0	[144]
Diameter [m]	1.3	1.3	1.3	1.3	[144]
Propulsion System	P.0 (x1)	P.0 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	UH25	NH_4ClO_4	MMH	[144]
Oxidizer	Al	N_2O_4	Al	MON 3	[144]
Propellant mass [tonnes]	138.0	41.0	7.6	2.5	[144]
Structure mass [tonnes]	30.0	5.3	0.7	0.42	[146]
Sea level thrust [kN]	3823.0	804.0	109.274	14.6	[28]
Vacuum thrust [kN]	Thrust profile (Solid)	804.0	x	14.6	[28] [112] [113]
SRM nozzle exit area [m^2]	4.52	N/A	x	N/A	[113]
SRM specific impulse [s]	269.0	N/A	x	N/A	[113]
Beginning time T+[s]	0.0	112.0	264.0	x	[144] [145]
Burn time [s]	102.0	148.0	112.0	x	[144] [145]
Jettison time T+[s]	111.0	262.0	587.0	x	[144] [145]

PSLV-G

Input [Unit]	Value						Reference
Fairing diameter [m]	3.2						[28]
Fairing length [m]	8.3						[28]
Fairing nose diameter [m]	1.0						[28]
Fairing nose length [m]	2.9						[28]
Fairing mass [kg]	1100.0						[28]
Fairing jettison time [s]	161.0						[147]
Latitude of the launch [°]	13.72						[111]
Payload mass [kg]	Between 675 and 1404						[31] [32] [33]
Stages	Stage 0			Stage 1	Stage 2	Stage 3	
Length [m]	20.0			12.8	3.6	2.4	[147]
Diameter [m]	1.3			1.3	1.3	1.3	[147]
Propulsion System	P.0 (x4)	P.1 (x2)	P.2 (x1)	P.0 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	NH_4ClO_4	NH_4ClO_4	UH25	NH_4ClO_4	MMH	[147]
Oxidizer	Al	Al	Al	N_2O_4	Al	MON 3	[147]
Propellant mass [tonnes]	8.9	8.9	138.0	42.0	7.6	1.6	[147]
Structure mass [tonnes]	2.01	2.01	30.0	5.0	0.84	0.42	[147]
Sea level thrust [kN]	386.682	386.682	3823.0	804.0	109.274	14.6	[28] [112] [113]
Vacuum thrust [kN]	Thrust profile (Solid)	Thrust profile (Solid)	Thrust profile (Solid)	804.0	x	14.6	[28] [112] [113]
SRM nozzle exit area $[m^2]$	4.52	4.52	4.52	N/A	x	N/A	[113]
SRM specific impulse [s]	265.0	265.0	269.0	N/A	x	N/A	[113]
Beginning time T+[s]	0.0	25.0	0.0	113.0	264.0	x	[147]
Burn time [s]	68.0	68.0	102.0	151.0	112.0	x	[147]
Jettison time T+[s]	68.0	90.0	111.0	264.0	587.0	x	[147]

PSLV-XL

Input [Unit]	Value						Reference
Fairing diameter [m]	3.2						[28]
Fairing length [m]	8.3						[28]
Fairing nose diameter [m]	1.0						[28]
Fairing nose length [m]	2.9						[28]
Fairing mass [kg]	1100.0						[28]
Fairing jettison time [s]	170.0						[148] [149]
Latitude of the launch [°]	13.72						[111]
Payload mass [kg]	Between 613 and 1858						[31] [32] [33]
Stages	Stage 0			Stage 1	Stage 2	Stage 3	
Length [m]	20.0			12.8	0.0	0.0	[148]
Diameter [m]	1.3			1.3	1.3	1.3	[148]
Propulsion System	P.0 (x4)	P.1 (x2)	P.2 (x1)	P.0 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	NH_4ClO_4	NH_4ClO_4	UH25	NH_4ClO_4	MMH	[148]
Oxidizer	Al	Al	Al	N_2O_4	Al	MON 3	[148]
Propellant mass [tonnes]	12.2	12.2	138.2	41.3	7.6	2.5	[148]
Structure mass [tonnes]	2.7	2.7	30.0	5.3	0.84	0.42	[146]
Sea level thrust [kN]	4275.252	4275.252	3823.0	800.0	109.274	14.6	[28]
Vacuum thrust [kN]	Thrust profile (Solid)	Thrust profile (Solid)	Thrust profile (Solid)	800.0	x	14.6	[28] [112] [113]
SRM nozzle exit area [m^2]	0.77	0.77	4.52	N/A	x	N/A	[113]
SRM specific impulse [s]	265.0	265.0	270.0	N/A	x	N/A	[113]
Beginning time T+[s]	0.0	25.0	0.0	114.0	264.0	x	[148] [149]
Burn time [s]	54.0	54.0	105.0	147.0	112.0	x	[148] [149]
Jettison time T+[s]	70.0	92.0	112.0	262.0	587.0	x	[148] [149]

Rokot-KM

Input [Unit]	Value			Reference
Fairing diameter [m]	2.5			[150]
Fairing length [m]	6.7			[150]
Fairing nose diameter [m]	0.313			[150]
Fairing nose length [m]	3.024			[150]
Fairing mass [kg]	800.0			[150]
Fairing jettison time [s]	171.0			[150]
Latitude of the launch [°]	62.92			[150]
Payload mass [kg]	Between 560 and 1419			[31] [32] [33]
Stages	Stage 0	Stage 1	Stage 2	
Length [m]	17.2	3.9	3.0	[150]
Diameter [m]	2.5	2.5	2.5	[150]
Propulsion System	P.0 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	UDMH	UDMH	UDMH	[150]
Oxidizer	N_2O_4	N_2O_4	N_2O_4	[150]
Propellant mass [tonnes]	71.45	10.71	4.975	Assumed from [150]
Structure mass [tonnes]	5.7	1.485	1.32	[150] (Upper stage) [151] [152]
Sea level thrust [kN]	1870.0	240.0	20.0	[150]
Vacuum thrust [kN]	2070.0	240.0	20.0	[150]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	
Beginning time T+[s]	0.0	122.0	305.0	[150]
Burn time [s]	121.0	183.0	577.0	[150]
Jettison time T+[s]	122.0	305.0	882.0	[150]

Safir-1

Input [Unit]	Value		Reference
Fairing diameter [m]	1.35		[153] (image)
Fairing length [m]	2.0		[153]
Fairing nose diameter [m]	0.45		[153]
Fairing nose length [m]	0.93		[153]
Fairing mass [kg]	100.0		[154]
Fairing jettison time [s]	150.0		[154]
Latitude of the launch [°]	35.24		[155]
Payload mass [kg]	27		[31] [32] [33]
Stages	Stage 0	Stage 1	
Length [m]	17.0	3.0	[60]
Diameter [m]	1.35	1.35	[153]
Propulsion System	P.0 (x1)	P.0 (x1)	
Fuel	UDMH	UDMH	[60]
Oxidizer	AK	AK	[60]
Propellant mass [tonnes]	21.4	2.7	[60]
Structure mass [tonnes]	2.6	0.3	From total mass [60]
Sea level thrust [kN]	284.4	19.6	[154]
Vacuum thrust [kN]	302.0	19.6	[60] & estimated
SRM nozzle exit area [m^2]	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	
Beginning time T+[s]	0.0	155.0	[154]
Burn time [s]	155.0	245.0	[154]
Jettison time T+[s]	155.0	400.0	[154]

Note: alternative values can be found but they represent calculated estimations. [156]

Safir-1A

Input [Unit]	Value		Reference
Fairing diameter [m]	1.35		[153] (image)
Fairing length [m]	2.0		[153]
Fairing nose diameter [m]	0.45		[153]
Fairing nose length [m]	0.93		[153]
Fairing mass [kg]	100.0		[154]
Fairing jettison time [s]	150.0		[154]
Latitude of the launch [°]	35.24		[155]
Payload mass [kg]	15.3		[31] [32] [33]
Stages	Stage 0	Stage 1	
Length [m]	17.0	3.0	[60]
Diameter [m]	1.35	1.35	[153]
Propulsion System	P.0 (x1)	P.0 (x1)	
Fuel	UDMH	UDMH	[60]
Oxidizer	AK	AK	[60]
Propellant mass [tonnes]	21.4	2.7	[60]
Structure mass [tonnes]	2.6	0.3	From total mass [60]
Sea level thrust [kN]	284.4	19.6	[154]
Vacuum thrust [kN]	302.0	19.6	[60] & estimated
SRM nozzle exit area [m^2]	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	
Beginning time T+[s]	0.0	155.0	[154]
Burn time [s]	155.0	245.0	[154]
Jettison time T+[s]	155.0	400.0	[154]

Safir-1B

Input [Unit]	Value		Reference
Fairing diameter [m]	1.35		[153] (image)
Fairing length [m]	2.0		[153]
Fairing nose diameter [m]	0.45		[153]
Fairing nose length [m]	0.93		[153]
Fairing mass [kg]	100.0		[154]
Fairing jettison time [s]	150.0		[154]
Latitude of the launch [°]	35.24		[155]
Payload mass [kg]	Between 50 and 52		[31] [32] [33]
Stages	Stage 0	Stage 1	
Length [m]	17.0	3.0	[60]
Diameter [m]	1.35	1.35	[153]
Propulsion System	P.0 (x1)	P.0 (x1)	
Fuel	UDMH	UDMH	[60]
Oxidizer	AK	AK	[60]
Propellant mass [tonnes]	21.4	2.7	[60]
Structure mass [tonnes]	2.6	0.3	From total mass [60]
Sea level thrust [kN]	333.4	19.6	[60]
Vacuum thrust [kN]	355.0	19.6	[60] & estimated
SRM nozzle exit area [m^2]	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	
Beginning time T+[s]	0.0	155.0	[154]
Burn time [s]	155.0	245.0	[154]
Jettison time T+[s]	155.0	400.0	[154]

The Safir launch vehicle has different variants: Safir-1, Safir-1A and Safir-1B that are not different alternatives but upgrades of each other. Since the data is already pretty scarce on the Iranian rocket, this increases uncertainty on the values. We target specifically the Safir-1B version.

