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Master's Degree in Aerospace Engineering

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From Production to Launch: a Comprehensive Life Cycle Assessment Applied to the Real Case Study of Ariane 6

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Abstract

This thesis analyses the biggest issues and gaps that affect current LCA (Life Cycle Assessment) practices in the space sector. It addresses how the difficulty in collecting data affects the results of the studies, indicating how an information-sharing framework, supported by the right set of laws and acts, could benefit all users. Future changes affecting launcher architectures, and therefore LCI (Life Cycle Inventory) databases, are also discussed.

The document then defines the steps that need to be included when determining the impact due to the testing and propellant combustion of a launcher. Moreover, it includes the modelling, based on data collection from available off-the-shelf products, of the main components usually found in a launch vehicle. The modelled components are then used to create and analyse the Ariane 6 launcher, with the goal of defining the environmental hotspots and eventual life cycle phases with a negligible impact. It can be concluded that most of the impacts, for which the manufacturing and launch phases are mostly responsible, affect the GWP (Global Warming Potential) and rare minerals resource use categories. Large structures, avionics and propellant production are the most impactful processes, and therefore the ones on which sustainability efforts should focus.

Lastly, it can be observed how architectures with more boosters, whose overall impact is higher, are actually the more sustainable choice when normalising the results per kilogram of payload placed into orbit: it is therefore less impactful to use a bigger launch vehicle with a higher payload capacity than multiple, smaller launchers.

A Nonna Maria Laura, nella speranza che i tuoi genietti matematici mi possano seguire ovunque io vada.

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Acronyms

ADEPLmu	Abiotic Resource Depletion.
AFAs	Applications for Authorisation.
AGV	Autonomous Guided Vehicle.
APU	Auxiliary Propulsion Unit.
ATR	Average Temperature Response.
C&DH	Command and Data Handling.
CEA	Chemical Equilibrium Applications.
CFRP	Carbon Fibre Reinforced Polymer.
CLH	Harmonised Classification and Labelling.
CLP	Classification, Labelling and Packaging.
CMFU	Centralized Multi-Functional Unit.
CNC	Computer Numerical Control.
CoRAP	Community Rolling Action Plan.
DC-DC	Direct Current - Direct Current.
ECHA	European Chemicals Agency.
EDM	Electrical Discharge Machining.
ELV	Expendable Launch Vehicle.
EMI	Electromagnetic Interference.
ERF	(Effective) Radiative Forcing.
ESA	European Space Agency.
EU	European Union.
EUSPA	European Union Agency for the Space Programme.
EX	Expander.
GFRP	Glass Reinforced Carbon Polymer.
GG	Gas Generator.
GHG	Greenhouse Gases.
GN&C	Guidance, Navigation and Control.

GTP	Global Temperature-Change Potential.
GWP	Global Warming Potential.
IAM	Integrated Assessment Model.
IC	Integrated Circuit.
INU	Inertial Navigation Unit.
IPCC	Intergovernmental Panel on Climate Change.
ISAC	Information Sharing and Analysis Centre.
ISRO	Indian Space Research Organisation.
LCA	Life Cycle Assessment.
LCI	Life Cycle Inventory.
LEAF	Large European Acoustic Facility.
LEO	Low Earth Orbit.
LVM3	Geosynchronous Satellite Launch Vehicle Mk III.
NASA	National Aeronautics and Space Administration.
OBC	On-Board Computer.
PCDU	Power Conditioning and Distribution Unit.
PDU	Power Distribution Unit.
PFAs	Per- and Polyfluoroalkyl Substances.
PFU	Pyrotechnic Firing Unit.
PWB	Printed Wiring Board.
REACH	Regulation on the Registration, Evaluation, Authorisation and Restriction of Chemicals.
RLV	Reusable Launch Vehicle.
R&D	Research and Development.
RTM	Resin Transfer Moulding.
SC	Staged Combustion.
SLS	Space Launch System.
SMU	Satellite Management Unit.
SSDR	Solid State Data Recorders.
SVHC	Substance of Very High Concern.
TIG	Tungsten Inert Gas.
TVC	Thrust Vector Control.

ULA	United Launch Alliance.
VVT	Verification, Validation and Testing.

Chapter 1

Introduction

1.1 Motivation

As the number of rocket launches continues to increase, although their emissions comprise only a small fraction of the total associated to the transport segment, it is important to understand and quantify their environmental impact on the planet [1]. Satellites, which are carried into space by launch vehicles, enable the monitoring of more than half of the so-called "Essential Climate Variables", thus playing an important role in the development of informed and sustainable decision-making [2, 3]. Since it is unlikely that the amount of spacecraft sent into space will undergo a significant decrease in the near future, it is mandatory to assess not only the impacts due to their production and use, but also those of the supporting segments, such as the launch one, which is responsible for a considerable portion of the emissions of a space mission [4].

In light of this new sustainable framework, a discipline called "Ecodesign" has been developed, in which a product is designed taking into account the emissions that characterise the entirety of its life cycle [3]; the goal is to influence early design choices, when it is easier to modify negative environmental impacts and thus develop a more sustainable end product [3, 2]. The Ecodesign approach has been implemented by ESA (European Space Agency) [5], whose launchers and other space products need to adhere to both environmental standards and regulations, such as the REACH (Regulation on the Registration, Evaluation, Authorisation and Restriction of Chemicals) and the Green Deal, and ESA's guidelines and standards, such as the Green Agenda [1]. ESA's Green Agenda aims to lower the agency GHG (Greenhouse Gases) emissions, 69% of which are due to the development, manufacturing and use of their own space products [5]. The use of tools such as the LCA (Life Cycle Assessment) thus facilitates the quantification of such impacts, and helps in defining which areas contribute the most and, therefore, on which mitigation efforts should focus.

The aim of this document is therefore to develop ready-made processes using secondary data, and to define life cycle steps whose impact is negligible in a first, high-level LCA; the end goal is to fill the research gaps that currently affect LCA studies in the space sector, streamlining the beginning of the design process and removing the need to characterise each component taken into consideration, therefore avoiding having to change the product later on, which is often impossible after qualifying the initial design [2].

1.2 Topic Overview

An LCA is a standardised tool that allows the calculation of the environmental impact of a certain product throughout its life cycle [4, 6]. Although LCAs and environmental analyses have already been established in most sectors for decades, the space industry lagged behind due to its initial exclusion from protocols, such as the Montreal one (1987) [3]. ESA only started working on the inclusion of LCA in the design process in 2009, and they soon reached the conclusion that none of the available databases could accurately picture the complexity of space products [3]. Launchers and satellites require specific materials and manufacturing processes, and are produced in small amounts [6, 2]. Launchers pollute directly into the stratosphere, and satellites create debris once no longer in use [2, 1]. It is for these reasons that LCAs in the space sector need to be adapted to these unique challenges [6], eventually implementing new and space specific indicators, but, at the same time, remaining in line with other sectors to enable comparisons [3].

1.3 Objectives

The main objectives of this thesis are to suggests some improvements to enhance the accuracy of both inputs and outputs in launch vehicle LCAs, and to research and implement the most common launcher components. These components are then used for a real case study, whose goal is to understand which life stages constitute an environmental hotspot, and to identify eventual phases whose impacts are negligible.

1.4 Thesis Outline

Chapter 1 introduces the thesis topic and motivation, which are then discussed more in depth in Chapter 2. Chapter 3 carries out the testing, propellant combustion and launch vehicle component implementation, which are used in Chapter 4 to perform the Ariane 6 LCA. Chapter 5 presents the results of the analysis. Lastly, Chapter 6 underlines the most important achievements of the work, alongside some suggestions for future related works.

Chapter 2

State of the Art

The following chapter will present a literature review of the current state of the art of LCAs in the space sector, focusing on the main issues affecting the practice, and suggesting possible solutions. It will also introduce some future changes affecting rocket architectures, which will lead to some additions in LCA databases and new considerations on the sustainability of launch vehicles.

2.1 Specific Data & Data Collection

Launchers and all space products are characterized by the use of specific materials [2, 7, 8, 9] and advanced manufacturing processes [2, 8, 9]. This means that normal products, available in databases such as EcoInvent [10], usually fail to capture the complexity and consequent energy consumption that characterize the development, testing and production of space products [9]. While some specific data is available, a substantial problem for LCAs in the space sector is data collection difficulty [2]. Data can be either primary or secondary: primary data is taken directly from production sites, while secondary data comes from indirect sources, such as databases or literature [9]. While primary data is often preferred, as it gives the most accurate results, it is also hard to obtain, as values for a singular product are rarely available, with yearly or factory-wide data often being the only available option [9].

Another problem afflicting both primary and secondary data is confidentiality [9, 11], which makes it hard to access and compare specific processes between companies [11] and, in the long term, could lead to the development of conflicting LCA methodologies that lower the credibility of the results and of the LCA method itself [8]. The consequences of the lack of organized and coordinated information sharing will only get worse with the increasing number of space missions and projects [12], thus requiring better cooperation between governments and space agencies in regulating information sharing.

Different companies might not want to cooperate with one another, even while having a common goal, for a set of reasons: first, by sharing their unique information and/or technology, they might feel like they are losing their competitive advantage and power [13]; secondly, even if openly collaborating, they might enter a mixed competitive-collaborative mindset, which could lead to partially withholding information from competitors or even lying to them [13]; lastly, studies show how a high level of trust is needed between different parties to actually share information and cooperate to reach the common goal [13].

It is clear how difficult it is to get companies to openly share their data without fear of damaging their own interests. For this reason, different regulations have been proposed that would strongly encourage companies to participate in data collection, setting an equal field for everyone involved and thus finally allowing to move towards the common goal of correctly characterizing materials and processes in an LCA.

A first regulation regarding the matter is the “European Data Governance Act” from the European Commission, whose goal is to increase trust in data sharing, strengthen mechanisms to increase data availability and overcome technical obstacles to the reuse of data [14]. Although not specific to the space sector, its purpose lies in making more data accessible and facilitate data sharing across EU (European Union) countries [14], which would surely help in the realisation of a complete and comprehensive LCA tool. A key point of the legislation is about letting companies maintain their competitive advantages while still being able to share their data: they will do so through data intermediaries, that will function as neutral third parties that connect individuals and companies with data users [14]. The European Commission is also encouraging voluntary data sharing for a common good [14]. Although preliminary results from the first year of application of the regulation are not available, it is surely a huge step forward in lowering the difficulty of accessing the specific data that characterizes the space sector.

A complementary regulation to the Data Governance Act is the “Data Act”, from the European Commission as well, which ensures fairness in the allocation of the value of data amongst the actors in the data economy and clarifies who can use what data and under which conditions [15]. One of its key points is guaranteeing access to performance data of industrial equipment [15], thus potentially helping achieve a better characterization of manufacturing processes involved in the space sector.

These two legislations give a great insight into how important and valuable information sharing is, especially when conducted in a cooperative rather than competitive environment. There is currently no specific regulation for space companies that encourages or forces them to share their data, but space agencies are quickly realising how important collaboration is in this sector; for example, the European Commission and EUSPA (European Union Agency for the Space Programme) are jointly developing the so called ISAC (Information Sharing and Analysis Centre), whose goal will be to "provide a safe environment for companies to voluntarily share information on a specific security situation or to solicit guidance on security implementation and governance for space applications" [16]. Even though its main objective is related to improving cyber security measures in space companies in EU state members [16], it once again shows how regulated data sharing can contribute to better results for all parties involved.

This developing information sharing framework should thus be further implemented by different actors, from space agencies (like ESA) to governing bodies (like the European Commission), in order to increase the overall benefits in having access to valid information.