Alternative values can be found but they represent calculated estimations. [156]

Shavit-2

Input [Unit]	Value			Reference
Fairing diameter [m]	1.35			[60]
Fairing length [m]	3.86			[60]
Fairing nose diameter [m]	0.281			[157]
Fairing nose length [m]	1.5			[157]
Fairing mass [kg]	57.0			[158]
Fairing jettison time [s]	120.0			[159]
Latitude of the launch [°]	31.8			[160]
Payload mass [kg]	Between 100 and 370			[31] [32] [33]
Stages	Stage 0	Stage 1	Stage 2	
Length [m]	7.5	7.5	2.6	[159] [60]
Diameter [m]	2.3	2.3	2.3	[159]
Propulsion System	P.0 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	NH_4ClO_4	NH_4ClO_4	[159]
Oxidizer	Al	Al	Al	[159]
Propellant mass [tonnes]	12.75	12.75	1.878	[159] [60]
Structure mass [tonnes]	1.24	1.38	0.17	[159]
Sea level thrust [kN]	564.0	564.0	60.4	[159] [60]
Vacuum thrust [kN]	x	x	x	
SRM nozzle exit area [m^2]	x	x	x	
SRM specific impulse [s]	x	x	x	
Beginning time T+[s]	0.0	56.0	x	[159]
Burn time [s]	56.0	56.0	x	[159]
Jettison time T+[s]	56.0	112.0	x	[159]

Soyuz-2.1a

Input [Unit]	Value			Reference
Fairing diameter [m]	3.715			[130]
Fairing length [m]	11.433			[130]
Fairing nose diameter [m]	1.1			[130]
Fairing nose length [m]	7.7			[130]
Fairing mass [kg]	1000.0			Estimated
Fairing jettison time [s]	158.0			[161]
Latitude of the launch [°]	45.6 or 62.92			[130]
Payload mass [kg]	Between 4000 and 7419			[31] [32] [33]
Stages	Stage 0		Stage 1	
Length [m]	27.1		6.7	[130]
Diameter [m]	2.95		2.95	[130]
Propulsion System	P.0 (x4)	P.1 (x1)	P.0 (x1)	
Fuel	RP-1	RP-1	RP-1	[130]
Oxidizer	O_2	O_2	O_2	[130]
Propellant mass [tonnes]	39.6	92.95	22.89	[130]
Structure mass [tonnes]	3.8	6.55	2.41	[130]
Sea level thrust [kN]	782.5	680.5	297.9	[130]
Vacuum thrust [kN]	951.3	850.2	297.9	[130]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	
Beginning time T+[s]	0.0	0.0	288.0	[161]
Burn time [s]	118.0	288.0	240.0	[161]
Jettison time T+[s]	118.0	288.0	528.0	[161]

Differences between the Soyuz-2.1. a and the Soyuz-FG include conversion from analogue to digital flight control system.

Soyuz-2.1a(Fregat)

Input [Unit]	Value				Reference
Fairing diameter [m]	3.715				[130]
Fairing length [m]	11.433				[130]
Fairing nose diameter [m]	1.1				[130]
Fairing nose length [m]	7.7				[130]
Fairing mass [kg]	1000.0				Estimated
Fairing jettison time [s]	158.0				[161]
Latitude of the launch [°]	62.92 or 5.23				[130]
Payload mass [kg]	Between 2000 and 3713				[31] [32] [33]
Stages	Stage 0		Stage 1	Stage 2	
Length [m]	27.1		6.7	1.5	[130]
Diameter [m]	2.95		2.95	2.95	[130]
Propulsion System	P.0 (x4)	P.1 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	RP-1	RP-1	RP-1	UDMH	[130]
Oxidizer	O_2	O_2	O_2	N_2O_4	[130]
Propellant mass [tonnes]	39.6	92.95	22.89	5.35	[130]
Structure mass [tonnes]	3.8	6.55	2.41	1.0	[130]
Sea level thrust [kN]	782.5	680.5	297.9	19.614	[130]
Vacuum thrust [kN]	951.3	850.2	297.9	19.614	[130]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	N/A	
Beginning time T+[s]	0.0	0.0	288.0	x	[161]
Burn time [s]	118.0	288.0	240.0	x	[161]
Jettison time T+[s]	118.0	288.0	528.0	x	[161]

Soyuz-2.1a(Fregat-M)

Input [Unit]	Value				Reference
Fairing diameter [m]	3.715				[130]
Fairing length [m]	11.433				[130]
Fairing nose diameter [m]	1.1				[130]
Fairing nose length [m]	7.7				[130]
Fairing mass [kg]	1000.0				Estimated
Fairing jettison time [s]	158.0				[161]
Latitude of the launch [°]	45.6 or 62.92 or 51.88				[130]
Payload mass [kg]	Between 840 and 4900				[31] [32] [33]
Stages	Stage 0		Stage 1	Stage 2	
Length [m]	27.1		6.7	1.55	[130] [162]
Diameter [m]	2.95		2.95	2.95	[130]
Propulsion System	P.0 (x4)	P.1 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	RP-1	RP-1	RP-1	UDMH	[130] [162]
Oxidizer	O_2	O_2	O_2	N_2O_4	[130] [162]
Propellant mass [tonnes]	39.6	92.95	22.89	5.25	[130] [162]
Structure mass [tonnes]	3.8	6.55	2.41	0.92	[130] [162]
Sea level thrust [kN]	782.5	680.5	297.9	20.01	[130] [162]
Vacuum thrust [kN]	951.3	850.2	297.9	20.01	[130] [162]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	N/A	
Beginning time T+[s]	0.0	0.0	288.0	x	[161]
Burn time [s]	118.0	288.0	240.0	x	[161]
Jettison time T+[s]	118.0	288.0	528.0	x	[161]

Soyuz-2.1a(Volga)

Input [Unit]	Value				Reference
Fairing diameter [m]	3.715				[130]
Fairing length [m]	11.433				[130]
Fairing nose diameter [m]	1.1				[130]
Fairing nose length [m]	7.7				[130]
Fairing mass [kg]	1000.0				Estimated
Fairing jettison time [s]	158.0				[161]
Latitude of the launch [°]	51.88				[130]
Payload mass [kg]	949				[31] [32] [33]
Stages	Stage 0		Stage 1	Stage 2	
Length [m]	27.1		6.7	1.025	[130] [163]
Diameter [m]	2.95		2.95	3.2	[130] [163]
Propulsion System	P.0 (x4)	P.1 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	RP-1	RP-1	RP-1	UDMH	[130] [163]
Oxidizer	O_2	O_2	O_2	N_2O_4	[130] [163]
Propellant mass [tonnes]	39.6	92.95	22.89	0.6	[130] [163]
Structure mass [tonnes]	3.8	6.55	2.41	0.84	[130] [163]
Sea level thrust [kN]	782.5	680.5	297.9	2.99	[130] [163]
Vacuum thrust [kN]	951.3	850.2	297.9	2.99	[130] [163]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	N/A	
Beginning time T+[s]	0.0	0.0	288.0	x	[161]
Burn time [s]	118.0	288.0	240.0	x	[161]
Jettison time T+[s]	118.0	288.0	528.0	x	[161]

Soyuz-2.1a/ST(Fregat)

Input [Unit]	Value				Reference
Fairing diameter [m]	4.1				[164]
Fairing length [m]	11.4				[164]
Fairing nose diameter [m]	1.6				[164]
Fairing nose length [m]	4.255				[164]
Fairing mass [kg]	1700.0				[164]
Fairing jettison time [s]	158.0				[161]
Latitude of the launch [°]	5.23				[164]
Payload mass [kg]	Between 1070 and 2191				[31] [32] [33]
Stages	Stage 0		Stage 1	Stage 2	
Length [m]	27.1		27.1	1.5	[164] [130]
Diameter [m]	2.95		2.95	2.95	[164]
Propulsion System	P.0 (x4)	P.1 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	RP-1	RP-1	RP-1	UDMH	[164] [130]
Oxidizer	O_2	O_2	O_2	N_2O_4	[164] [130]
Propellant mass [tonnes]	39.16	90.1	22.89	5.35	[164] [130]
Structure mass [tonnes]	3.784	6.545	2.41	1.0	[164] [130]
Sea level thrust [kN]	782.5	680.5	297.9	19.614	[164] [130]
Vacuum thrust [kN]	951.3	850.2	297.9	19.614	[164] [130]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	N/A	
Beginning time T+[s]	0.0	0.0	288.0	x	[164] [161]
Burn time [s]	118.0	288.0	270.0	x	[164] [161]
Jettison time T+[s]	118.0	288.0	558.0	x	[164] [161]

Soyuz-2.1a/ST(Fregat-M)

Input [Unit]	Value				Reference
Fairing diameter [m]	4.1				[164]
Fairing length [m]	11.4				[164]
Fairing nose diameter [m]	1.6				[164]
Fairing nose length [m]	4.255				[164]
Fairing mass [kg]	1700.0				[164]
Fairing jettison time [s]	158.0				[161]
Latitude of the launch [°]	5.23				[164]
Payload mass [kg]	Between 2272 and 3099				[31] [32] [33]
Stages	Stage 0		Stage 1	Stage 2	
Length [m]	27.1		6.7	1.55	[164] [162]
Diameter [m]	2.95		2.95	2.95	[164]
Propulsion System	P.0 (x4)	P.1 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	RP-1	RP-1	RP-1	UDMH	[164] [162]
Oxidizer	O_2	O_2	O_2	N_2O_4	[164] [162]
Propellant mass [tonnes]	39.16	90.1	22.89	5.25	[164] [162]
Structure mass [tonnes]	3.784	6.545	2.41	0.92	[164] [162]
Sea level thrust [kN]	782.5	680.5	297.9	20.01	[164] [162]
Vacuum thrust [kN]	951.3	850.2	297.9	20.01	[164] [162]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	N/A	
Beginning time T+[s]	0.0	0.0	288.0	x	[164] [161]
Burn time [s]	118.0	288.0	270.0	x	[164] [161]
Jettison time T+[s]	118.0	288.0	558.0	x	[164] [161]

Soyuz-2.1b

Input [Unit]	Value			Reference
Fairing diameter [m]	3.715			[130]
Fairing length [m]	11.433			[130]
Fairing nose diameter [m]	1.1			[130]
Fairing nose length [m]	7.7			[130]
Fairing mass [kg]	1000.0			Estimated
Fairing jettison time [s]	158.0			[161]
Latitude of the launch [°]	62.92 or 45.6			[130]
Payload mass [kg]	Between 5920 and 7142			[31] [32] [33]
Stages	Stage 0		Stage 1	
Length [m]	27.1		27.1	[130]
Diameter [m]	2.95		2.95	[130]
Propulsion System	P.0 (x4)	P.1 (x1)	P.0 (x1)	
Fuel	RP-1	RP-1	RP-1	[130]
Oxidizer	O_2	O_2	O_2	[130]
Propellant mass [tonnes]	39.6	92.95	22.89	[130]
Structure mass [tonnes]	3.8	6.55	2.41	[130]
Sea level thrust [kN]	782.5	680.5	297.9	[130]
Vacuum thrust [kN]	951.3	850.2	297.9	[130]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	
Beginning time T+[s]	0.0	0.0	288.0	[161] [164]
Burn time [s]	118.0	288.0	270.0	[161] [164]
Jettison time T+[s]	118.0	288.0	558.0	[161] [164]

For Stage 1, the RD-0110 engine is replaced by the RD-0124. RD-0124 has the same thrust as RD-0110 but higher specific impulse thus smaller mass flow rate. [165] We assume that the propellant mass is not changed from Soyuz-2.1. a. The burn time is taken from [164].

Soyuz-2.1b(Fregat)

Input [Unit]	Value				Reference
Fairing diameter [m]	3.715				[130]
Fairing length [m]	11.433				[130]
Fairing nose diameter [m]	1.1				[130]
Fairing nose length [m]	7.7				[130]
Fairing mass [kg]	1000.0				Estimated
Fairing jettison time [s]	158.0				[161]
Latitude of the launch [°]	45.6				[130]
Payload mass [kg]	3300				[31] [32] [33]
Stages	Stage 0		Stage 1	Stage 2	
Length [m]	27.1		27.1	1.5	[130]
Diameter [m]	2.95		2.95	2.95	[130]
Propulsion System	P.0 (x4)	P.1 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	RP-1	RP-1	RP-1	UDMH	[130]
Oxidizer	O_2	O_2	O_2	N_2O_4	[130]
Propellant mass [tonnes]	39.6	92.95	22.89	5.35	[130]
Structure mass [tonnes]	3.8	6.55	2.41	1.0	[130]
Sea level thrust [kN]	782.5	680.5	297.9	19.614	[130]
Vacuum thrust [kN]	951.3	850.2	297.9	19.614	[130]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	N/A	
Beginning time T+[s]	0.0	0.0	288.0	x	[161] [164]
Burn time [s]	118.0	288.0	270.0	x	[161] [164]
Jettison time T+[s]	118.0	288.0	558.0	x	[161] [164]

Soyuz-2.1b(Fregat-M)

Input [Unit]	Value				Reference
Fairing diameter [m]	3.715				[130]
Fairing length [m]	11.433				[130]
Fairing nose diameter [m]	1.1				[130]
Fairing nose length [m]	7.7				[130]
Fairing mass [kg]	1000.0				Estimated
Fairing jettison time [s]	158.0				[161]
Latitude of the launch [°]	62.92 or 45.6 or 51.88				[130]
Payload mass [kg]	Between 935 and 3200				[31] [32] [33]
Stages	Stage 0		Stage 1	Stage 2	
Length [m]	27.1		27.1	1.55	[130] [162]
Diameter [m]	2.95		2.95	2.95	[130]
Propulsion System	P.0 (x4)	P.1 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	RP-1	RP-1	RP-1	UDMH	[130] [162]
Oxidizer	O_2	O_2	O_2	N_2O_4	[130] [162]
Propellant mass [tonnes]	39.6	92.95	22.89	5.25	[130] [162]
Structure mass [tonnes]	3.8	6.55	2.41	0.92	[130] [162]
Sea level thrust [kN]	782.5	680.5	297.9	20.01	[130] [162]
Vacuum thrust [kN]	951.3	850.2	297.9	20.01	[130] [162]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	N/A	
Beginning time T+[s]	0.0	0.0	288.0	x	[161] [164]
Burn time [s]	118.0	288.0	270.0	x	[161] [164]
Jettison time T+[s]	118.0	288.0	558.0	x	[161] [164]

Soyuz-2.1b/ST(Fregat-M)

Input [Unit]	Value				Reference
Fairing diameter [m]	4.1				[164]
Fairing length [m]	11.4				[164]
Fairing nose diameter [m]	1.6				[164]
Fairing nose length [m]	4.255				[164]
Fairing mass [kg]	1700.0				[164]
Fairing jettison time [s]	158.0				[161]
Latitude of the launch [°]	5.23				[164]
Payload mass [kg]	4212				[31] [32] [33]
Stages	Stage 0		Stage 1	Stage 2	
Length [m]	27.1		27.1	1.55	[164] [162]
Diameter [m]	2.95		2.95	2.95	[164]
Propulsion System	P.0 (x4)	P.1 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	RP-1	RP-1	RP-1	UDMH	[164] [162]
Oxidizer	O_2	O_2	O_2	N_2O_4	[164] [162]
Propellant mass [tonnes]	39.16	90.1	22.89	5.25	[164] [162]
Structure mass [tonnes]	3.784	6.545	2.41	0.92	[164] [162]
Sea level thrust [kN]	782.5	680.5	297.9	20.01	[164] [162]
Vacuum thrust [kN]	951.3	850.2	297.9	20.01	[164] [162]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	N/A	
Beginning time T+[s]	0.0	0.0	288.0	x	[164] [161]
Burn time [s]	118.0	288.0	270.0	x	[164] [161]
Jettison time T+[s]	118.0	288.0	558.0	x	[164] [161]

Soyuz-2.1b/ST(Fregat-MT)

Input [Unit]	Value				Reference
Fairing diameter [m]	4.1				[164]
Fairing length [m]	11.4				[164]
Fairing nose diameter [m]	1.6				[164]
Fairing nose length [m]	4.255				[164]
Fairing mass [kg]	1700.0				[164]
Fairing jettison time [s]	158.0				[161]
Latitude of the launch [°]	5.23				[164]
Payload mass [kg]	Between 1580 and 3343				[31] [32] [33]
Stages	Stage 0		Stage 1	Stage 2	
Length [m]	27.1		27.1	1.55	[164] [162]
Diameter [m]	2.95		2.95	2.95	[164]
Propulsion System	P.0 (x4)	P.1 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	RP-1	RP-1	RP-1	UDMH	[164] [162]
Oxidizer	O_2	O_2	O_2	N_2O_4	[164] [162]
Propellant mass [tonnes]	39.16	90.1	22.89	6.55	[164] [162]
Structure mass [tonnes]	3.784	6.545	2.41	1.03	[164] [162]
Sea level thrust [kN]	782.5	680.5	297.9	20.01	[164] [162]
Vacuum thrust [kN]	951.3	850.2	297.9	20.01	[164] [162]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	N/A	
Beginning time T+[s]	0.0	0.0	288.0	x	[164] [161]
Burn time [s]	118.0	288.0	270.0	x	[164] [161]
Jettison time T+[s]	118.0	288.0	558.0	x	[164] [161]

Soyuz-2.1v

Input [Unit]	Value		Reference
Fairing diameter [m]	3.715		[130]
Fairing length [m]	11.433		[130]
Fairing nose diameter [m]	1.1		[130]
Fairing nose length [m]	7.7		[130]
Fairing mass [kg]	1000.0		Estimated
Fairing jettison time [s]	158.0		[130]
Latitude of the launch [°]	62.92		[130]
Payload mass [kg]	150		[31] [32] [33]
Stages	Stage 0	Stage 1	
Length [m]	27.7	6.7	[130] [166]
Diameter [m]	2.95	2.95	[130]
Propulsion System	P.0 (x1)	P.0 (x1)	
Fuel	RP-1	RP-1	[131]
Oxidizer	O_2	O_2	[131]
Propellant mass [tonnes]	119.0	22.89	[131] [130]
Structure mass [tonnes]	10.0	2.41	[131] [130]
Sea level thrust [kN]	1748.8	297.9	[131]
Vacuum thrust [kN]	1911.0	297.9	[131]
SRM nozzle exit area [m^2]	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	
Beginning time T+[s]	0.0	210.0	[131]
Burn time [s]	210.0	270.0	[131]
Jettison time T+[s]	210.0	480.0	[131]

Soyuz-2.1v(Volga)

Input [Unit]	Value			Reference
Fairing diameter [m]	3.715			[130]
Fairing length [m]	11.433			[130]
Fairing nose diameter [m]	1.1			[130]
Fairing nose length [m]	7.7			[130]
Fairing mass [kg]	1000.0			Estimated
Fairing jettison time [s]	158.0			[130]
Latitude of the launch [°]	62.92			[130]
Payload mass [kg]	Between 143 and 441			[31] [32] [33]
Stages	Stage 0	Stage 1	Stage 2	
Length [m]	27.7	6.7	1.025	[130] [166] [163]
Diameter [m]	2.95	2.95	3.2	[130] [163]
Propulsion System	P.0 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	RP-1	RP-1	UDMH	[131] [163]
Oxidizer	O_2	O_2	N_2O_4	[131] [163]
Propellant mass [tonnes]	119.0	22.89	0.6	[131] [130] [163]
Structure mass [tonnes]	10.0	2.41	0.84	[131] [130] [163]
Sea level thrust [kN]	1748.8	297.9	2.99	[131]
Vacuum thrust [kN]	1911.0	297.9	2.99	[131]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	
Beginning time T+[s]	0.0	210.0	x	[131]
Burn time [s]	210.0	270.0	x	[131]
Jettison time T+[s]	210.0	480.0	x	[131]