2.1.1 Regionalisation

An issue related to the difficulty of having access to specific data is the lack of regionalisation of products and processes in current LCA databases; however, it is extremely important when working with secondary data in LCAs. Correctly accounting for emissions due to one country's energy mix or local transportation emissions allows for more accurate and precise results [11]. For example, the Guianese[1] energy mix is made up of 47% renewable energy [7], leading to a lower impact compared to a nation that still fully relies on fossil energy. Given its importance, regional data has been implemented, where possible, in the current ESA LCA database [17], which is here considered as the most complete example available. In Tab. 2.1, the current implementation of regionalisation of products can be observed.

It is clear how, in most cases, the data is generalized for the whole European region, if not for the whole world. This, as underlined before, can lead to significant changes in the end result of an LCA, and should gradually be improved to include more country-specific data. While it is true that this only refers to ESA's database, implementing products also related to other regions of the world could lead to a better application of the LCA practice throughout the whole sector, allowing for a complete study and comparison of different products. Being able to correctly specify how much different countries or regions are polluting, however, requires access to data from industries located on those territories, once again leading to the problem of data collection addressed in Subsec. 2.1. It is reasonable to expect that, if the suggestions in the

[1]The French Guiana launch complex is where Ariane rockets, among others, are launched.

Table 2.1: Regionalisation of Products in ESA's LCI (Life Cycle Inventory) database [17].

PRODUCT	GLOBAL	EUROPE ONLY	EU COUNTRIES	FOREGROUND DATA
Material Groups		X		
Chemicals		X		X
Propellants		X		
Electrical Components	X			
Energy			X	
Transport		X		
Process Group		X		
Propellant Handling	X			X
Ground Segment	X	X		
Manpower & Travels			X	
Launch Segment	Missing Data			
Space Segment	X	X		
Testing & Inspections		X		
Missions	Missing Data			
Waste Treatment	X			

aforementioned section are followed, the regionalisation of product should then be fairly easy to accomplish and implement.

2.2 Material Obsolescence and the Use of Prospective LCA Tools

Launchers are characterised by a long development cycle [8, 9], lasting 5 to 10 years [2], and an even longer life cycle that can last up to 30 years [2]. For these reasons, missions need to be designed using long-term planning and strategic thinking [18], which include being able to take into account how things will change from the beginning of a programme to its very end, while performing the LCA in the design phase. The challenge in doing so lies in having to predict how material and energy policies will evolve in a long period of time. The energy evolution problem has already been tackled by [19] through the implementation of their Premise Tool: its working principle lies in incorporating Integrated Assessment Model (IAM) prospective scenarios in the EcoInvent database, whose purpose is to evaluate expected transformations within five major energy-intensive sectors, namely power generation, cement and steel production, freight and passenger road transportation, and supply of conventional and alternative fuels [19]. While not specifically designed to work with databases such as ESA's [17], these fields are most definitely present in LCA space specific studies, making the tool a useful resource when planning for future missions. The tool is in continuous evolution, with new fields being explored and developed, and further collaborations could either expand the fields it is able to cover, or lead to the creation of similar tools specifically for the space sector.

The biggest problem in developing a new launcher and dealing with long-term issues, however, lies in the choice of materials to use, as they could undergo material obsolescence long before the predicted end of the programme [7], thus posing the choice of whether to retire the launcher early (which implies economic losses and impacts due to design and infrastructure being split over a smaller number of launches) or substitute the problem materials with new, equivalent ones (which require further design, testing and manufacturing, in addition to eventual changes in performance). That is, however, if companies employ a reactive approach to material obsolescence, rather than dealing with it in a proactive way from the first stages of development. It has been proven that proactive approaches lead to having to spend less time and materials to fix the problem [20, 21]: in order to apply them, it is important to accurately determine the risk of a certain component or material obsolescence [20].

Products can incur two types of obsolescence: regulatory obsolescence arises when substances are legally banned, while commercial obsolescence arises when a product that was formerly employed in the space industry stops being produced by suppliers as larger sectors stop using it [2]. Regulatory obsolescence can be at least partially avoided through resources such as the REACH tool, which is the main EU law to protect human health and the environment from the risks that can be posed by chemicals [22], and it is the strictest law to date regulating chemical substances [23]. Many substances currently in use in the space sector are under observation by REACH, and it was reported that up to 20% of materials commonly used in the sector could potentially be affected in the long term [2], leading to space industries having to substitute them through lengthy and costly processes [23]. In Tab. 2.2, a list of substances that are commonly used in the space sector and might undergo material obsolescence due to REACH regulations can be found.

The "Classification" column refers to which list the substance currently belongs to. The different possibilities are:

- Classification, Labelling and Packaging (CLP): notification to inform about possible hazards on substance labels [24].
- Community Rolling Action Plan (CoRAP): a European Chemicals Agency (ECHA) member state has evaluated or will evaluate the substance over the coming years [24].
- Substance of Very High Concern (SVHC): substances that may have serious and often irreversible effects on human health and the environment [24].
- Candidate List: includes SVHCs. Suppliers of substances in the Candidate List are required to provide a safety data sheet, safe use guidelines and notify ECHA if the article they produce contains an SVHC in quantities above one tonne per producer/importer per year, and if the substance is present in those articles above a concentration of 0.1% (w/w) [24].

- Annex XIV: Authorisation List. Companies that want to continue using a substance included in the Authorisation List after the sunset date need to prepare an application for authorisation [24].
- Annex XVII: Restriction List. Includes substances whose use is restricted or prohibited in the EU market [24].

The "Risk Level" column refers to an index, going from 0 to 7, which indicates how likely a substance is to be restricted or banned (and therefore undergo material obsolescence) in the near future. The indexes have the following meanings [25]:

- 0 = substance has no known regulatory risks.
- 1 = substance has a hazard code in Harmonised Classification and Labelling (CLH).
- 2 = substance meets SVHC hazard threshold but not is not in the Candidate List.
- 3 = substance is an SVHC and has entered the Candidate List.
- 4 = substance has either been prioritised for inclusion in the Authorisation List (Annex XIV), or contains Per- and Polyfluoroalkyl Substances (PFAs) which do not have higher regulatory limitations yet.
- 5 = substance is in the Authorisation List (Annex XIV), but either pre-sunset date or targets space-irrelevant use in the Restriction List (Annex XVII).
- 6 = substance is in the Authorisation List (Annex XIV), but there are some Applications for Authorisation (AFAs) for specific use of the substance, or the substance has an indirectly space relevant entry in the Restriction List (Annex XVII).
- 7 = substance is in the Authorisation List (Annex XIV) but no AFAs have been received for use of the substance, or the substance has an entry in the Restriction List (Annex XVII) that bans its use relevant to space industry.

PFAs could be completely banned from any type of use in the near future [23]. However, this would pose serious challenges to the space sector: PFAs are present in lubricants, coatings, creep barriers, functionalised polymer surfaces, cleaning agents, coolant fluids, sealants, pyrotechnic compositions, thermal insulation, adhesives, fire suppressants and electronic assemblies [25]. A complete ban on PFAs would greatly damage the aerospace industry, as it is hard to find

Table 2.2: Possible REACH Affected Substances

SUBSTANCE	ISSUES	CLASSIFICATION	RISK LEVEL	POSSIBLE ALTERNATIVE
Surface Treatment - Chromic Acid [26, 27]	Carcinogenic, Skin Sensitising [24]	SVHC, Candidate List, Annex XIV [24]	6	Tartaric Sulphuric Acid, Phosphoric Sulphuric Acid, Boric Sulfuric Acid [28, 29]
Surface Treatment - Chromium Trioxide [27]	Carcinogenic, Mutagenic, Skin Sensitising, Toxic to Reproduction, Respiratory Sensitising	SVHC, Candidate List, Annex XIV [24]	6	Surtec 650 [28]
Nitric Acid [26]	Corrosive to Metals and Skin [24]	CLP for Handling [24]	1	Citric Acid [26]
Pyrotechnic Initiator Powders - MIRA (Ammonium Dichromate) [26, 30]	Carcinogenic, Mutagenic, Toxic to Reproduction, Skin Sensitising, Respiratory Sensitising [24]	SVHC, Candidate List, Annex XIV, Annex XVII [24]	7	Zirconium Potassium Perchlorate [30]
Hydrazine [23]	Carcinogenic, Skin Sensitising [24]	SVHC, Candidate List [24]	3	ASCENT (Advanced Spacecraft Energetic Non-Toxic) Propellant, LMP-103S Green Propellant [31, 32, 33]
Bisphenol A [23]	Toxic to Reproduction, Skin Sensitising, Endocrine Disrupting [24]	CoRap, SVHC, Candidate List, Annex XVII (Not Aerospace Related) [24]	5	Depends on Application
Bisphenol F [23]	Skin Sensitising [24]	CLP Notification [24]	1	Depends on Application
Toluene [23]	Carcinogenic, Mutagenic, Toxic to Reproduction [24]	CoRAP, Annex XVII [24]	6/7	Pinacolone [34]

replacements with the same level of performance and validation [35]. For this reason, multiple aerospace corporations and national space agencies gathered to discuss the effects of a possible ban on PFAs, reaching the conclusion that the environmental impact of PFAs used in the space sector is negligible compared to other sectors, as its production volumes are small and the PFA-containing components are sent to space and then burned in the atmosphere, thus not posing any risk to human health [35]. Due to the complexity of this topic and its possible implications, no risk level or possible alternatives were explored; however, it remains of utmost importance to find viable and less impactful alternatives to PFAs.

Although ESA's analysis method description reminds the user to check if any of the selected substances are affected by REACH [17], a more straightforward approach could facilitate the process. In Fig. 2.1, an example for "Nitric Acid" can be found.

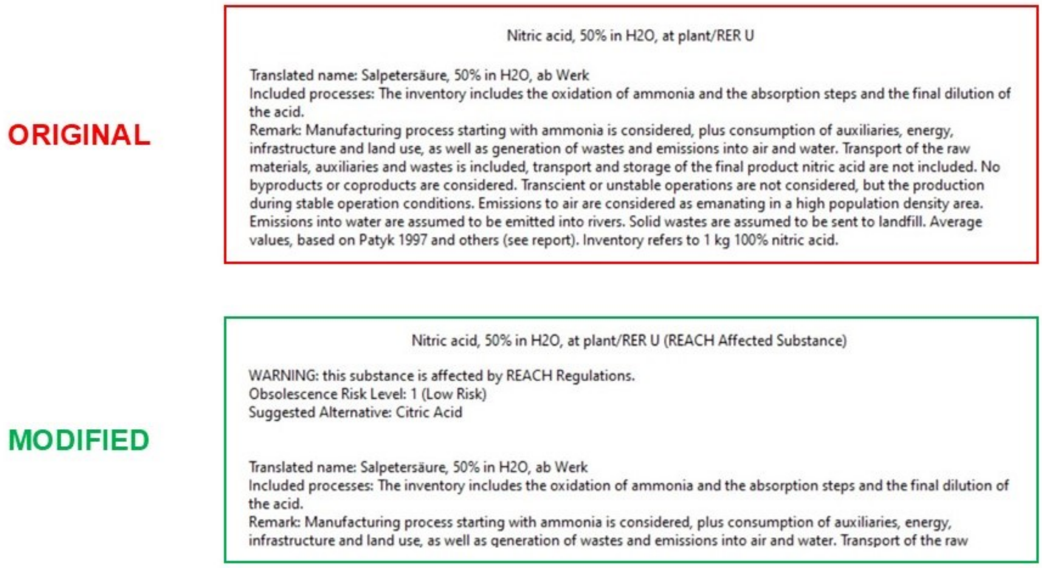

Figure 2.1: Example of REACH Affected Substance Banner [17]

Adding a banner both in the name and description of affected substances makes it easier for users to spot them. The banner also includes a definition of the level of risk (low if the index goes from 0 to 2, medium-low if it goes from 3 to 4, medium-high if it goes from 5 to 6 and high if it is 7 [25]) and suggestions on possible alternatives with lower or no risk level.