Soyuz-FG

Input [Unit]	Value			Reference
Fairing diameter [m]	3.715			[130]
Fairing length [m]	11.433			[130]
Fairing nose diameter [m]	1.1			[130]
Fairing nose length [m]	7.7			[130]
Fairing mass [kg]	1000.0			Estimated
Fairing jettison time [s]	158.0			[161]
Latitude of the launch [°]	45.6			[130]
Payload mass [kg]	Between 7380 and 7520			[31] [32] [33]
Stages	Stage 0		Stage 1	
Length [m]	27.1		6.7	[130]
Diameter [m]	2.95		2.95	[130]
Propulsion System	P.0 (x4)	P.1 (x1)	P.0 (x1)	
Fuel	RP-1	RP-1	RP-1	[130]
Oxidizer	O_2	O_2	O_2	[130]
Propellant mass [tonnes]	39.6	92.95	22.89	[130]
Structure mass [tonnes]	3.8	6.55	2.41	[130]
Sea level thrust [kN]	782.5	680.5	297.9	[130]
Vacuum thrust [kN]	951.3	850.2	297.9	[130]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	
Beginning time T+[s]	0.0	0.0	288.0	[161]
Burn time [s]	118.0	288.0	240.0	[161]
Jettison time T+[s]	118.0	288.0	528.0	[161]

Soyuz-FG(Fregat)

Input [Unit]	Value				Reference
Fairing diameter [m]	3.715				[130]
Fairing length [m]	11.433				[130]
Fairing nose diameter [m]	1.1				[130]
Fairing nose length [m]	7.7				[130]
Fairing mass [kg]	1000.0				Estimated
Fairing jettison time [s]	158.0				[161]
Latitude of the launch [°]	45.6				[130]
Payload mass [kg]	1423				[31] [32] [33]
Stages	Stage 0		Stage 1	Stage 2	
Length [m]	27.1		6.7	1.5	[130]
Diameter [m]	2.95		2.95	2.95	[130]
Propulsion System	P.0 (x4)	P.1 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	RP-1	RP-1	RP-1	UDMH	[130]
Oxidizer	O_2	O_2	O_2	N_2O_4	[130]
Propellant mass [tonnes]	39.6	92.95	22.89	5.35	[130]
Structure mass [tonnes]	3.8	6.55	2.41	1.0	[130]
Sea level thrust [kN]	782.5	680.5	297.9	19.614	[130]
Vacuum thrust [kN]	951.3	850.2	297.9	19.614	[130]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	N/A	
Beginning time T+[s]	0.0	0.0	288.0	x	[161]
Burn time [s]	118.0	288.0	240.0	x	[161]
Jettison time T+[s]	118.0	288.0	528.0	x	[161]

Soyuz-U

Input [Unit]	Value			Reference
Fairing diameter [m]	3.0			[28]
Fairing length [m]	7.9			[28]
Fairing nose diameter [m]	0.3			[28]
Fairing nose length [m]	3.44			[28]
Fairing mass [kg]	4500.0			[28]
Fairing jettison time [s]	187.0			[130]
Latitude of the launch [°]	45.6 or 62.92			[130]
Payload mass [kg]	Between 1050 and 7450			[31] [32] [33]
Stages	Stage 0		Stage 1	
Length [m]	27.1		6.7	[130]
Diameter [m]	2.6		2.6	[130]
Propulsion System	P.0 (x1)	P.1 (x1)	P.0 (x1)	
Fuel	RP-1	RP-1	RP-1	[130]
Oxidizer	O_2	O_2	O_2	[130]
Propellant mass [tonnes]	158.4	92.95	22.59	[130]
Structure mass [tonnes]	15.2	6.55	2.71	[130]
Sea level thrust [kN]	3255.808	745.305	298.024	[131]
Vacuum thrust [kN]	3966.984	941.438	298.024	[131]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	
Beginning time T+[s]	0.0	0.0	280.0	[130]
Burn time [s]	118.0	280.0	240.0	[130]
Jettison time T+[s]	118.0	280.0	520.0	[130]

Space Shuttle-Atlantis

Input [Unit]	Value			Reference
Fairing diameter [m]	8.4			[28]
Fairing length [m]	10.8			[28]
Fairing nose diameter [m]	1.12			[28]
Fairing nose length [m]	10.8			[28]
Fairing mass [kg]	N/A			
Fairing jettison time [s]	N/A			
Latitude of the launch [°]	28.52			[28]
Payload mass [kg]	Between 10000 and 16027			[31] [32] [33]
Stages	Stage 0		Stage 1	
Length [m]	49.0		0.0	[28]
Diameter [m]	8.4		8.4	[28]
Propulsion System	P.0 (x2)	P.1 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	H_2	MMH	[28]
Oxidizer	Al	O_2	N_2O_4	[28]
Propellant mass [tonnes]	502.0	721.0	14.2	[28]
Structure mass [tonnes]	88.0	27.0	78.4	[28]
Sea level thrust [kN]	12450.0	5004.0	26.7	[167]
Vacuum thrust [kN]	Thrust profile (Solid)	6271.992	26.7	[167] [168]
SRM nozzle exit area [m^2]	11.35	N/A	N/A	[169]
SRM specific impulse [s]	268.0	N/A	N/A	[12]
Beginning time T+[s]	6.0	0.0	2638.0	[170]
Burn time [s]	123.0	512.0	96.0	[170]
Jettison time T+[s]	130.0	530.0	2734.0	[170]

Space Shuttle-Discovery

Input [Unit]	Value			Reference
Fairing diameter [m]	8.4			[28]
Fairing length [m]	10.8			[28]
Fairing nose diameter [m]	1.12			[28]
Fairing nose length [m]	10.8			[28]
Fairing mass [kg]	N/A			
Fairing jettison time [s]	N/A			
Latitude of the launch [°]	28.52			[28]
Payload mass [kg]	Between 14120 and 16600			[31] [32] [33]
Stages	Stage 0		Stage 1	
Length [m]	49.0		0.0	[28]
Diameter [m]	8.4		8.4	[28]
Propulsion System	P.0 (x2)	P.1 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	H_2	MMH	[28]
Oxidizer	Al	O_2	N_2O_4	[28]
Propellant mass [tonnes]	502.0	721.0	14.2	[28]
Structure mass [tonnes]	88.0	27.0	78.7	[28]
Sea level thrust [kN]	12450.0	5004.0	26.7	[167]
Vacuum thrust [kN]	Thrust profile (Solid)	6271.992	26.7	[167] [168]
SRM nozzle exit area [m^2]	11.35	N/A	N/A	[169]
SRM specific impulse [s]	268.0	N/A	N/A	[12]
Beginning time T+[s]	6.0	0.0	2638.0	[170]
Burn time [s]	123.0	512.0	96.0	[170]
Jettison time T+[s]	130.0	530.0	2734.0	[170]

Space Shuttle-Endeavour

Input [Unit]	Value			Reference
Fairing diameter [m]	8.4			[28]
Fairing length [m]	10.8			[28]
Fairing nose diameter [m]	1.12			[28]
Fairing nose length [m]	10.8			[28]
Fairing mass [kg]	N/A			
Fairing jettison time [s]	N/A			
Latitude of the launch [°]	28.52			[28]
Payload mass [kg]	Between 13645 and 16500			[31] [32] [33]
Stages	Stage 0		Stage 1	
Length [m]	49.0		0.0	[28]
Diameter [m]	8.4		8.4	[28]
Propulsion System	P.0 (x2)	P.1 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	H_2	MMH	[28]
Oxidizer	Al	O_2	N_2O_4	[28]
Propellant mass [tonnes]	502.0	721.0	14.2	[28]
Structure mass [tonnes]	88.0	27.0	78.8	[28]
Sea level thrust [kN]	12450.0	5004.0	26.7	[167]
Vacuum thrust [kN]	Thrust profile (Solid)	6271.992	26.7	[167] [168]
SRM nozzle exit area [m^2]	11.35	N/A	N/A	[169]
SRM specific impulse [s]	268.0	N/A	N/A	[12]
Beginning time T+[s]	6.0	0.0	2638.0	[170]
Burn time [s]	123.0	512.0	96.0	[170]
Jettison time T+[s]	130.0	530.0	2734.0	[170]

SS-520

Input [Unit]	Value			Reference
Fairing diameter [m]	0.52			[171]
Fairing length [m]	1.0			[171]
Fairing nose diameter [m]	0.12			[171]
Fairing nose length [m]	0.8			[171]
Fairing mass [kg]	1.6			[172]
Fairing jettison time [s]	67.0			[173]
Latitude of the launch [°]	31.25			[96]
Payload mass [kg]	Between 3 and 3			[31] [32] [33]
Stages	Stage 0	Stage 1	Stage 2	
Length [m]	7.0	1.7	0.0	[174] (Estimated)
Diameter [m]	0.52	0.52	0.52	[174]
Propulsion System	P.0 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	NH_4ClO_4	NH_4ClO_4	[60]
Oxidizer	Al	Al	Al	[60]
Propellant mass [tonnes]	1.587	0.325	0.078	[60]
Structure mass [tonnes]	0.57	0.028	0.008	From [60] [174] and [171]
Sea level thrust [kN]	176.5	x	x	[172]
Vacuum thrust [kN]	x	x	x	[175]
SRM nozzle exit area [m^2]	x	x	x	
SRM specific impulse [s]	x	x	x	
Beginning time T+[s]	0.0	180.0	238.0	[173]
Burn time [s]	32.0	25.0	26.0	[173]
Jettison time T+[s]	68.0	235.0	450.0	[173]

Strela

Input [Unit]	Value				Reference
Fairing diameter [m]	1.55				[28]
Fairing length [m]	5.4				[28]
Fairing nose diameter [m]	0.298				[176] (Picture)
Fairing nose length [m]	1.94				[176]
Fairing mass [kg]	400.0				[177]
Fairing jettison time [s]	164.0				[176]
Latitude of the launch [°]	45.6				[28]
Payload mass [kg]	Between 1150 and 1150				[31] [32] [33]
Stages	Stage 0	Stage 1		Stage 2	
Length [m]	17.2	3.9		0.5	[176]
Diameter [m]	2.5	2.5		2.5	[176]
Propulsion System	P.0 (x1)	P.0 (x1)	P.1 (x1)	P.0 (x1)	
Fuel	UDMH	UDMH	UDMH	UDMH	[176]
Oxidizer	N_2O_4	N_2O_4	N_2O_4	N_2O_4	[176]
Propellant mass [tonnes]	71.5	10.71	0.0	0.375	[176]
Structure mass [tonnes]	5.7	1.485	0.0	0.725	[176]
Sea level thrust [kN]	1870.0	240.0	15.76	4.9	[176]
Vacuum thrust [kN]	2070.0	240.0	15.76	4.9	[176]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	N/A	
Beginning time T+[s]	0.0	127.0	127.0	1575.0	[176]
Burn time [s]	126.0	183.0	183.0	185.0	[176]
Jettison time T+[s]	126.0	310.0	310.0	1760.0	[176]

Tsiklon-3

Input [Unit]	Value			Reference
Fairing diameter [m]	2.7			[28]
Fairing length [m]	9.54			[28]
Fairing nose diameter [m]	0.4			[28]
Fairing nose length [m]	3.58			[28]
Fairing mass [kg]	1000.0			[178]
Fairing jettison time [s]	200.0			[28]
Latitude of the launch [°]	62.92			[28]
Payload mass [kg]	1900			[31] [32] [33]
Stages	Stage 0	Stage 1	Stage 2	
Length [m]	18.9	10.0	2.72	[28]
Diameter [m]	3.0	3.0	3.0	[28]
Propulsion System	P.0 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	UDMH	UDMH	UDMH	[28]
Oxidizer	N_2O_4	N_2O_4	N_2O_4	[28]
Propellant mass [tonnes]	121.0	49.0	3.2	[28]
Structure mass [tonnes]	6.3	3.5	1.4	[28]
Sea level thrust [kN]	2745.0	995.0	78.7	[28]
Vacuum thrust [kN]	3032.0	995.0	78.7	[28] [179]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	
Beginning time T+[s]	0.0	120.0	280.0	[28]
Burn time [s]	120.0	160.0	125.0	[28]
Jettison time T+[s]	120.0	280.0	405.0	[28]

Unha-2

Input [Unit]	Value			Reference
Fairing diameter [m]	1.3			[60]
Fairing length [m]	2.0			[60]
Fairing nose diameter [m]	0.22			[60]
Fairing nose length [m]	1.34			[60]
Fairing mass [kg]	100.0			[180]
Fairing jettison time [s]	200.0			Estimated
Latitude of the launch [°]	40.85			[181]
Payload mass [kg]	250			[31] [32] [33]
Stages	Stage 0	Stage 1	Stage 2	
Length [m]	15.0	9.3	3.7	[60]
Diameter [m]	2.4	2.4	2.4	[182]
Propulsion System	P.0 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	UDMH	UDMH	RP-1	[182]
Oxidizer	N_2O_4	N_2O_4	O_2	[182]
Propellant mass [tonnes]	65.0	10.0	3.2	[182]
Structure mass [tonnes]	10.0	2.0	0.3	[182]
Sea level thrust [kN]	1100.0	250.0	54.0	[60]
Vacuum thrust [kN]	x	250.0	54.0	[60]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	
Beginning time T+[s]	0.0	135.0	365.0	[182]
Burn time [s]	135.0	230.0	200.0	[182]
Jettison time T+[s]	135.0	365.0	565.0	[182]

There are two versions during the time period we are looking at: Unha-2 and Unha-3. Unha-2 uses a solid upper stage. However, Unha-2 was launched only once, in 2009, and its upper stage that was above 100 km of altitude did not ignite [183]. Since we have little information on this North-Korean launch vehicle, we target mostly the data we have from Unha-3.

Unha-3

Input [Unit]	Value			Reference
Fairing diameter [m]	1.3			[60]
Fairing length [m]	2.0			[60]
Fairing nose diameter [m]	0.22			[60]
Fairing nose length [m]	1.34			[60]
Fairing mass [kg]	100.0			[180]
Fairing jettison time [s]	200.0			Estimated
Latitude of the launch [°]	39.66			[181]
Payload mass [kg]	Between 100 and 200			[31] [32] [33]
Stages	Stage 0	Stage 1	Stage 2	
Length [m]	15.0	9.3	3.7	[60]
Diameter [m]	2.4	2.4	2.4	[182]
Propulsion System	P.0 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	UDMH	UDMH	RP-1	[182]
Oxidizer	N_2O_4	N_2O_4	O_2	[182]
Propellant mass [tonnes]	65.0	10.0	3.2	[182]
Structure mass [tonnes]	10.0	2.0	0.3	[182]
Sea level thrust [kN]	1100.0	250.0	54.0	[60]
Vacuum thrust [kN]	x	250.0	54.0	[60]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	
Beginning time T+[s]	0.0	135.0	365.0	[182]
Burn time [s]	135.0	230.0	200.0	[182]
Jettison time T+[s]	135.0	365.0	565.0	[182]

Vega

Input [Unit]	Value				Reference
Fairing diameter [m]	2.6				[184]
Fairing length [m]	7.88				[184]
Fairing nose diameter [m]	0.536				[184]
Fairing nose length [m]	2.632				[184]
Fairing mass [kg]	540.0				[184]
Fairing jettison time [s]	231.0				[184]
Latitude of the launch [°]	5.23				[184]
Payload mass [kg]	Between 638 and 1932				[31] [32] [33]
Stages	Stage 0	Stage 1	Stage 2	Stage 3	
Length [m]	11.2	8.39	4.12	2.04	[184]
Diameter [m]	1.9	1.9	1.9	1.9	[184]
Propulsion System	P.0 (x1)	P.0 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	NH_4ClO_4	NH_4ClO_4	NH_4ClO_4	UDMH	[184]
Oxidizer	Al	Al	Al	N_2O_4	[184]
Propellant mass [tonnes]	87.71	23.814	10.567	0.577	[184]
Structure mass [tonnes]	8.533	2.486	1.433	0.111	[184]
Sea level thrust [kN]	3015.0	1120.0	317.0	2.45	[184] [185]
Vacuum thrust [kN]	Thrust profile (Solid)	Thrust profile (Solid)	Thrust profile (Solid)	2.45	[184] [185]
SRM nozzle exit area [m^2]	7.07	2.88	2.88	N/A	[186]
SRM specific impulse [s]	280.0	287.5	295.9	N/A	[184]
Beginning time T+[s]	0.0	115.0	226.0	354.0	[184]
Burn time [s]	115.0	87.0	128.0	520.0	[184]
Jettison time T+[s]	115.0	202.0	354.0	x	[184]

Zenit-2SLB(Fregat-SB)

Input [Unit]	Value			Reference
Fairing diameter [m]	3.9			[187]
Fairing length [m]	13.7			[187]
Fairing nose diameter [m]	0.6			[187]
Fairing nose length [m]	4.5			[187]
Fairing mass [kg]	8367.0			[188]
Fairing jettison time [s]	291.0			[187]
Latitude of the launch [°]	45.6			[187]
Payload mass [kg]	2000			[31] [32] [33]
Stages	Stage 0	Stage 1	Stage 2	
Length [m]	32.9	10.4	2.5	[187]
Diameter [m]	3.9	3.9	3.9	[187]
Propulsion System	P.0 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	RP-1	RP-1	UDMH	[187]
Oxidizer	O_2	O_2	N_2O_4	[187]
Propellant mass [tonnes]	326.78	82.4	10.15	[187]
Structure mass [tonnes]	27.564	8.367	1.41	[187]
Sea level thrust [kN]	7256.921	991.452	79.463	[187]
Vacuum thrust [kN]	7908.083	991.452	79.463	[187]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	
Beginning time T+[s]	0.0	151.0	x	[187]
Burn time [s]	143.0	242.0	x	[187]
Jettison time T+[s]	145.0	890.0	x	[187]

Zenit-3F

Input [Unit]	Value			Reference
Fairing diameter [m]	4.1			[187]
Fairing length [m]	10.4			[187]
Fairing nose diameter [m]	0.7			[187]
Fairing nose length [m]	3.9			[187]
Fairing mass [kg]	1800.0			[188]
Fairing jettison time [s]	315.0			[187]
Latitude of the launch [°]	45.6			[187]
Payload mass [kg]	Between 1647 and 3660			[31] [32] [33]
Stages	Stage 0	Stage 1	Stage 2	
Length [m]	32.9	10.4	2.5	[187]
Diameter [m]	3.9	3.9	3.9	[187]
Propulsion System	P.0 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	RP-1	RP-1	UDMH	[187]
Oxidizer	O_2	O_2	N_2O_4	[187]
Propellant mass [tonnes]	326.78	82.4	10.15	[187]
Structure mass [tonnes]	27.564	8.307	1.41	[187]
Sea level thrust [kN]	7256.921	7256.921	79.463	[187]
Vacuum thrust [kN]	7908.083	7908.083	79.463	[187]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	
Beginning time T+[s]	0.0	150.0	x	[187]
Burn time [s]	143.0	278.0	x	[187]
Jettison time T+[s]	145.0	504.0	x	[187]

Zenit-3SL

Input [Unit]	Value			Reference
Fairing diameter [m]	3.9			[187]
Fairing length [m]	11.39			[189]
Fairing nose diameter [m]	1.7			[189]
Fairing nose length [m]	4.69			[189]
Fairing mass [kg]	1800.0			[188]
Fairing jettison time [s]	206.0			[189]
Latitude of the launch [°]	0.0			[189]
Payload mass [kg]	Between 3038 and 6241			[31] [32] [33]
Stages	Stage 0	Stage 1	Stage 2	
Length [m]	32.9	10.4	1.1	[187]
Diameter [m]	3.9	3.9	3.9	[187]
Propulsion System	P.0 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	RP-1	RP-1	RP-1	[187]
Oxidizer	O_2	O_2	O_2	[187]
Propellant mass [tonnes]	325.401	81.854	14.6	[187] [189]
Structure mass [tonnes]	28.469	10.619	4.0	[187] [189]
Sea level thrust [kN]	7234.365	932.612	78.453	[189]
Vacuum thrust [kN]	7883.566	932.612	78.453	[189]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	
Beginning time T+[s]	0.0	151.0	527.0	[189]
Burn time [s]	143.0	291.0	642.0	[189]
Jettison time T+[s]	146.0	517.0	1669.0	[189]

Zenit-3SLB

Input [Unit]	Value			Reference
Fairing diameter [m]	4.1			[187]
Fairing length [m]	10.4			[187]
Fairing nose diameter [m]	0.7			[187]
Fairing nose length [m]	3.9			[187]
Fairing mass [kg]	1800.0			[188]
Fairing jettison time [s]	315.0			[187]
Latitude of the launch [°]	45.6			[187]
Payload mass [kg]	Between 2370 and 4012			[31] [32] [33]
Stages	Stage 0	Stage 1	Stage 2	
Length [m]	32.9	10.4	1.03	[187]
Diameter [m]	3.9	3.9	3.9	[187]
Propulsion System	P.0 (x1)	P.0 (x1)	P.0 (x1)	
Fuel	RP-1	RP-1	RP-1	[187]
Oxidizer	O_2	O_2	O_2	[187]
Propellant mass [tonnes]	326.78	82.4	17.8	[187]
Structure mass [tonnes]	27.564	8.307	3.22	[187]
Sea level thrust [kN]	7256.921	7256.921	991.452	[187]
Vacuum thrust [kN]	7908.083	7908.083	991.452	[187]
SRM nozzle exit area [m^2]	N/A	N/A	N/A	
SRM specific impulse [s]	N/A	N/A	N/A	
Beginning time T+[s]	0.0	150.0	513.0	[187]
Burn time [s]	143.0	278.0	190.0	[187]
Jettison time T+[s]	145.0	504.0	x	[187]

References

[1] E. Kyle, «Space Launch Report: Orbital Launch Summary by Year».*https://www.spacelaunchreport.com/logyear.html.*

[2] O. Schmid, J. M. Reeves, J. C. Wilson, C. Wiedinmyer, C. A. Brock, D. W. Toohey, L. M. Avallone, A. M. Gates et M. N. Ross, «Size-resolved particle emission indices in the stratospheric plume of an Athena II rocket,» *Journal of geophysical research,* 2002.