Commercial obsolescence has not yet found direct applications in LCA tools, but there are many different algorithms that can be employed to accurately understand how the situation will change during the life cycle of a product. Although the main focus is currently on electronic components [21, 20] the same algorithms can be used for other parts specific to the space sector. The different methodologies that can be used to determine when a specific material or component will become commercially obsolete are:

- Regression Analysis [21, 20]: gathers data on the product in question and related variables to predict its behaviour on the market [36].
- Time Series Analysis or Trend Analysis [21]: identify patterns, trends and irregularities in the data over different periods to determine future behaviours [37].
- Exponential Smoothing [21]: uses an exponentially weighted average of past observations to predict future values. It assigns more weight to recent observations and less to older observations, allowing the forecast to adapt to changing trends in the data [38].

These algorithms could be further improved by using artificial intelligence and machine learning [20], and it is not too far-fetched to think that they could deliver results similar to the ones of the Premise tool [19] that can be directly implemented in LCA evaluation tools. These methods would first need to identify the possible evolution over the launcher's lifetime of a specific material or component, and then modify the database taking it into account, thus underlining which components and materials could potentially become critical and would be better to replace while still in the design phase.

Although this implementation has not yet been taken into account, these algorithms are valuable tools to correctly identify which materials to use when developing a new product, and could serve as a starting point for future predictive LCA evolutions.

2.3 Addressing Heritage

When talking about technical products, heritage can be defined as "proven components that are being modified to meet new requirements" [39]. The question of how to properly address it in LCAs [9] stems from the fact that their development and testing has already been carried out, and their relative impacts can no longer be taken into account when developing, for example, a new launcher within a family [11]; however, further developments and integration with new systems need to be considered [11]. To properly identify the extent of inherited impacts of a component, the same methodologies employed to assess heritage-related risks can be applied. The approach suggested by [39] is based on two fundamental aspects of an inherited product: the degree of VVT (Verification, Validation and Testing) needed, and the degree to which the design, suppliers, and manufacturing processes have changed [39]. VVT is directly related to the amount of modifications operated on the component, while the readiness of the manufacturing chain depends on if and when the dedicated establishment was shut down [39]. For each of the two, a scale from 0 to 1 has been defined (Tab. 2.3).

Table 2.3: Modification & Manufacturing Heritage Scores [39]

SCORE	MODIFICATIONS	MANUFACTURING
0	No Significant Modification	Running Line
0.5	Significant Modifications	Running Line Recently Shut Down
1	Component Developed from Scratch	Running Line Shut Down for Several Years

The overall score can therefore range from 0 to 1; if a component has a score of 0, it means that both testing and manufacturing do not need significant changes with respect to its predecessor, and thus its development can be neglected when performing an LCA. In all other cases, however, either due to significant changes to the component itself or to the production line, the impact of the new component needs to be assessed.

2.4 Future Sustainable Launchers

The main changes that will take place in the forthcoming years regarding launchers concern their reusability and the use of green propellants.

2.4.1 Reusability

The introduction of reusable launchers has led to high recycling rates of first stages [40], which can lower the amount of resources needed for their production and the ocean pollution due to the disposal of expendable stages [41]; however, it is important to investigate and quantify eventual additional impacts deriving from the features that make them reusable [40].

Firstly, reusable stages need to survive the re-entry: this might require new thermal protection systems, usually not implemented on an ELV (Expendable Launch Vehicle), and critical components might need to be more robust in order to survive multiple launches [41]; moreover, in case of landing, launchers need additional landing gear. Re-entry emissions for first stages can be considered negligible, as they typically re-enter at speeds much lower than those of upper stages or capsules [40]; this assumption might need to be revisited in the future should single-stage-to-orbit reusable vehicles be produced, or high altitude emissions be better characterised, and it surely does not stand true for re-entering higher stages [40].

Secondly, once re-entered, the reusable stages need to be recovered: this can happen either by landing the stage itself on a platform, or through recovery operations that involve boats or planes [40]. All these methods consume propellant, and therefore lead to additional emissions; however, it is not yet clear whether these impacts are compensated from the reuse of the stage or not [40]. Reusable stages also need to be inspected and refurbished before they are ready to fly again, which leads to additional impacts and the manufacturing of eventual components that need to be replaced [41]. However, similarly to the recovery operations, it is not clear if the recovery and refurbishment of the launcher lead to and overall reduced impact [41].

It is estimated that, depending on the choice of propellant, there is a reduction of 20 to 40% of the impacts for reusable launch fleets compared to their expendable counterparts [41]. This seemingly positive result might be offset by two secondary effects of reusable launchers: firstly, they lower the cost of space access, leading to an increase of the number of launches and, therefore, to more emissions related to the launch phase [40]. Moreover, due to the additional components required to make a launcher reusable, the payload capacity is reduced, which again leads to an increased amount of launches to carry the same payload as ELVs into orbit [40]. Nonetheless, it has been demonstrated that even after two or three reuses, if no major component substitution is required, the additional production impacts of a RLV (Reusable Launch Vehicle) are already compensated in terms of material depletion and ocean pollution [41].

Lastly, reusable launchers and architectures similar to the Space Shuttle need to perform drop and landing tests during the testing campaign [42]. These are especially useful to test the landing gear, which is usually missing in ELVs, but require an additional use of resources, mainly in the form of propellant [42].

It is reasonable to conclude that further studies need to be conducted to determine if reusable launchers truly mitigate environmental impacts, and are therefore a valid solution on which the space sector should focus.

2.4.2 Green Propellants

The production and use of propellants make up a sensible part of the impacts related to a launch vehicle life cycle [40, 11, 7], as they comprise most of its weight and need a high level of purity [7]. Investing in the production and development of greener fuel can thus help offset the overall environmental impact of a launcher [11, 4], and can help make reusable systems the most sustainable option, as the additional propellant needed with respect to ELVs would be produced in a less impactful way [4]. Most current propellant production facilities could easily adapt to green propellant production, making the space industry a leader in the sector [4]; however, the cost of developing and maintaining new hardware, together with the increased risk of failure due to inexperience, could make the change slow [43]. Additional costs could also arise from the testing of new materials and technologies [43].

Two examples of greener fuels that are currently being tested or already implemented are:

- Carbon neutral fuels [41]: already implemented in the aviation sector [41], their goal is to compensate the CO_2 emitted during exhaust by capturing a similar amount from the atmosphere while being produced [40]. They include both biofuels (produced using bio-waste, agricultural waste or wood), and fuels synthesised using CO_2 collected through direct air capture [40]. However, they still create stratospheric black carbon, NO_x and water vapour while being burned, thus creating pollutants that are not counterbalanced by a greener production line [41]: it is estimated that, during combustion, the GWP (Global Warming Potential) of carbon neutral methane was reduced only by 10% with respect to its conventional counterpart, and that carbon neutral kerosene's GWP decreased only by 2-3% [40]. Moreover, options such as biofuels require massive land use, for which the space sector would need to compete with other industries [40].

- Green hydrogen: can be produced through wind energy, which could lead to a lowered production carbon footprint of an order of magnitude [41], or through solar-powered electrolysis of water, which can reduce the impact even more, up to five times less than traditional hydrogen production [44]. However, using new techniques could also lead to a shift of the impacts onto other categories, such as mineral resource depletion [41].

The low amount of possible green alternatives, together with the problems that might arise from their production and use, is due to the fact that the industry main goal is to find a green propellant that meets the requirements in terms of I_{sp}, while being environmentally sustainable and non-toxic [43]. While not straightforward to understand, toxicity directly affects the sustainability of a fuel: the level of toxicity is directly proportional to the to the storage and handling costs, as a spill would impact both ecosystems and human health, but safety precautions tend to increase operational costs while decreasing the overall sustainability of the product [43]. Toxic propellants also need decontamination and waste treatment steps, which further increase their environmental impacts [7]. Moreover, fuels that need to be stored for a long time required increased storage resources, maintenance and monitoring, and increase the risk of wasting fuel [43].

All of these requirements make finding a suitable green propellant extremely hard. It can therefore be concluded that, although some alternatives are already being developed and soon will be put to use, it will take more time to develop a product that is sustainable in every aspect of its life cycle, while also respecting performance requirements.

2.5 Current Research Gaps

The consequences of the issues highlighted in Sec. 2.1 and 2.2 are three main research gaps that affect current LCAs analyses performed on space products and, more specifically, on launch vehicles.

The first gap concerns the exclusion of testing from most studies, although its impact is non negligible in certain impact categories [9]. Most common testing processes have already been implemented in databases such as ESA's one [17], but the lack of clear boundaries on what needs to be included, together with the difficulty of assessing the overall impact, which cannot be assigned to a single launch, results in testing not being accounted for in LCAs [11].

The other main research gaps concern the lack of ready-made LCI database processes for both the combustion of propellants and launch vehicle components. While propellant production has been thoroughly discussed and developed in ESA's database [17], there are no processes which include the products of combustion due to the use of the propellants. Although the development of such processes would need to rely on some assumptions that might not represent the entire spectrum of possibilities, it would allow for a fast comparison in the design phase of a launcher, leading to the choice of the most sustainable alternative.

Lastly, ESA's database includes a huge variety of components to model a satellite and some ground segment activities, but it still lacks the main subsystems and components that usually make up a launcher [17]. As previously mentioned for propellant combustion, the availability of ready-made processes, even if simplified, would allow for a fast and easy comparison of possible alternatives, both between rocket architectures and different materials, thus helping to make more informed and sustainable choices while developing a new product.

Chapter 3

Methodology

The goal of this chapter is to introduce the biggest gaps that affect current launch vehicle LCA studies: the inclusion of testing in LCAs, and the lack of propellant combustion products and vehicle components in databases. While the first section will mainly deal with outlining the most important steps of a typical testing program, the remaining sections will introduce new LCI processes to include both propellant combustion emissions and metrics and launch vehicle components in databases.

3.1 Testing

Launchers are subject to extensive and rigorous testing [11, 6], the environmental impacts of which, even if smaller than the ones due to production and use of the product [11], need to be taken into account when performing an LCA to not significantly bias its results [11]. The difference in testing-related impacts compared to non-space products is due to the low production rates, long development cycles and use of specialised materials [8], whose operational environment is extreme and therefore requires long testing and qualification steps to comply with space industry standards [7]. It is important to note that, although testing will never cease to be an energy-consuming step, the use of renewable rather than fossil-based energy, along with an optimised facility management, could greatly lower the impacts related to this phase [7].

In the following subsections, the general testing steps a launcher goes through are presented [42], alongside a potential implementation, wherever possible, of testing practises in LCAs. It is important to note that only tests regarding whole subsystems or the entire launcher were considered, while component-level testing (such as destructive tests for components subjected to fatigue [42]) was not taken into account.

3.1.1 Thrust Measuring Tests

To measure the thrust developed by a rocket engine, a thrust meter (or thrust gauge) is used [42]. Its main components are a test stand (the main structure, used to hold the engine), a load cell (scale) and a computer for data acquisition [42]. While the load cell and computer are likely to remain the same for different engines, the test stand might need adjustments to properly fit around each one. However, due to the high number of performed tests [45, 46, 47, 48], the impact of the production of this component can be considered negligible.

The main impacts of this type of test are therefore due to the burned propellant and the water used in water deluge systems. Propellants are already implemented in databases, and their specific amount can be considered from case to case, rather than creating a generic process (which would not be representative of reality due to the huge range of propellants and mass of propellant used by different engines). The impacts due to the combustion of propellant can be found in Sec. 3.2. The water employed in water deluge systems is mainly sourced from natural bodies [49, 50], and can thus be modelled in LCAs through the "Water, unspecified natural origin" process [10]. Since the amount of water can greatly vary based on the scale and timing of the test taking place [49, 50], a generic model would not accurately represent specific situations; it is therefore better to consider specific data for a test rather than create an inaccurate model. A real case example can be found in Chapter 4.