[3] C. H. Jackman, D. B. Considine et E. L. Flemming, «A global modelling study of solid rocket aluminum oxide emission effects on stratospheric ozone,» *Geophysical Research Letters,* 1998.

[4] M. Ross, D. Toohey, M. Peinemann et P. Ross, «Limits on the space launch market related to stratospheric ozone depletion,» *Astropolitics,* vol. 7, 2009.

[5] M. N. Ross et P. M. Sheaffer, «Radiative forcing caused by rocket engine emissions,» *Earth's Future, 2, 177–196,* 2014.

[6] D. B. Kirk-Davidoff, E. J. Hintsa, G. J. Anderson et D. W. Keith, «The effect of climate change on ozone depletion through changes in stratospheric water vapour,» *Nature,* vol. 402, 1999.

[7] M. H. Stevens, R. R. Meier, X. Chu, M. T. DeLand et J. M. Plane, «Antarctic mesospheric clouds formed from space shuttle exhaust,» *Geophysical Research Letters 32,* 2005.

[8] B. B. Brady, E. W. Fournier, L. R. Martin et R. B. Cohen, «Stratospheric Ozone Reactive Chemicals Generated by Space Launches Worldwide,» Prepared for Space and Missile Systems Center, Air Force Materiel Command, El Segundo, CA, 1994.

[9] J. D. DeSain et B. B. Brady, «Potential Atmospheric Impact Generated by Space Launches Worldwide - Update for Emission Estimated from 1985 to 2013,» Aerospace Report prepared for Space and Missile Systems Center, Air Force Space Command, El Segundo, CA, 2014.

[10] J.-I. Shu, J.-W. Kim, J.-W. Lee et S. Kim, «Multidisciplinary mission design optimization for space launch vehicles based on sequential design process,» *Journal of Aerospace Engineering,* vol. 230, 2016.

[11] E. L. Fleeman, «Missile Design and Systems Engineering,» 2012.

[12] G. P. Sutton et O. Biblarz, Rocket propulsion elements, Wiley, 2016.

[13] SpaceX, Falcon 9 Launch Vehicle PAYLOAD USER'S GUIDE, 2019.

[14] C. Bruno, «Spacecraft Propulsion,» *The International Handbook of Space Technology,* 2014.

[15] D. Lee, D. Fahey, A. Skowron, M. Allen, U. Burkhardt, Q. Chen, S. Doherty, S. Freeman, P. Forster, J. Fuglestvedt, A. Gettelman, R. De Leon, L. Lim, M. Lund, R. Millar, B. Owen, J. Penner, G. Pitari, M. Prather, R. Sausen et L. Wilcox, «The contribution of global aviation to anthropogenic climate forcing for 2000 to 2018,» *Atmospheric Environment,* 2020.

[16] D. Fourie, I. M. Hedgecok, F. De Simone, E. M. Sunderland et N. Pirrone, «Are mercury emissions from satellite electric propulsion an environmental?,» *Environmental Research Letters,* vol. 14, n° %112, 2019.

[17] E. Kyle, «Space Launch Report Archive 2015 - October-December, 2015».*http://spacelaunchreport.com/slr2015q4.html.*

[18] M. C. Baldwin et P. Lu, «Optimal Deorbit Guidance,» *Journal of Guidance, Control, and Dynamics,* vol. 35, n° %11, 2012.

[19] E. Kyle, «Space Launch Report Archive - January-December, 2017».*Delta II launch from November 18, 2017: http://spacelaunchreport.com/slr2017q4.html.*

[20] E. Kyle, «Space Launch Report Archive - January-December, 2018».*Rokot launch from November 30, 2018: http://www.spacelaunchreport.com/slr2018q4.html.*

[21] J. Zinn, C. D. Sutherland, S. N. Stone, L. M. Duncan et R. Behnke, «Ionospheric effects of rocket exhaust products- heao-c, skylab,» *Journal of Atmospheric and Terrestrial Physics,* vol. 44, 1982.

[22] P. R. Ratcliff, A. D. Taylor et J. A. McDonnell, «The LEO microparticle population: Computer studies of space debris drag depletion and of interplanetary capture processes,» *Advances in Space Research,* vol. 13, 1993.

[23] E. Richter, H. J. Fahr et H. U. Nass, «Satellite particle exospheres of plantes: Application to Earth,» *Planetary and Space Science,* vol. 27, 1979.

[24] G. Kockarts, «Neutral Atmosphere Modeling,» *Atmospheres of Earth and the Planets pp 235-243,* vol. 51, 1974.

[25] P. Bernhardt, J. Zinn, M. Mendillo et J. Baumgardner, «Modification of the upper atmosphere with chemicals found in rocket exhaust,» *20th Aerospace Sciences Meeting,* 1982.

[26] G. W. Bowden, P. Lorrain et M. Brown, «Numerical Simulation of Ionospheric Depletions Resulting From Rocket Launches Using a General Circulation Model,» *Journal of Geophysical Research: Space Physics,* vol. 125, 2020.

[27] National Aeronautics and Space Administration, «Rocket Thrust Summary».*https://www.grc.nasa.gov/www/k-12/rocket/rktthsum.html.*

[28] S. J. Isakowitz, J. B. Hopkins et J. P. Hopkins Jr., «International Reference Guide to Space Launch Systems,» *AIAA,* 2004.

[29] A. Zak, «Angara-5 to replace Proton».*http://www.russianspaceweb.com/angara5.html.*

[30] W. Graham, «Russia's Angara 5 debuts with maiden launch,» *https://www.nasaspaceflight.com/2014/12/russias-angara-5-maiden-voyage/,* 2014.

[31] E. Kyle, «Space Launch Report Archive».*http://spacelaunchreport.com/archive.html.*

[32] C. Bergin, C. Gebhardt, P. Sloss, W. Graham, P. Harding, R. C. Barbosa, M. Baylor, T. Burghardt, I. Atkinson, J. Davenport, T. Gray et T. Corbett, «NASASpaceflight.com».*https://www.nasaspaceflight.com/.*

[33] G. D. Krebs, «Gunter's Space Page».*https://space.skyrocket.de/.*

[34] E. Kyle, «New launchers - Angara».*http://spacelaunchreport.com/angara.html.*

[35] Orbital Sciences Corporation, «Taurus II User's Manual,» 2009.

[36] W. Graham, «Orbital's Antares launches Cygnus on debut mission to ISS,» *https://www.nasaspaceflight.com/2013/09/orbitals-antares-loft-cygnus-debut-mission-iss/ ,* 2013.

[37] E. Kyle, «Antares (Taurus II)».*http://spacelaunchreport.com/taurus2.html.*

[38] P. Blau, «NASA report on Antares launch failure places blames on AJ26 engines,» *https://spaceflight101.com/nasa-report-on-antares-launch-failure-places-blame-on-aj26-engines/,* 2015.

[39] Orbital ATK, Propulsion Products Catalog, 2016.

[40] W. Graham, «Antares 230 successfully returns with launch of OA-5 Cygnus,» *https://www.nasaspaceflight.com/2016/10/antares-230-launch-oa-5-cygnus/,* 2016.

[41] Northrop Grumman, «Antares User's Guide,» 2018.

[42] Arianespace, Ariane 5 User's Manual, October 2016.

[43] Arianespace, «Press Kits».*https://www.arianespace.com/press-kits/.*

[44] D. Boury, A. Esnault, J. Montheillet, N. Cesco, C. Talbot et E. Robert, «Large Solid Propulsion for European Building Blocks Launcher,» *AIAA,* 8-11 July 2007.

[45] D. Capdevilla, «Ariane 5,» *Capcom Espace,* vol. http://www.capcomespace.net/dossiers/espace_europeen/ariane/ariane5/caracteristiques.htm.

[46] Arianespace, «VA 219 ATV Georges Lemaître,» *https://www.arianespace.com/wp-content/uploads/2015/09/VA219-launchkit-EN1.pdf,* 2014.

[47] E. Kyle, «Space Launch Report: Ariane 5 Data Sheet».*http://spacelaunchreport.com/ariane5.html.*

[48] Arianespace, «Helios 2B GB,» *https://www.arianespace.com/wp-content/uploads/2017/05/HELIOS_2B_GB.pdf,* 2009.

[49] United Launch Alliance (ULA), «Atlas V Launch Services User's Guide,» March 2010.

[50] Northop Grumman, «Propulsion Products Catalog,» 2016.

[51] China Academy of Launch Vehicle, LM-2C User's Manual, 1999.

[52] Norbert Brügge, CZ-2C/D, http://www.b14643.de/Spacerockets_1/China/CZ-2/Description/Frame.htm.

[53] B. Harvey, China in Space - The Great Leap Forward, Second edition, 2019.

[54] Baidu Baike, Long March 2, https://baike.baidu.com/item/%E9%95%BF%E5%BE%81%E4%BA%8C%E5%8F%B7/9561816?fromtitle=%E9%95%BF%E5%BE%81%E4%BA%8C%E5%8F%B7%E8%BF%90%E8%BD%BD%E7%81%AB%E7%AE%AD&fromid=9022342#2_7.