3.1.2 Pressure Vessel Tests

Pressure vessel tests are carried out in order to verify that propellant or pressurizer tanks are able to withstand the harsh conditions they will have to experience [42]. The different types of testing they might be subjected to are, not necessarily in order:

- Burst Test: the vessel is pressurized until it bursts [42]. The impact of this type of testing is modelled in ESA's database under "Pressure Cycle inspection" [17], and it should be taken into account alongside the possible loss of material, should the vessel burst.

- Vacuum Test: extremely low pressure is created inside the vessel, with atmospheric pressure acting on the outside of the tank [42]. The impact of this type of testing is modelled in ESA's database under "Negative Pressure inspection" [17], and it should be taken into account alongside the possible loss of material, should the vessel implode.

- Drop Test: the tank is dropped from a high altitude before going through the burst and vacuum tests. The only relevant impact of this type of test is the eventual loss of material, should the vessel break.

- Chemical Compatibility Test: the vessel is filled with propellant to test eventual chemical reactions with the tank material [42]. Propellants can be computed from ESA's database [17], alongside the possible loss of the material, should a destructive chemical reaction happen.

- Impulse Response Testing: the tank is impacted with a known force and the resulting vibrations are measured. The energy needed to carry out this test can be considered negligible. Actual vibration tests can be modelled through the "Vibration test" process from ESA's database [17].

- Non-Destructive Testing: since vessels are expensive to build, it is better to limit the amount of tests that could lead to their destruction and exploit non-destructive evaluations whenever possible [42]. Examples of such tests include: x-rays ("X-ray/radiographic inspection" [17]), acoustic tests ("Acoustic [LEAF]" [17]) and ultrasonic tests ("Ultrasonic Inspection" [17]) [42].

3.1.3 Shake and Bake Tests

The shake test is used to test possible destructive vibrational modes of the rocket, while the bake test (which actually involves exposing the rocket to both high and low temperatures) is carried out to ensure the launcher can handle big temperature changes [42].

- Vibration Test: can be modelled through ESA's database "Vibration Test" [17].

- Bake Test: both hot and cold environments can be modelled through "Thermal Vacuum Chamber, LSS, Sun Simulation ON" and "Thermal Vacuum Chamber, LSS, Sun Simulation OFF" from ESA's database [17].

3.1.4 Environment Tests

Environment tests consist in subjecting the rocket to extreme environments that simulate the real ones, before the actual flight test [42]. Example of facilities employed for such tests are the aforementioned thermal vacuum chambers and wind tunnels [42]. The average amount of energy used by a supersonic wind tunnel is about 50 MW per square meter of test section [51], which can be modelled in SimaPro if the dimensions of the used facility are known.

3.1.5 Modelling and Simulation

Using software modelling and simulation tools reduces the number of physical tests needed, although they cannot replace them entirely as models need verification and refining [42]. To account for this type of testing in LCAs, the model "Operation, computer, desktop, office use" [10] can be taken into account. Its impacts are regionalised, meaning the results are accurate and allow for a fast comparison between activities carried out in different countries [11]. Although the amount of hours spent on this task is rarely publicly available, a company developing a rocket could either use data from past projects, or estimate the amount of hours needed based on the size of the project and existing guidelines.

3.1.6 Roll-Out Test

The roll-out test consists in testing the transport of the vehicle from the assembly building to the launch pad [42]; its environmental impact is therefore comparable to the impact of actual roll-out of a launcher, an example of which is available in Subsec. 4.4.3.

3.1.7 Flight Test

The flight test consists in flying the rocket in its true operational environment, rather than simulated ones [42]; its environmental impact is therefore comparable to the impact of an actual launch event (see Sec. 3.2).

3.2 Sector Specific Pollution

Evaluating emissions related to a launch vehicle use phase is not straightforward, as they pollute differently from any other known products [2]: they emit pollutants throughout all layers of the atmosphere [52], and we generally lack correct models for high altitude and high speed emissions [53, 1]. The aim of this section is to characterise the main emissions from launcher engines and introduce eventual new impact indicators that are specific only to the space sector. It is important to note that some other examples of space specific pollution include the creation of space debris or the use of extraterrestrial resources [7]; however, they do not figure in the cradle-to-use part of the LCA and, as such, will not be discussed.

One of the problems in determining the overall impact of a given fuel is that it might be cleaner than other choices at a certain altitude, but become way more impacting at another altitude [1]. The evolution of emissions must therefore be examined throughout all phases of the launch. Emissions in the troposphere can be generally considered negligible, as they cause no long-term impact [2], while emissions in the stratosphere need to be well characterised, as they are more dangerous. This is due to the fact that the stratosphere hosts the ozone layer and is dynamically isolated from the troposphere, meaning that pollutants emitted at high altitudes might remain there for several years [2]; moreover, gases in the stratosphere undergo faster horizontal mixing, spreading emissions through the layer and making them more long lasting [54].

The main effects that need to be studied are those on global warming and ozone depletion, as their metrics are still to be determined for many rocket-specific emissions [11]. It is also necessary to define the main products of the propellant combustion, in order to correctly model a process for the launch phase of a vehicle.

3.2.1 Global Warming

When considering a rocket launch, warming due to common greenhouse gases such as H_2O and CO_2 is overshadowed by the amount produced by other sectors [2], making their effects negligible [52]. One should therefore focus on particles such as black carbon and alumina (Al_2O_3), which are respectively responsible for 70% and 28% of the offsetting of the radiative balance of the atmosphere [52]. Black carbon particles accumulate in the stratosphere [2] and absorb incident short wave solar radiation [52], which results in the warming of the stratosphere [2]. Alumina particles have a more complex behaviour, both warming the stratosphere through the reflection of incoming radiation into space and the absorption of upwelling long wave radiation from the Earth, and cooling the troposphere and the ground due to the accumulation of particles in the stratosphere that reduces the solar flux [2, 52]. In order to quantify their impact, it is first necessary to determine the amount of particles produced by the most commonly used propellants. Tab. 3.1 shows the amount of black carbon and alumina produced by different fuels in terms of mass of pollutant per mass of burned fuel [52].

Table 3.1: Black Carbon Emissions for Most Common Fuel Types [52]

FUEL	BLACK CARBON EMISSIONS	ALUMINA EMISSIONS
RP-1	35 g/kg	-
HTPB	35 g/kg	-
Hypergolic (MMH, UDMH) [11]	4 g/kg	-
Solid	4 g/kg	380 g/kg

ESA's global warming potential analysis is conducted using the GWP100 (Global Warming Potential of a substance compared to that of CO_2 over a 100-year span) method [17]; however, since both black carbon and alumina are short-lived climate forcers [52] which have a lifetime of a few months to two decades [11], the GWP100 method might not produce the most accurate results [11]. Even if shorter time periods were considered, for example by selecting the GWP20 method, the full scale effects of these particles would not be correctly characterised: the choice in time scale (20, 100 or 500 year period [55]) should only be dictated by the time period over which one wants to reach its goals [55], not the life duration of the substances taken into account. The application of the GWP method is based on the assumption that the emissions being compared produce radiative forcing that is evenly spread across the globe [55]; however, rocket emissions are not distributed homogenously around the globe, and can therefore cool certain regions while warming others [2]. This means that the GWP values for both black carbon and alumina, regardless of the chosen time scale, might not actually represent a CO_2 equivalent [55]. Additional metrics that are specific to the space sector, such as ERF ((Effective) Radiative Forcing), GTP (Global Temperature-Change Potential), and ATR (Average Temperature Response) [11], can be added to the traditional ones to better quantify the effects of short-lived climate forcers. Radiative Forcing is defined as the radiative response to the forcing agent, but does not account for rapid tropospheric adjustments; the inclusion of such adjustments is often called Effective Radiative Forcing [56]. The Global Temperature-Change Potential is the ratio of temperature change from a pulse emission of a climate species to a pulse emission of carbon dioxide [55], while the Average Temperature Response is the global mean temperature change caused by operation of a particular aerospace vehicle, based on the radiative forcing generated by each emission species [57]. In Tab. 3.2, the values for the aforementioned additional metrics for black carbon can be found. No values have yet been defined for alumina particles, as their effect on radiation balance are yet to be completely understood [58].

Table 3.2: Additional Metrics for Black Carbon

	GTP20	**GTP100**	**(E)RF**	**ATR**
BLACK CARBON	470 [55]	64 [55]	0.11 W/m^2 [59]	$1.16 \cdot 10^{-11}$ °C/kg [60]

ESA's analysis method [17] can be modified and expanded to take into account these additional metrics. In order to perform a more accurate analysis, the values for other, more well known pollutants can also be computed. In Tab. 3.3, the values of the additional metrics can be found, if available, for all the main species emitted during combustion in a rocket engine [58]. Substances that make up less than 0.01% in weight of the emitted species[1] have not been

[1]Species not evaluated: H_2O_2, AlCl, $AlCl_2$, $AlCl_3$, AlOH, $AlOHCl_2$, $Al(OH)_2Cl$ [58].

taken into account.

Table 3.3: Additional Metrics for Most Common Rocket Pollutants

	GTP100	**(E)RF**	**ATR**
CO	0.28 [61]	-	-
CO_2	1 [62]	2.16 W/m^2 [62]	-
H	-	-	-
HO_2	-	-	-
H_2	2 [63]	0.12 W/m^2 [64]	$1 \cdot 10^{-13}$ °C/kg [65]
H_2O	0.008[2]	0.07 W/m^2[1][66]	-
O	-	-	-
OH	-	-	-
O_2	-	-	-
NO	-2.5 [61]	-0.25 W/m^2 [59]	-
N_2	-	-	-
Cl	-	-	-
HCl	-	-	-

[1] Value relative to aviation emissions, effects in higher atmosphere still not characterised.
[2] Radiative Forcing value, not (E)RF.

It is clear how most pollutants still lack the necessary characterisation to apply these additional metrics in LCA practice. Although most of them are mentioned in IPCC (Intergovernmental Panel on Climate Change) reports [59, 62, 66] as contributors to global warming, their effects are mainly indirect, making it hard to quantify them and surely needing further research to better understand their behaviour. Wherever possible, they will be implemented in the modified ESA analysis method, alongside eventual additional ozone depleting substances and metrics that will be presented in the next subsection. The ATR index, due to the insufficient amount of values found, has not been implemented in the analysis method.

An example analysis of one kg of LO_x/RP-1, analysed with both ESA's original analysis method and the modified method, which includes the metrics for GTP and ERF, can be found in Fig. 3.1 and 3.2.