[55] China Great Wall Industry Corporation, LM-2C, http://www.cgwic.com/LaunchServices/LaunchVehicle/LM2C.html.

[56] E. Kyle, Space Launch Report: Chang Zheng Data Sheet, http://spacelaunchreport.com/cz.html.

[57] Wikipedia the free encyclopedia, «Yuanzheng».*https://en.wikipedia.org/wiki/Yuanzheng.*

[58] China Great Wall Industry Corporation, LM-2D, http://www.cgwic.com/LaunchServices/LaunchVehicle/LM2D.html.

[59] SciNews, Long March-2D launches SuperView-1 03/04 satellites (高景一号, GaoJing-1 03/04), https://www.youtube.com/watch?v=xB_KV6eJxrQ&ab_channel=SciNews, 2018.

[60] Federal Aviation Administration, «The Annual Compendium of Commercial Space Transportation: 2018,» 2018.

[61] Yantar, «Chang-Zheng 2F/G».*Destination Orbite, Les Lanceurs Spatiaux dans le Monde.*

[62] Yantar, «Chang-Zheng 2F/T».*https://lanceurs.destination-orbite.net/chang_zheng_2f_t.php.*

[63] G. D. Krebs, «CZ-2E/-F».*http://www.b14643.de/Spacerockets_1/China/CZ-2EF/Description/Frame.htm.*

[64] China Academy of Launch Vehicle Technology, China Great Wall Industry Corporation et China Satellite Launch & Tracking Control General, LM-3A Series Launch Vehicle User's Manual Issue 2011.

[65] N. Brügge, CZ-3, http://www.b14643.de/Spacerockets_1/China/CZ-3/Description/Frame.htm.

[66] P. Blau, Long March 3B/E Launch Vehicle, https://spaceflight101.com/spacerockets/long-march-3be/.

[67] N. Brügge, CZ-4, http://www.b14643.de/Spacerockets_1/China/CZ-4/Description/Frame.htm.

[68] SciNews, Long March-4 launches Gaofen-1 02, Gaofen-1 03 and Gaofen-1 04, Launch of Gaofen 1-02, 1-03 and 1-04, 2018.

[69] R. C. Barbosa, «China conducts Long March 5 maiden launch,» *https://www.nasaspaceflight.com/2016/11/china-long-march-5-maiden-launch/,* 2016.

[70] S. Clark, «China launches Long March 5, one of the world's most powerful rockets,» *https://spaceflightnow.com/2016/11/03/china-launches-long-march-5-one-of-the-worlds-most-powerful-rockets/,* 2016.

[71] Norbert Brügge, «CZ-5».*http://www.b14643.de/Spacerockets_1/China/CZ-5/Description/Frame.htm.*

[72] P. Blau, «Long March 5 Launch Vehicle».*https://spaceflight101.com/spacerockets/long-march-5/.*

[73] P. Blau, «Long March 6 Launch Vehicle».*https://spaceflight101.com/spacerockets/long-march-6/.*

[74] Shanghai Academy of Spaceflight Technology (SAST), «LM-6».*http://www.sast.net/product/14.html.*

[75] Norbert Brügge, «CZ-6».*http://www.b14643.de/Spacerockets_1/China/CZ-6/Description/Frame.htm.*

[76] R. C. Barbosa, «Long March 6 launches Jilin-1 trio,» *https://www.nasaspaceflight.com/2017/11/long-march-6-launches-jilin-1-trio/,* 2017.

[77] E. Kyle, «Space Launch Report: CZ-5-7 Data Sheet».*http://spacelaunchreport.com/cz5.html.*

[78] P. Blau, «Long March 7 Launch Vehicle».*https://spaceflight101.com/spacerockets/long-march-7/.*

[79] Arabic@people.cn, «China's first cargo spaceship ready for launch,» *http://arabic.people.com.cn/n3/2017/0419/c31657-9205106-2.html,* 2017.

[80] N. Brügge, «CZ-7».*http://www.b14643.de/Spacerockets_1/China/CZ-7/Description/Frame.htm.*

[81] GalacticPenguinSST, «Long March 7 Maiden Flight - T - 1 minute to upper stage separation».*https://www.youtube.com/watch?v=alFgWFVKc04&ab_channel=GalacticPenguinSST.*

[82] R. C. Barbosa, «China successfully debuts Long March 7 - Recovers capsule».

[83] United Launch Alliance (ULA), «Delta II Payload Planners Guide,» 2006.

[84] E. Kyle, «Space Launch Report: Delta II Data Sheet».*http://spacelaunchreport.com/delta2.html.*

[85] United Launch Alliance (ULA), Delta IV Launch Services User's Guide, June 2013.

[86] Aerojet Rocketdyne, RS-68A, https://www.rocket.com/space/liquid-engines/rs-68a.

[87] AllCentralFlorida, ULA Delta IV GPS IIF-9 launch, https://www.youtube.com/watch?app=desktop&v=C9ZYl3BFedM&ab_channel=AllCentralFlorida, 2015.

[88] N. Brügge, «Dnepr».*http://www.b14643.de/Spacerockets_1/East_Europe_3/Dnepr/Description/Frame.htm.*

[89] W. Graham et C. Bergin, «Russia's Dnepr rocket launches Kompsat-3A mission,» *https://www.nasaspaceflight.com/2015/03/russias-dnepr-rocket-kompsat-3a-mission/#more-39725,* 2015.

[90] International Space Company Kosmotras, «Space Launch System DNEPR User's Guide,» 2001.

[91] E. Kyle, «Space Launch Report: Dnepr Data Sheet».*http://spacelaunchreport.com/dnepr.html.*

[92] Rocket Lab, «Payload User's Guide,» 2019.

[93] Rocket Lab, «Make It Rain Press Kit December 2018,» *Launching on electron vehicle four: 'This one's for pickering',* 2018.

[94] E. Kyle, «Space Launch Report: Rocket Lab Electron Data Sheet».*http://spacelaunchreport.com/electron.html.*

[95] Rocket Lab, USA, «Rocket Lab,» *https://www.rocketlabusa.com/,* 2019.

[96] Japan Aerospace Exploration Agency (JAXA), «Epsilon Launch Vehicle User's Manual,» 2018.

[97] W. Graham, «Epsilon rocket launches Japanese ERG mission,» *https://www.nasaspaceflight.com/2016/12/epsilon-2-rocket-launch-japanese-erg-mission/,* 2016.

[98] E. Kyle, «Space Launch Report: Epsilon Data Sheet».*http://spacelaunchreport.com/epsilon.html.*

[99] S. Tokudome, H. Habu, K. Ui, F. Shimizu, Y. Yachi et N. Sano, «Solid Propulsion Systems for Epsilon Launch Vehicle,» *AIAA,* 2012.

[100] SpaceX, «Falcon 1 Launch Vehicle Payload User's Guide Rev.7,» 2008.

[101] E. Kyle, «Space Launch Report: Falcon Data Sheet».*http://spacelaunchreport.com/falcon.html.*

[102] Ed Kyle, Space Launch Report: SpaceX Falcon 9 v1.2 Data Sheet, http://spacelaunchreport.com/falcon9ft.html.

[103] N. Brügge, Falcon-9, http://www.b14643.de/Spacerockets_2/United_States_1/Falcon-9/Description/Frame.htm.

[104] P. Blau, Falcon 9 FT (Falcon 9 v1.2), https://spaceflight101.com/spacerockets/falcon-9-ft/.

[105] SpaceX, «CRS-8 Dragon Hosted Webcast,» https://www.youtube.com/watch?v=7pUAydjne5M&ab_channel=SpaceX.

[106] shahar603, «Telemetry-Data».*https://github.com/shahar603/Telemetry-Data.*

[107] J. B. Bjarnø, Fra ELV til RLV, GO SpaceX !, https://ing.dk/blog/fra-elv-til-rlv-go-spacex-174114, 2015.

[108] C. Gebhardt, «Falcon 9 Block 4 debut a success, Dragon arrives for Station berthing,» *https://www.nasaspaceflight.com/2017/08/falcon-9-block-4-debut-success-dragon-station-berthing/,* 2017.

[109] S. Clark, «Launch timeline for Falcon Heavy's maiden flight,» *https://spaceflightnow.com/2018/02/06/launch-timeline-for-falcon-heavys-maiden-flight/,* 2018.

[110] Indian Space Research Organisation (ISRO), «GSLV-FO8/GSAT-6A,» *Launch Brochure,* 2018.

[111] Wikipedia the free encyclopedia, «Satish Dhawan Space Centre».*https://en.wikipedia.org/wiki/Satish_Dhawan_Space_Centre.*

[112] L. De Luca, T. Shimada, V. P. Sinditskii et M. Calabro, «Chemical Rocket Propulsion,» 2017.

[113] R. Nagappa, «Evolution of Solid Propellant Rockets in India,» 2014.

[114] Indian Space Research Organisation (ISRO), «GSLV MkIII-D1/GSAT-19 Mission,» *Launch Brochure,* 2017.

[115] Indian Space Research Organisation (ISRO), «LVM3 (Archived)».*https://web.archive.org/web/20141225031155/http://www.isro.gov.in/launchers/lvm3.*

[116] N. Brügge, «LVM3 (GSLV Mk.3)».*http://www.b14643.de/Spacerockets_1/India/LVM3/Description/Frame.htm.*

[117] MHI Launch Services, «H-IIA User's Manual,» 2015.

[118] Japan Aerospace Exploration Agency (JAXA), «H-IIA Launch Vehicle».*https://global.jaxa.jp/projects/rockets/h2a/.*

[119] Y. Fukushima, H. Nakatzuki, R. Nagao, K. Kishimoto, K. Hasegawa, T. Koganezawa et S. Warashina, «Development Status of LE-7A and LE-5B engines for H-IIA family,» *Acta Astronautica,* vol. 50, 2002.

[120] N. Sano, S. Kinai et T. Yamada, «Development of Advanced Solid Rocket Booster (SRB-A) for H-II A Rocket,» *Journal of the Japan Society for Aeronautics and Astronautics,* 1998.

[121] J. Horikawa, T. Maemura, K. Nimura, T. Goto et T. Nara, «Return to flight and the future of H-IIA,» *AIAA,* 2005.