It is clear how the propellant combustion, if not correctly characterised, could be considered as responsible only for about 60% of the total GWP, while it is actually the main driver in the GTP category as well; moreover, the effects of the propellant production on the radiative balance of the Earth, which were initially discarded, proved to be the biggest contributors to the category, with only a small fraction of the (E)RF impact being due to the propellant combustion. Therefore, although many species still lack the correct environmental characterisation and thus pose some limitations to the accuracy and veracity of such results, metrics that are specific to the space sector shall

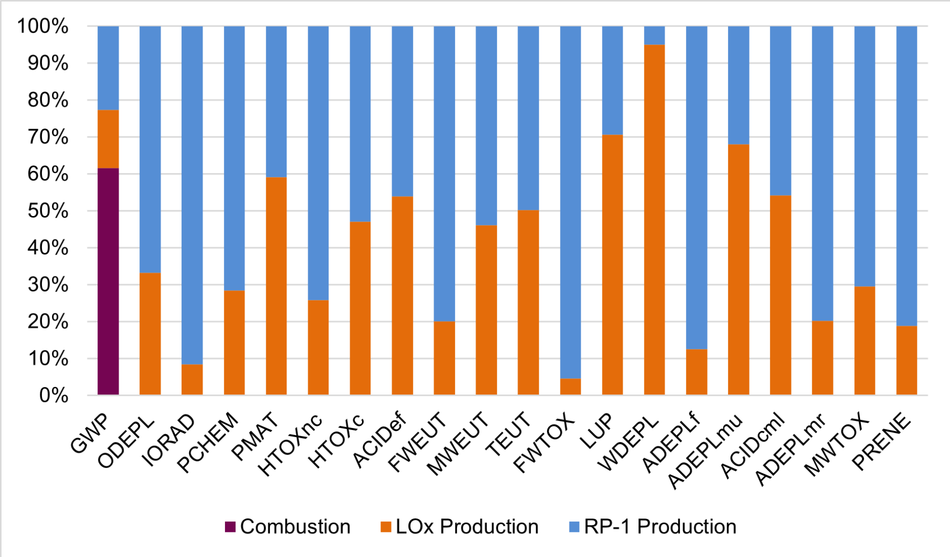

Figure 3.1: Impact Analysis of 1 kg of LO_x/RP-1 with Original ESA Method

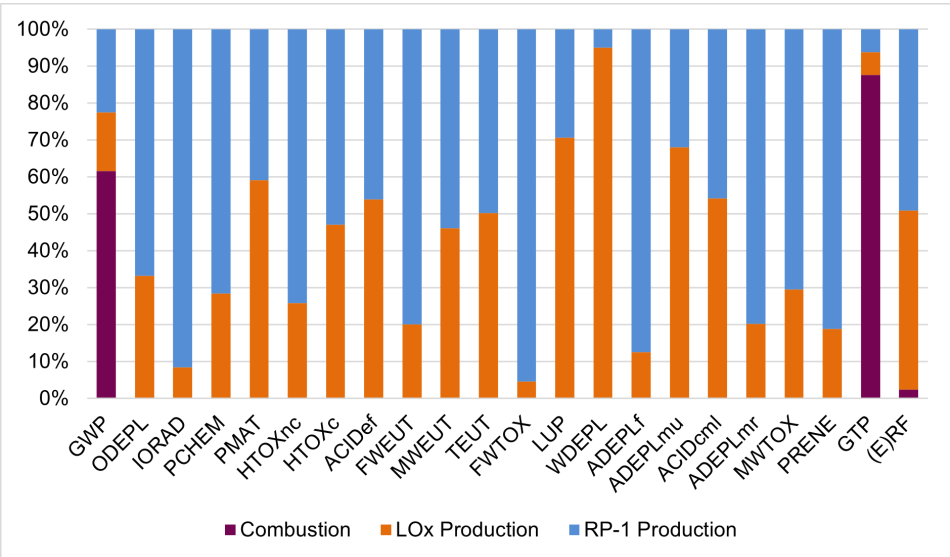

Figure 3.2: Impact Analysis of 1 kg of LO_x/RP-1 with Modified ESA Method

most definitely be investigated and developed to correctly analyse all possible affected categories.

3.2.2 Ozone Depletion

The use phase of a launch vehicle is responsible for nearly 100% of the ozone depletion registered during its life cycle [2, 58]. Ozone is mostly depleted by highly reactive radicals, such as chlorine, nitrogen, bromine, and hydrogen oxides [2]; the nitrogen, chlorine, hydrogen and oxygen cycles respectively account for 32%, 19%, 26% and 23% of the depletion, meaning that the impact is almost evenly distributed between different pollutants [58]. While all main radical containing substances have already been implemented in ESA's database [17], two different effects still need to be accounted for: the direct injection of water vapour in the stratosphere [52], and the enhanced Cl-activated ozone loss due to Alumina particles [52]. H_2O contributes to the formation of stratospheric polar clouds, which are largely responsible for Antarctic ozone destruction [52], while Al_2O_3 creates Cl radicals by reacting with chlorine-containing substances that are present in the stratosphere, leading to ozone depletion [52]. However, both processes have indirect effects, as they do not directly involve ozone depleting substances but rather facilitate the formation of such species. Since the mechanisms related to both H_2O and Al_2O_3 are complex, they still need to be studied in depth in order to associate an ozone depletion index to these indirect forcing agents.

3.2.3 Launch Vehicle Emissions

Aside from particle pollutants, rockets emit a variety of chemical species during combustion. The most common propellant combustions have been simulated using the NASA (National Aeronautics and Space Administration) CEA (Chemical Equilibrium Applications) tool [67], assuming typical mixture ratio values [58]. The applied assumptions are: shifting equilibrium, ambient pressure $p_0 = 70$ bar, pressure ratio $\frac{p_0}{p_e} = 70$ [58]. The combustion processes can be implemented in SimaPro by considering a mass of one kilogram of propellant and the subsequent mass fractions of the various pollutants, available in Tables 3.4, 3.5, 3.6, 3.7, 3.8, 3.9. For carbon containing propellants, an additional amount of black carbon is added (although this might lead to a sum of the mass fractions that is higher than one), based on Tab. 3.1.

Table 3.4: LO_x/LH_2 Emissions

LO_x/LH_2 - MR = 6.2 [58]	
LO_x Mass	0.8611 kg
LH_2 Mass	0.1389 kg
SPECIES	**MASS FRACTION**[1]
H	0.001 [67]
H_2	0.03034 [67]
H_2O	0.96878 [67]
OH	0.00077 [67]
O_2	0.00001 [67]

[1] At nozzle exit.

Table 3.5: LO_x/CH_4 Emissions

LO_x/CH_4 - MR = 3.8 [58]	
LO_x Mass	0.7917 kg
CH_4 Mass	0.2083 kg
SPECIES	**MASS FRACTION**[1]
CO	0.08443 [67]
CO_2	0.43886 [67]
H	0.00018 [67]
HO_2	0.00001 [67]
H_2	0.00238 [67]
H_2O	0.43730 [67]
O	0.00125 [67]
OH	0.01458 [67]
O_2	0.02101 [67]
BC	0.0155 [58]

[1] At nozzle exit.

Table 3.6: LO_x/RP-1 Emissions

LO_X/RP-1 - MR = 2.6 [58]	
LO_x Mass	0.7222 kg
RP-1 Mass	0.2778 kg
SPECIES	**MASS FRACTION**[1]
CO	0.30104 [67]
CO_2	0.40169 [67]
H	0.00015 [67]
H_2	0.00656 [67]
H_2O	0.28787 [67]
O	0.00006 [67]
OH	0.00237 [67]
O_2	0.00025 [67]
BC	0.035 [52]

[1] At nozzle exit.

Table 3.7: N_2O_4/CH_6N_2 Emissions

N_2O_4/CH_6N_2 (MMH) - MR = 2.05 [58]	
N_2O_4 Mass	0.6721 kg
CH_6N_2 Mass	0.3279 kg
SPECIES	**MASS FRACTION**[1]
CO	0.08568 [67]
CO_2	0.17858 [67]
H	0.00001 [67]
H_2	0.00666 [67]
H_2O	0.32502 [67]
NO	0.00001 [67]
N_2	0.40399 [67]
OH	0.00007 [67]
BC	0.004 [52]

[1] At nozzle exit.

Table 3.8: $N_2O_4/C_2H_8N_2$ Emissions

$N_2O_4/C_2H_8N_2$ (UDMH) - MR = 2.67 [58]	
N_2O_4 Mass	0.7275 kg
$C_2H_8N_2$ Mass	0.2725 kg
SPECIES	**MASS FRACTION**[1]
CO	0.00412 [67]
CO_2	0.25382 [67]
H	0.00001 [67]
H_2	0.00018 [67]
H_2O	0.31600 [67]
NO	0.00311 [67]
N_2	0.38572 [67]
O	0.00019 [67]
OH	0.00377 [67]
O_2	0.03308 [67]
BC	0.004 [52]

[1] At nozzle exit.

Table 3.9: NH_4ClO_4/Al/HTPB Emissions

NH_4ClO_4/Al/HTPB - MR = 69:19:12 [58]	
NH_4ClO_4 Mass	0.69 kg
Al Mass	0.19 kg
HTPB Mass	0.12 kg
SPECIES	**MASS FRACTION**[1]
CO	0.2189 [58]
CO_2	0.0148 [58]
Cl	0.0022 [58]
H	0.0058 [58]
HCl	0.016 [58]
H_2	0.3022 [58]
H_2O	0.1158 [58]
N_2	0.0826 [58]
OH	0.0004 [58]
Al_2O_3 *(s)*	0.0498 [58]
Al_2O_3 *(l)*	0.0474 [58]
BC	0.035 [52]

[1] At nozzle exit.

In Tab. 3.10, the values of the heat of reaction for the previously mentioned propellants combustion can be found. A positive value shall be interpreted as heat released into the environment.

Table 3.10: Heat of Reaction for Most Common Rocket Propellant Combustion

PROPELLANT	MIXTURE RATIO	HEAT [kJ/kg][1]
LO_x/LH_2	6.2	7946.23 [67]
LO_x/CH_4	3.8	5946.31 [67]
LO_x/RP-1	2.6	5123.13 [67]
N_2O_4/CH_6N_2 (MMH)	2.05	3683.69 [67]
N_2O_4/CH_8N_2 (UDMH)	2.67	3561.09 [67]
NH_4ClO_4/Al/HTPB	69:19:12	4,013.32 [68]

[1] At nozzle exit.

Five of the species that figure in the previous tables (H, OH, HO_2, O, Cl) do not appear in either ESA's database or EcoInvent as airborne emissions, and no impact characterisation index has yet been established, therefore meaning that they cannot be characterised and their eventual environmental impacts will be neglected in this analysis. Most of the other species have already been characterised, although they do not appear in all possible impact categories, either because they have no impact or because an index has yet to be established. However, a value for the GWP100 index of molecular hydrogen has been published (11.6 kg CO_2 eq [69]), and has thus been added to the analysis method.

3.3 Implementation of Launch Vehicle Components

Unlike components belonging to the space or ground segment, launcher vehicle segments are currently not implemented in LCA databases: ESA's LCI database [17], which can be considered one of the most complete resources available to date due to Europe's lead over other regions in applying LCAs to space products [7], does not contain launch vehicle components. Databases such as the Strathclyde Space System Database [70] still lack generalised components for this segment [40], thus making it difficult for companies to have access to data that, even if simplified, could lead to more sustainable choices in the first phases of product design. For this reason, this section deals with a rather simplified implementation of the main components of a launch vehicle, based on architectures that are currently in use. The following high level component identification is based on the scheme presented in [42]:

- Structure: airframe, environmental protection and support structures (modelled all together as the main structure of lower and upper stages), fairings, mechanisms

- Propulsion System: pumps, combustion chamber, igniters, nozzle, flow systems, engine structure
- Power System: storage, conditioning, transfer
- Communication System: radios, antennas
- Guidance, Navigation and Control System: thrust vectoring, control surfaces, navigation sensors, computers
- Command and Data Handling System: command computers, data processing, storage and distribution
- Emergency Systems: abort systems, fire control, redundant systems

The following subsections will deal with the implementation of the components of each subsystem: in order to develop generalised processes, data from available off-the-shelf products will be averaged. Each component will be modelled by considering the materials it is made of and the manufacturing processes it is subjected to. Materials will be chosen from the "Transformation" datasets, therefore taking into account raw material extraction emissions.

Two possible approximations for components that lack the necessary data to be implemented can be found in Chapter 4.

3.3.1 Structure

Rockets can be classified into different categories depending on the amount of payload they are able to deliver to different orbits [71]:

- Small Lift: up to 2,000 kg to LEO (Low Earth Orbit)
- Medium Lift: 2,000 to 20,000 kg to LEO
- Heavy Lift: 20,000 to 50,000 kg to LEO
- Super-Heavy Lift: 50,000 kg to LEO

The size and materials of rockets belonging to different categories might differ substantially; their structure is therefore analysed by dividing it into the aforementioned categories.

For each category, a maximum of three launchers has been considered; ideally, the number of vehicles to base the estimation on should be much greater, but, as mentioned in Subs. 2.1, specific, up-to-date data is difficult to obtain. The choice thus fell on launchers whose data was available through reliable sources and which can be considered good representations of their respective launcher category.

Whenever needed, the following mass estimation relations are used to approximate the mass of the avionics, wiring and thrust frame respectively [72]:

$$m_{avionics} = 10(M_0)^{0.361} \tag{3.1}$$

$$m_{wiring} = 1.058\sqrt{M_0}l^{0.25} \tag{3.2}$$

$$m_{thrust\ frame=2.55\cdot10^{-4}\times T} \tag{3.3}$$

where M_0 is the lift-off mass of the launch vehicle, l is its overall length, and T is the total stage thrust expressed in Newtons.

Small Lift Liquid Propellant Launch Vehicles

The two vehicles taken into account for this category are: Electron by RocketLab and Alpha by Firefly Aerospace [73, 74]. Configurations with more than two stages were not taken into account, as they are quite uncommon.

Lower Stage Structure

Both the Electron and Alpha rockets run on LO_x and RP-1 [73, 74] and, even though their architectures are different (Electron has a common tank bulkhead [73], while Alpha has an inter-tank [74]), their size and weight are comparable; they are also both made of carbon composite [73, 74]. In Tab. 3.11, the structural masses of the lower stage of both rockets (including CFRP (Carbon Fibre Reinforced Polymer) components only) can be found.

Table 3.11: Lower Stage Structural Mass of Small-Lift Launchers

ELECTRON	ALPHA	MEAN STRUCTURE MASS
950 kg [75]	492.5 kg [76, 77]	**721.25 kg**

Since ESA's database [17] already includes every necessary step for the production of a CFRP product, the process "CFRP, cured by autoclave process" can be used to completely model the production of a generic small-lift lower stage.

Interstage

Both Electron and Alpha's interstages are made of carbon composite. Assuming a thickness of 2 cm for Electron and of 1 cm for Alpha (due to its higher payload to structural mass ratio [77]), the estimated masses can be found in Tab. 3.12.

Table 3.12: Interstage Mass of Small-Lift Launchers

ELECTRON	ALPHA	MEAN INTERSTAGE MASS
355.88 kg [73]	345.38 kg [74]	**350.63 kg**

The same process used to model the lower stage can be applied to the interstage, by considering the mean interstage mass for the "CFRP, cured by autoclave process" [17] in SimaPro.

Upper Stage Structure

The same similarities and differences underlined for the lower stage apply to the upper stage as well. In Tab. 3.13, the structural masses of the upper stage of both rockets can be found. They are both made of CFRP, like the rest of the rocket [73, 74].

Table 3.13: Upper Stage Structural Mass of Small-Lift Launchers

ELECTRON	ALPHA	MEAN STRUCTURE MASS
465 kg [75]	154.64 kg[1]	**309.82 kg**

[1] Assuming the same structural mass to dry mass ratio as the lower stage [76].

Considering the mean upper stage structural mass for the ”CFRP, cured by autoclave process” [17], the generic small lift upper stage can be modelled.

Fairing

Alpha’s fairing, made of CFRP, is assumed to be cylindrical for the first two thirds of the height and conical for the last third, with a thickness of 0.009525 m [76]. Electron’s fairing mass is known. Tab. 3.14 shows their values and the mean amount of material considered for a generic fairing.

Table 3.14: Fairing Mass of Small-Lift Launchers

ELECTRON	ALPHA	MEAN FAIRING MASS
44 kg [73]	279.35 kg [74, 76]	**161.675 kg**

Considering the mean fairing mass for the ”CFRP, cured by autoclave process” [17], the generic small lift fairing can be modelled.

Medium Lift Liquid Propellant Launch Vehicles

The three vehicles taken into account for this category are: Atlas V (400 series) by ULA (United Launch Alliance), Soyuz-2 by RKK Progress, and the LVM3 (Geosynchronous Satellite Launch Vehicle Mk III) by ISRO (Indian Space Research Organisation).
These rockets are made of two liquid stages and eventual solid rocket boosters, which are implemented separately. Configurations with more than two liquid stages were not taken into account, as they are quite uncommon.

In Tab. 3.15, the necessary parameters to estimate the avionics and wiring masses for all three rockets can be found. The results can be observed in Tab. 3.16, 3.20 and 3.22.

Table 3.15: Medium-Lift Launch Vehicle Parameters

PARAMETER	ATLAS V	SOYUZ-2	LVM3
LOWER STAGE			
Lift-Off Mass (M_0)	305,143 kg [78]	99,765 kg [79]	125,600 kg [80]
Length (l)	32.46 m [78]	27.1 m [79]	21.26 m [80]
Engine Thrust (T)	3,827 kN [78]	792 kN [79]	1,600 kN [81]
UPPER STAGE			
Lift-Off Mass (M_0)	23,073 kg [78]	27,755 kg [79]	18,300 kg [80]
Length (l)	12.68 m [78]	6.7 m [79]	13.32 m [80]
Engine Thrust (T)	99.2 kN [78]	297.9 kN [79]	200 kN [81]

Lower Liquid Stage Structure

The core stages of the three selected launchers use different types of propellants: Atlas V and Soyuz-2 run on LO_x and RP-1 [78, 79], while LVM3 runs on UH25 and N_2O_4 [80]. Due to this difference, some minor changes are to be expected in the internal structure of the rockets; however, since their dry masses are roughly of the same order of magnitude, the difference in architecture is not considered a result-changing factor. All three structures are made of aluminium lithium alloys [78, 79, 81] and are subjected to similar manufacturing processes.

In order to determine the amount of material needed for the main structures, all non-aluminium alloy component masses were subtracted from the empty mass of the lower stage (Tab. 3.16).

Table 3.16: Lower Stage Structural Mass of Medium-Lift Launchers

	ATLAS V	SOYUZ-2	LVM3
Dry Mass	21,054 kg [78]	6,545 kg [79]	10,600 kg [80]
Engines Mass	5,480 kg [82]	1,230 kg [83]	1,700 kg[1]
Thrust Frame Mass	975.885 kg	201.96 kg	408 kg
Insulation Mass	398.63 kg [84, 78]	257.68 kg [79, 84]	-[2]
Avionics Mass	954.785 kg	637.72 kg	693 kg
Wiring Mass	1,395 kg	762.46 kg	805.14 kg
STRUCTURE MASS	**11,849.7 kg**	**3,455.18 kg**	**6,993.86 kg**

[1] The Vikas engine mass is assumed to be the same as the Viking 4B [85], since it is an evolution of the latter [85].

[2] LVM3 uses a hypergolic propellant, which does not need cryogenic tanks [80]. The mass of the insulation is therefore neglected.

The average amount of aluminium alloys needed to manufacture the lower stage of a medium-lift launch vehicle is therefore 7,432.91 kg.

The manufacturing processes taken into account are: sheet rolling, die forging, anodizing and friction stir welding [86, 81]. Taking into account the values for each of the three rockets, a final mean value for all the manufacturing processes can be evaluated (Tab. 3.17).

Table 3.17: Manufacturing Processes for Medium-Lift Launchers - Lower Stage

PROCESS	FUNCTIONAL UNIT	ATLAS V	SOYUZ-2	LVM3	MEAN VALUE
Sheet Rolling	kg [10]	11,849.7	3,455.18	6,993.86	7,432.91 (See Tab. 3.16)
Die Forging	kg [17]	11,849.7	3,455.18	6,993.86	7,432.91 (See Tab. 3.16)
Anodising	m^2 [17]	388.53 [78]	251.15 [79]	206.09 [80]	281.92
Friction Stir Welding	m [17]	305.94 [78, 86]	221.02 [79]	262.37 [81]	263.11

Given the mean values, assuming that a real "mean" medium-lift lower stage could be built while respecting those values, the finished product can be implemented in SimaPro. A mean value of 328.155 kg of spray-on, rigid foam insulation is also added to the structure, while eventual thermal coatings (i.e. white paint) are neglected for lack of characterizing data.

Interstage

Atlas V and Soyuz-2 employ two different types of interstages: the first is made of a composite structure, with an aluminium honeycomb core [78], while the latter is an aluminium skin-stringer structure [79]. No specific detail on the structure of LVM3's interstage is available. The two processes to implement a generic interstage are therefore based on the data from a single rocket.

The data used for the composite interstage is available in Tab. 3.18. The selected process includes both the materials used and the production of the final component.

Table 3.18: Medium-Lift Launcher Interstage (Sandwich Structure) Manufacturing

PROCESS	FUNCTIONAL UNIT	ATLAS V
AL 5052 Honeycomb with CFRP facing [mass]	kg [17]	2219,5 [78]

Soyuz's interstage is assumed to be made of Aluminium AA 7075, 400 kg of which are computed to create the process [79]. The manufacturing process taken into account are sheet rolling, die forging, anodising and friction stir welding, as it is likely that the interstage goes through the same manufacturing processes as the main structure. The quantities associated to each process can be found in Tab. 3.19.

By computing the aluminium mass and the aforementioned processes, the aluminium interstage can be modelled in SimaPro.

Upper Liquid Stage Structure

Table 3.19: Medium-Lift Launcher Interstage (Aluminium Structure) Manufacturing

PROCESS	FUNCTIONAL UNIT	QUANTITY
Sheet Rolling	kg [10]	400 [79]
Die Forging	kg [17]	400 [79]
Anodising	m^2 [17]	8.83 [79]
Friction Stir Welding	m [17]	55.58 [79]

Although the upper stages of both Atlas V and LVM3 run on LO_x and LH_2 [78, 80], their structures are made of different materials: the Centaur upper stage is made of stainless steel [78], while the C25-X stage is made of AA2219 aluminium [80]. The latter is therefore similar to Soyuz's upper stage, which is also made of aluminium alloys [79], and their respective data can be averaged. In order to determine the amount of material necessary for the production of the main structures, all the main non-aluminium alloy components masses are subtracted from the total dry mass of the two stages (Tab. 3.20).

Table 3.20: Upper Stage Structural Mass of Medium-Lift Launchers (Aluminium)

	SOYUZ-2	LVM3
Dry Mass	2,355 kg [79]	3,300 kg [80]
Engines Mass	480 kg [87]	587 kg [88]
Thrust Frame Mass	75.96 kg	51 kg
Insulation Mass	57.44 kg [79, 84]	87.03 kg [89, 84]
Avionics Mass	401.83 kg	345.74 kg
Wiring Mass	283.58 kg	273.42 kg
STRUCTURE MASS	**1,056.19 kg**	**1,955.81 kg**

The average amount of aluminium alloys needed to manufacture the upper stage of a medium-lift launch vehicle is therefore 1,506 kg.

The manufacturing processes taken into account are: sheet rolling, die forging, anodizing and friction stir welding [81, 79]. Taking into account the values for each rocket, a final mean value for all the manufacturing processes can be evaluated (Tab. 3.21).

Given the mean values, assuming that a real "mean" medium-lift upper stage could be built while respecting those values, the finished product can be implemented in SimaPro. A mean value of 72.235 kg of spray-on, rigid foam insulation is also added to the structure, while eventual thermal coatings (i.e. white paint) are neglected for lack of characterizing data.