[122] Mitsubishi Heavy Industries, «MHI Launch Services: Launch Vehicles».*https://www.mhi.com/products/space/launch_srv_lineup.html.*

[123] Japan Aerospace Exploration Agency (JAXA), «About H-IIB Launch Vehicle».*https://global.jaxa.jp/projects/rockets/h2b/index.html.*

[124] Orbital Sciences Corporation, «Minotaur I User's Guide,» 2006.

[125] ATK, «ATK Space Propulsion Products Catalog,» 2008.

[126] E. Kyle, «Space Launch Report Minotaur 4 Data Sheet».*http://spacelaunchreport.com/mintaur4.html.*

[127] W. Graham, «Minotaur IV launches first Space Based Space Surveillance satellite,» *https://www.nasaspaceflight.com/2010/09/live-minotaur-first-space-based-space-surveillance-satellite/,* 2010.

[128] C. Bergin, «Orbital ATK Minotaur-C launches SkySat mission out of Vandenberg,» *https://www.nasaspaceflight.com/2017/10/orbital-atk-minotaur-c-skysat-mission-vandenberg/,* 2017.

[129] E. Kyle, «Space Launch Report Taurus Data Sheet».*https://www.spacelaunchreport.com/taurus.html.*

[130] Starsem, Soyuz User's Manual, 2001.

[131] E. Kyle, «Space Launch Report: R-7/Soyuz Data Sheet».*http://spacelaunchreport.com/soyuz.html.*

[132] E. Kyle, «Space Launch Report: New Launchers - KSLV».

[133] Korea Aerospace Research Institute (KARI), «Naro, the First Korea Space Launch Vehicle».*https://www.kari.re.kr/eng/sub03_03_02.do#link.*

[134] R. C. Barbosa, «South Korea launch of KSLV-1 - Russians claim it failed».*https://www.nasaspaceflight.com/2009/08/south-korea-launch-of-kslv-1/.*

[135] Korea Aerospace Research Institute (KARI), «Naro Space Center».*https://www.kari.re.kr/eng/sub03_03_04.do.*

[136] W. Graham, «Orbital's Pegasus XL successfully lofts IRIS spacecraft,» *https://www.nasaspaceflight.com/2013/06/orbital-pegasus-launch-iris-spacecraft/,* 2013.

[137] Orbital Sciences Corporation, «Pegasus User's Guide,» 2007.

[138] N. Brügge, «Proton».*http://www.b14643.de/Spacerockets_1/East_Europe_2/Proton/Description/Frame.htm.*

[139] International Launch Services (ILS), «Proton Launch System Mission Planner's Guide,» 2004.

[140] ILS, Proton Launch System Mission Planner's Guide, 2009.

[141] ILS, «Proton Breeze M- Heritage Launch System with 97 Commercial Missions to date».*https://www.ilslaunch.com/launch-vehicle/proton-breeze-m/.*

[142] A. Zak, «Third time is a charm for Block DM-03».*http://www.russianspaceweb.com/block-dm03.html.*

[143] N. Brügge, «Proton-M».*http://www.b14643.de/Spacerockets_1/East_Europe_2/Proton-M/Description/Frame.htm.*

[144] Indian Space Research Organisation (ISRO) et Antrix Corporation Limited, «PSLV-C29 TeLEOS-1,» *PSLV-C29/TeLEOS-1 Brochure,* 2015.

[145] Doordarshan National, «Launch of PSLV-C23 - Live on 30 June 2014,» *https://www.youtube.com/watch?v=UR5S5nNbd4U,* 2014.

[146] E. Kyle, «PSLV».*http://spacelaunchreport.com/pslv.html.*

[147] Indian Space Research Organisation (ISRO), «PSLV-C35 Scatsat-1,» *PSLV-C35/Scatsat-1 Brochure,* 2016.

[148] Indian Space Research Organisation (ISRO), «PSLV-C27 IRNSS-1D,» 2015.

[149] Doordarshan National, «The Launch of PSLV-C30/ASTROSAT Mission Satellite from SDSC, SHAR, Sriharikota - LIVE,» *https://www.youtube.com/watch?v=hZCRiNpfB-8,* 2015.

[150] Eurockot Launch Service Provider, «Rockot User's Guide,» 2011.

[151] E. Kyle, «Space Launch Report: Rokot/Strela».*http://spacelaunchreport.com/rokot.html.*

[152] P. Blau, «Rockot Launch Vehicle».*https://spaceflight101.com/spacerockets/rockot/.*

[153] C. P. Vick, «Safir 1, 2 Booster - Iran First Satellite Booster».*https://www.globalsecurity.org/space/world/iran/safir-1-2.htm.*

[154] E. Kyle, «Space Launch Report: Safir/Qased Data Sheet».*http://spacelaunchreport.com/safir.html.*

[155] Wikipedia the free encyclopedia, «Iranian Space Agency».*https://en.wikipedia.org/wiki/Iranian_Space_Agency#Space_centers.*

[156] D. Wright, «A Model of a 2-Stage Safir Launch Vehicle,» *Union of Concerned Scientist - Citizens and Scientists for Environmental Solutions,* 2009.

[157] P. Blau, «Israel Launches Advanced Optical Reconnaissance Satellite,» *https://spaceflight101.com/israel-launches-advanced-optical-reconnaissance-satellite/,* 2016.

[158] N. Brügge, «Shavit».*http://www.b14643.de/Spacerockets_1/Rest_World/Shavit/Description/Frame.htm.*

[159] E. Kyle, «Space Launch Report: Shavit».*http://spacelaunchreport.com/shavit.html.*

[160] Wikipedia the free encyclopedia, «Palmachim Airbase».*https://en.wikipedia.org/wiki/Palmachim_Airbase.*

[161] Роскосмос ТВ, «Союз МС». 9 минут до космоса, https://www.youtube.com/watch?v=UNlglX8IiFo&ab_channel=%D0%A0%D0%BE%D1%81%D0%BA%D0%BE%D1%81%D0%BC%D0%BE%D1%81%D0%A2%D0%92.

[162] A. Zak, «Fregat upper stage».*http://www.russianspaceweb.com/fregat.html.*

[163] A. Zak, «Volga upper stage».*http://www.russianspaceweb.com/volga.html.*

[164] Arianespace, Soyuz User's Manual Issue 2 Revision, March 2012.

[165] Starsem, «RD-0124 An optimized propulsion system for Soyuz/ST,» *Presentation in Versailles,* 2002.

[166] N. Brügge, «Soyuz-1 (2.1v)».*http://www.b14643.de/Spacerockets_1/East_Europe_1/Soyuz-1/Description/Frame.htm.*

[167] National Aeronautics and Space Administration NASA, «Space Shuttle».*https://www.nasa.gov/returntoflight/system/system_SSME.html.*

[168] M. Cook et L. Gruet, «Characterization of Space Shuttle Reusable Solid Rocket Motor Static Test Stand Thrust Measurements,» *Aerospace Research Central,* 2003.

[169] Thiokol Corporation Space Operations, «Space Shuttle Development Motor No. 9 (DM-9) Final Test Report Volume 1,» *Prepared for National Aeronautics and Space Administration,* 1990.

[170] Spacevidcast (now TMRO -- see new channel), «STS-133 The Final Launch of Space Shuttle Discovery including T-5 hold,» *https://www.youtube.com/watch?v=Hy1Nc4YpAUI,* 2011.

[171] Y. Inatani et H. Ohtsuka, «SS-520 Nano satellite launcher and its flight result,» *32nd Annual AIAA/USU Conference on Small Satellites,* 2018.

[172] E. Kyle, «Space Launch Report: SS-520 Data Sheet».*http://spacelaunchreport.com/ss520.html.*

[173] W. Graham, «Japanese sounding rocket claims record-breaking orbital launch,» *https://www.nasaspaceflight.com/2018/02/japanese-rocket-record-borbital-launch/,* 2018.

[174] Japan Aerospace Exploration Agency (JAXA), «S-310/S-520/SS-520 (Sounding Rockets)».*https://global.jaxa.jp/projects/rockets/s_rockets/.*

[175] P. Blau, «Japan's SS-520 CubeSat Launch Vehicle Achieves Success on Second Try,» *https://spaceflight101.com/japan-ss-520-5-launch-success/,* 2018.

[176] P. Blau, «Strela Launch Vehicle».*https://spaceflight101.com/spacerockets/strela/.*

[177] N. Brügge, «Strela (UR-100N)».*http://www.b14643.de/Spacerockets_1/East_Europe_3/Strela/Description/Frame.htm.*

[178] N. Brügge, «Tsiklon (R-36)».*http://www.b14643.de/Spacerockets_1/East_Europe_2/Tsiklon/Description/Frame.htm.*

[179] E. Kyle, «Space Launch Report: Tsyklon Data Sheet».*http://spacelaunchreport.com/tsyklon.html.*

[180] N. Brügge, «Unha 2».*http://www.b14643.de/Spacerockets_1/Rest_World/Unha-2/Description/Frame.htm.*

[181] Wikipedia the Free Encyclopedia, «Unha».*https://en.wikipedia.org/wiki/Unha.*

[182] E. Kyle, «Space Launch Report: Unha Data Sheet».*http://spacelaunchreport.com/unha.html.*

[183] D. Wright et T. A. Postol, «A post-launch examination of the Unha-2,» *https://thebulletin.org/2009/06/a-post-launch-examination-of-the-unha-2/,* 2009.

[184] Arianespace, «Vega User's Manual Issue 4 Revision 0,» 2014.

[185] M. Fiorillo, F. Giliberti, M. Angelone, C. Milana et F. Serraglia, «VEGA Solid Rocket Motors Flight Performances results,» *American Institute of Aeronautics and Astronautics,* 2013.

[186] A. Neri, S. Bianchi, P. Pascal, M. Cutroni, «An overview of Vega Solid Rocket Motors development and qualification program,» *American Institute of Aeronautics and Astronautics,* 2003.

[187] Sea Launch Company, «Land Launch User's Guide,» 2004.

[188] N. Brügge, «Zenit (Land Launch)».*http://www.b14643.de/Spacerockets_1/East_Europe_2/Zenit-LL/Description/Frame.htm.*

[189] Sea Launch Limited Partnership, «Sea Launch User's Guide,» 2000.