Although the data from Atlas V's upper stage cannot be averaged with

Table 3.21: Manufacturing Processes for Medium-Lift Launchers - Upper Stage (Aluminium)

PROCESS	FUNCTIONAL UNIT	SOYUZ-2	LVM3	MEAN VALUE
Sheet Rolling	kg [10]	1,056.19	1,955.81	1,506 (See Tab. 3.20)
Die Forging	kg [17]	1,056.19	1,955.81	1,506 (See Tab. 3.20)
Anodising	m^2 [17]	56 [79]	84.82 [89]	70.41
Friction Stir Welding	m [17]	63.54 [79]	123.98 [81]	93.76

data from other launch vehicles, a model for a stainless steel upper stage is implemented, based solely on the Common Centaur upper stage [78]. In order to determine the amount of material necessary for the production of the main structure, all the main non-stainless steel components masses are subtracted from the total dry mass of the stage (Tab. 3.22).

Table 3.22: Upper Stage Structural Mass of Medium-Lift Launchers (Stainless Steel)

	ATLAS V
Dry Mass	2,243 kg[1][78]
Engine Mass	215 kg [78]
Thrust Frame Mass	25.3 kg
Insulation Mass	103.13 kg [84, 78]
Avionics Mass	375.91 kg
Wiring Mass	303.26 kg
STRUCTURE MASS	**1,220.4 kg**

[1] 4x1 configuration considered.

The average amount of stainless steel needed to manufacture the upper stage of a medium-lift launch vehicle is therefore 1,220.4 kg.

The manufacturing processes taken into account are: stretching (modelled in SimaPro as "Forging, steel" [10]), cutting ("Laser Cutting, Stainless Steel Alloys, CO2 Laser" [17]), sheet rolling ("Sheet rolling, chromium steel" [10]) and resistance arc welding ("Welding, arc, steel" [10]) [86]. Since not all processes are specifically modelled after stainless steel products, the closest available option was chosen. The values associated to each process can be found in Tab. 3.23.

Given the values related to Atlas V only, the stainless steel upper stage can be implemented in SimaPro. A mass of 103.13 kg of spray-on, rigid foam insulation is taken into account, while eventual thermal coatings (i.e. white paint) are neglected for lack of characterizing data.

Table 3.23: Manufacturing Processes for Medium-Lift Launchers - Upper Stage (Stainless Steel)

PROCESS	FUNCTIONAL UNIT	ATLAS V
Forging	kg [10]	1,220.4 (See Table 3.22)
Laser Cutting	m [17]	198.595 [90, 91, 86]
Sheet Rolling	kg [10]	1,220.4 (See Table 3.22)
Arc Welding	m [10]	198.595 [90, 91, 86]

Fairing

The payload fairing for both Soyuz-2 and LVM3 is made of CFRP [79, 80], while Atlas V's is made of a sandwich structure, with an aluminium honeycomb core with CFRP face sheets. [92]. Due to this difference, two processes are implemented separately. The first one makes use of the process "CFRP, cured by autoclave process" [17]. Soyuz's fairing mass is available, while LVM3's is estimated assuming a thickness of 1 cm and a simplified shape made of a cylinder for two thirds of the length, and a cone for the remaining third. The results are available in Tab. 3.24.

Table 3.24: Fairing Mass of Medium-Lift Launchers (CFRP)

SOYUZ-2	LVM3	MEAN FAIRING MASS
1,700 kg [79]	2,270 kg [80, 93]	**1,985 kg**

The generic fairing can therefore be implemented in SimaPro using the mean mass and the aforementioned process, as it contains both the material and the manufacturing process.

Atlas V's fairing is made of aluminium with CFRP face sheets, with a mean mass (that takes into account Large, Extended and Extra Extended configurations) of 2,306.33 kg [78]. It can be modelled in SimaPro by using the "AL 5052 Honeycomb with CFRP facing [mass]" process [17].

Heavy Lift Liquid Propellant Launch Vehicles

The three vehicles taken into account for this category are: Falcon 9 by SpaceX, Ariane 6 by Ariane Group and Vulcan Centaur by ULA. These rockets are made of two liquid stages and eventual solid rocket boosters, which are implemented separately. Configurations with more than two liquid stages were not taken into account, as they are quite uncommon. It is important to note that Falcon 9 could be considered a medium-lift vehicle due to its usual payload mass; however, for the purpose of this estimation, it is considered in its expendable version, whose maximum payload fits into the heavy-lift category [94].

In Tab. 3.25, the necessary parameters to estimate the avionics and wiring masses for all three rockets can be found. The results can be observed in Tab. 3.26, 3.30 and 3.32.

Table 3.25: Heavy-Lift Launch Vehicle Parameters

PARAMETER	FALCON 9	ARIANE 6	VULCAN CENTAUR
LOWER STAGE			
Lift-Off Mass (M_0)	421,300 kg [95]	173,000 kg [96, 97]	162,289 kg [98, 99]
Length (l)	41.2 m [95]	32 m [96]	33.3 m [100]
Engine Thrust (T)	7,686 kN [101]	1,370 kN [97]	4,893 kN [102]
UPPER STAGE			
Lift-Off Mass (M_0)	96,570 kg [95]	35,100 [96]	57,311 kg [100, 98]
Length (l)	13.8 m [95]	12 m [96]	11.7 m [100]
Engine Thrust (T)	981 kN [101]	180 kN [97]	221.6 kN [103]

Lower Liquid Stage Structure

The three selected launchers all use different types of propellants: Falcon 9 runs on LO_x and RP-1 [101], Ariane 6 on LO_x and LH_2 [97] and Vulcan Centaur on LO_x and LNG [102]. Due to this difference, some minor changes are to be expected in the internal structure of the rockets: for example, Falcon 9 and Vulcan Centaur both have two tanks separated by a common bulkhead [101, 100], while Ariane 6 has an inter-tank separating the two tanks [97]. However, since their dry masses are of the same order of magnitude and the additional dome does not significantly influence the impact of manufacturing, the difference in architecture is not considered a result-changing factor, considering all three structures are made of aluminium-lithium alloys [101, 97, 86] and are produced with the same manufacturing processes [101, 104, 86].

In order to determine the amount of material needed for the main structures, all non-aluminium alloy component masses were subtracted from the empty mass of the first stage (Tab. 3.26).

Table 3.26: Lower Stage Structural Mass of Heavy-Lift Launchers

	FALCON 9	ARIANE 6	VULCAN CENTAUR
Dry Mass	25,600 kg [105]	23,000 kg [96]	32,689 kg [98, 106]
Engines Mass	4,410 kg [107]	2,000 kg [108]	6,244 kg [109, 110]
Thrust Frame Mass	1,959.93 kg	349.35 kg	1,247.72 kg
Insulation Mass	768 kg	450 kg [97, 84]	537 kg [100, 84]
Avionics Mass	1,072.7 kg	777.92 kg	760.18 kg
Recovery Systems Mass	2000 kg [111]	-	-
Wiring Mass	1,739.83 kg	1,046.64 kg	1,023.86 kg
STRUCTURE MASS	**13,649.54 kg**	**18,376.09 kg**	**22,876.24 kg**

The average amount of aluminium-lithium alloys needed to manufacture a heavy-lift launch vehicle is therefore 18,300.62 kg.

The manufacturing processes taken into account are: sheet rolling, die forging, anodizing and friction stir welding [101, 104, 86]. Taking into account the values for each rocket, a final mean value for all the manufacturing processes can be evaluated (Tab. 3.27).

Table 3.27: Manufacturing Processes for Heavy-Lift Launchers - Lower Stage

PROCESS	FUNCTIONAL UNIT	FALCON 9	ARIANE 6	VULCAN CENTAUR	MEAN VALUE
Sheet Rolling	kg [10]	13,649.54	18,376.09	22,876.24	18,300.62 (See Tab. 3.26)
Die Forging	kg [17]	13,649.54	18,376.09	22,876.24	18,300.62 (See Tab. 3.26)
Anodising	m^2 [17]	487 [101, 105]	580 [97, 104]	530 [100]	532.3
Friction Stir Welding	m [17]	458.2 [101, 112, 113, 114]	338 [97, 104]	302.7 [86, 100]	366.3

Given the mean values, assuming that a real "mean" heavy-lift lower stage could be built while respecting those values, the finished product can be implemented in SimaPro. A mean value of 585 kg of spray-on, rigid foam insulation is also added to the structure, while eventual thermal coatings (i.g. white paint) are neglected for lack of characterizing data.

Interstage

Falcon 9 and Vulcan Centaur were again taken into consideration to estimate the material and manufacturing processes needed to produce an interstage for heavy-lift launchers. Both launchers have an interstage made of composite panels with an aluminium honeycomb core and carbon fibre face sheet plies [101, 92], while Ariane 6's interstage is made of a carbon-fibre composite without the aluminium core [97] and thus cannot be taken into consideration to model this kind of interstage.

The sandwich structure is available in ESA's LCI database [17], with an assumed thickness comparable with the one employed on Falcon 9 [101], and can thus be modelled with the values available in Tab. 3.28.

Table 3.28: Heavy-Lift Launcher Interstage (Sandwich Structure) Manufacturing

PROCESS	FUNCTIONAL UNIT	FALCON 9	VULCAN CENTAUR	MEAN VALUE
AL 5052 Honeycomb with CFRP facing	m^2 [17]	78.53 m^2 [101, 105]	93.31 m^2 [100]	85.92 m^2

In order to model both options currently in use, an interstage made of GFRP (Glass Reinforced Carbon Polymer) cured in an industrial oven [96, 97] was modelled, even though data from Ariane 6 could not be averaged with data from other launchers. With an assumed thickness of 2 cm (comparable to that of the Falcon 9 interstage [101]), the results are available in Tab. 3.29.

Upper Liquid Stage Structure

Table 3.29: Heavy-Lift Launcher Interstage (GFRP Structure) Manufacturing

PROCESS	FUNCTIONAL UNIT	ARIANE 6
GFRP, cured by Autoclave process	kg [17]	1693 kg [96, 97]

For the upper stage structure, a similar approach to the lower stage structure can be applied; however, while both Falcon 9 and Ariane 6's upper stages are made of friction stir welded aluminium-lithium alloys [101, 97], the Centaur upper stage is made of stainless steel [100], thus requiring a distinction between the two possible architectures.

For the two aluminium-lithium stages, the amount of material needed is computed in Tab. 3.30.

Table 3.30: Upper Stage Structural Mass of Heavy-Lift Launchers (Aluminium)

	FALCON 9	ARIANE 6
Dry Mass	3,900 kg [95]	5,100 kg[1]
Engines Mass	550 kg [115]	280 kg [116]
Thrust Frame Mass	250.16 kg	45.9 kg
Insulation Mass	127 kg [101, 105, 84]	224 kg [97, 84]
Avionics Mass	630.27 kg	437.38 kg
Wiring Mass	633.69 kg	368.92 kg
STRUCTURE MASS	**1,708.88 kg**	**3,743.8 kg**

[1] Assuming the same propellant/dry mass ratio as the lower stage.

The average amount of aluminium-lithium alloys needed to manufacture an upper stage is therefore 2,726.34 kg. The manufacturing processes presented for the lower stage are taken into account for the upper stage as well (Tab. 3.31).

Given the mean values, assuming that a real "mean" heavy-lift upper stage could be built while respecting those values, the finished product can be implemented in SimaPro. A mean value of 175.5 kg of spray-on, rigid foam insulation is also added to the structure, while eventual thermal coatings (i.e. white paint) are neglected for lack of characterizing data.

Although the data from Vulcan Centaur's upper stage cannot be averaged with data from other launch vehicles, a model for a stainless steel upper stage is implemented, based solely on the upgraded Centaur V second stage [102]. In order to determine the amount of material necessary for the production of the main structure, all the main non-stainless steel components masses are subtracted from the total dry mass of the stage (Tab. 3.32).

The average amount of stainless steel needed to manufacture the upper stage of a medium-lift launch vehicle is therefore 1,027.93 kg.

The manufacturing processes taken into account are: stretching (modelled

Table 3.31: Manufacturing Processes for Heavy-Lift Launchers - Upper Stage

PROCESS	FUNCTIONAL UNIT	FALCON 9	ARIANE 6	MEAN VALUE
Sheet Rolling	kg [10]	1,708.88	3,743.8	2,726.34 (See Tab. 3.30
Die Forging	kg [17]	1,708.88	3,743.8	2,726.34 (See Tab. 3.30)
Anodising	m^2 [17]	115 [101, 105]	203.6 [97]	159.3 m
Friction Stir Welding	m [17]	154.25[1][101, 105]	224.86 [104, 97]	189.555

[1] Missing data was scaled from lower stage.

Table 3.32: Upper Stage Structural Mass of Heavy-Lift Launchers (Stainless Steel)

	VULCAN CENTAUR
Dry Mass	2,880 kg [100, 98]
Engine Mass	634 kg [103]
Thrust Frame Mass	56.51 kg
Insulation Mass	171.06 kg [100, 84]
Avionics Mass	522.06 kg
Wiring Mass	468.44 kg
STRUCTURE MASS	**1,027.93 kg**

in SimaPro as "Forging, steel" [10]), cutting ("Laser Cutting, Stainless Steel Alloys, CO2 Laser" [17]), sheet rolling ("Sheet rolling, chromium steel" [10]) and resistance arc welding ("Welding, arc, steel" [10]) [86]. Since not all processes are specifically modelled after stainless steel products, the closest available option was chosen. The values associated to each process can be found in Tab. 3.33. Given the values related to Vulcan Centaur only, the stainless steel upper stage can be implemented in SimaPro. A mass of 171.06 kg of spray-on, rigid foam insulation is taken into account, while eventual thermal coatings (i.e. white paint) are neglected for lack of characterizing data.

Fairing

Much like the interstage structures, both Falcon 9 and Vulcan Centaur's fairings are made of a composite sandwich structure, with an aluminium honeycomb core and carbon fibre face sheets [101, 100], while Ariane 6's is made of GFRP [96]. For all launchers, the fairing is assumed to be cylindrical for the first two thirds of the height, and conical for the last third. Although Vulcan Centaur and Ariane 6 both have a "short" and "long" fairing option [97, 102], only the "short" one is considered, as its dimensions are compatible with Falcon 9's.

The values taken into account for the first type of fairing are available in Tab. 3.34.

The values for the second type of fairing are available in Tab. 3.35.

Table 3.33: Manufacturing Processes for Heavy-Lift Launchers - Upper Stage (Stainless Steel)

PROCESS	FUNCTIONAL UNIT	VULCAN CENTAUR
Forging	kg [10]	1,027.93 (See Table 3.32)
Laser Cutting	m [17]	245.1 [86, 100, 90, 91]
Sheet Rolling	kg [10]	1,027.93 (See Table 3.32)
Arc Welding	m [10]	245.1 [86, 100, 90, 91]

Table 3.34: Heavy-Lift Launcher Fairing (Sandwich Structure) Manufacturing

PROCESS	FUNCTIONAL UNIT	FALCON 9	VULCAN CENTAUR	MEAN VALUE
AL 5052 Honeycomb with CFRP facing	m^2 [17]	185.5 m^2 [101]	224.69 m^2 [100]	205.095 m^2

Super Heavy Lift Launch Vehicles

The structure for super heavy-lift vehicles has not been implemented, since little no data is available on current launchers. Falcon Heavy can be implemented by considering three heavy-lift lower stages (corresponding to three Falcon 9 first stages) [101], while both Starship and NASA's SLS (Space Launch System) still lack the necessary information to correctly characterise and model them.

Solid Rocket Motors

Solid rocket motors can be characterized based on the level of thrust they are able to produce: since most of the solid rocket boosters currently in use produce high levels of thrust, the two categories considered are <1,000 kN and >1,000 kN.

The first category is represented only by the Zefiro 9 booster; the availability of only one motor in this category poses some limitation to its accuracy, but it has been implemented nonetheless to encompass all possible options. In Tab. 3.36, the specifications for the considered motor can be found.

The motor structure can be implemented in SimaPro by considering 870 kg (only the case and nozzle masses have been considered) of the process "CFRP, cured by autoclave process" [17], which includes both the material and the manufacturing.

The second category comprises of the GEM63, GEM63XL, P120C and SRB-3A boosters. In Tab. 3.37, their specifications can be found.

The motor structure can be implemented in SimaPro by considering a mean amount of 7,323.5 kg of the process "CFRP, cured by autoclave process" [17], which includes both the material and the manufacturing.

Table 3.35: Heavy-Lift Launcher Fairing (Carbon - Glass Fibre Composite Structure) Manufacturing

PROCESS	FUNCTIONAL UNIT	ARIANE 6
GFRP, cured by Autoclave process	kg [17]	1,800 kg [117]

Table 3.36: Solid Rocket Motor Structure, <1,000 kN Thrust

ZEFIRO 9	
Material	CFRP [118]
Inert Mass	1,000 kg [118]
Case Mass	717 kg [1]
Nozzle Mass	153 kg [2]
Insulation Mass	102 kg [3]
Avionics Mass	14 kg [4]
Wiring Mass	14 kg [4]

[1] Assumed 71.7% of inert mass, based on [119].
[2] Assumed 15.3% of inert mass, based on [119].
[3] Assumed 10.2% of inert mass, based on [119].
[4] Wiring and avionics are assumed to be 50% each of the remaining inert mass, once the case, nozzle and insulation masses have been subtracted from the total dry mass.

Mechanisms

Mechanisms are also part of the structure [42]; the main components that need to be implemented are stage separation mechanisms, as other components such as hydraulic actuators are already implemented in ESA's database [17]. The main type of stage separation mechanisms taken into account is pyrotechnic mechanisms. In order to start the event, a stage separation controller is needed, whose function is the same as the solid booster ignition controller [123], which is implemented in Subs. 3.3.2. The pyrotechnic component of the mechanism can be also be modelled through the pyrotechnic ignition powder, also implemented in Subs. 3.3.2.

The main element that needs to be modelled for a pyrotechnic separation system is the frangible joint. For the sake of simplicity, it is modelled as a hollow cylinder made of Aluminium 6061 [124]. Considering a mean launch vehicle diameter of 4.025 m [78, 79, 80, 96, 101, 100], a thickness of 1.59 cm and a length of 0.29 m [124], the total amount of aluminium needed to manufacture

Table 3.37: Solid Rocket Motor Structure, >1,000 kN Thrust

	GEM63	**GEM63XL**	**P120C**	**SRB-3A**
Material	CFRP [100]	CFRP [120]	CFRP [96]	CFRP [121]
Inert Mass	5,100 kg [99]	5,400 kg [99]	11,000 [122]	8,700 kg [121]
Case Mass	3,656.7 kg[1]	3,871.8 kg[1]	8,300 kg [122]	6,237.9 kg[1]
Nozzle Mass	780.3 kg[2]	826.2 kg[2]	1,683 kg[2]	1,331.1 kg[2]
Insulation Mass[3]	520.2 kg	550.8 kg	704 kg[4]	887.4 kg
Avionics Mass[5]	71.4 kg	75.6 kg	156.5 kg	121.8 kg
Wiring Mass[5]	71.4 kg	75.6 kg	156.5 kg	121.8 kg

[1] Assumed 71.7% of inert mass, based on [119].
[2] Assumed 15.3% of inert mass, based on [119].
[3] Assumed 10.2% of inert mass, based on [119].
[4] Assumed to be 6.4% of the inert mass, as the case makes up more than 71.7%.
[5] Wiring and avionics are assumed to be 50% each of the remaining inert mass, once the case, nozzle and insulation masses have been subtracted from the total dry mass.

the joint is 156.8 kg. The manufacturing processes taken into consideration are "Impact extrusion of aluminium, 1 stroke" and "Sheet Rolling, Aluminium" [10, 124].

Other types of separation mechanisms, such as pneumatic pushers, have not been taken into account as they are less common than the pyrotechnic one.

3.3.2 Propulsion System

Rocket engines can run on either solid, liquid or hybrid propellants [125]. Different propellant combinations lead to different architectures; liquid rocket engines will be thoroughly explained in this section, while solid rocket motors, since their nozzle is typically made from the same material of the case, have already been discussed in Subs. 3.3.1. Hybrid engines are an uncommon design choice and, as such, will not be implemented. The last part of this section will deal with engine igniters.

Liquid Rocket Engines

The main components of a liquid rocket engine are the combustion chamber, the nozzle, the pumps, the igniters, and the flow systems [42]. According to [126], the average mass composition (in percentage) for the SC (Staged Combustion), GG (Gas Generator) and EX (Expander) cycles in liquid rocket engines can be found in Tab. 3.38 and 3.39.

Table 3.38: Mass Percentage of Main Engine Components - 1 [126]

	NOZZLE				**COMBUSTION CHAMBER**
CYCLE	TUBES	MANIFOLD	JACKET	RADIATIVE NOZZLE	COMBUSTION CHAMBER
SC	6.4%	17.17%	4.7%	15.66%	11.37%
GG	9.87%	1.46%	5.89%	19.69%	19.34%
EX	6.29%	23.01%	2.1%	0	20.07%

Table 3.39: Mass Percentage of Main Engine Components - 2 [126]

	PUMPS			**FLOW SYSTEMS**		**STRUCTURE**
CYCLE	GAS GENERATOR	OX TURBO-PUMP	FUEL TURBO-PUMP	OX VALVE	FUEL VALVE	STRUCTURE
SC	5.51%	10.89%	14.07%	6.98%	6.9%	0.35 %
GG	4.38%	12.54%	13.54%	3.52%	3.52%	6.25%
EX	0	14.11%	14.11%	2.02%	1.85%	16.44%

Since the thrust influences the engine dimensions, five different levels have arbitrarily been defined:

1. 0 - 1 kN
2. 1 kN - 10 kN
3. 10 kN - 100 kN
4. 100 kN - 1,000 kN
5. More than 1,000 kN

No engines with Level 1 thrust have been located.

Eventual engines which do not perform any of the three aforementioned cycles (e.g. which are pressure-fed or use electric pumps) are not taken into account for lack of characterising data. For this reason, no engines with Level 2 thrust figure in the following tables. In Tab. 3.40, 3.42 and 3.44, the engine mass for different categories (both in thrust level and type of cycle performed) can be found. In Tab. 3.41, 3.43 and 3.45, the respective mean component mass (based on Tab. 3.38 and 3.39) for each engine architecture and level of thrust can be found. Level 3 engines are characterised by only one real case component for each cycle, while for Level 4 thrust an average mass of 940 kg for the staged combustion engine, 968.33 kg for the gas generator engine, and 288.67 kg for the expander engine has been taken into account. For Level 5 thrust, an average mass of 3,203.5 is taken into account for staged combustion engines, while the gas generator architecture is based only on the Vulcain 2.1 engine, and no expander engine for this level of thrust has been identified.

Table 3.40: Level 3 Thrust Engine Mass

Level 3 Thrust	
ENGINE	MASS
Staged Combustion	
CE-7.5	435 kg [127]
Gas Generator	
YF-75	550 kg [128]
Expander	
YF-75D	265 kg [128]

Table 3.41: Level 3 Thrust Engine Components Mass

Level 3 Thrust				
NOZZLE	COMBUSTION CHAMBER	PUMPS	FLOW SYSTEMS	STRUCTURE
Staged Combustion				
191.1 kg	49.46 kg	132.54 kg	60.38 kg	1.52 kg
Gas Generator				
203 kg	106.37 kg	167.53 kg	38.72 kg	34.38 kg
Expander				
83.21 kg	53.19 kg	74.78 kg	10.25 kg	43.57 kg

Since specific data on the material composition of each engine is usually classified, the most commonly used materials are taken into account:

- Nozzle extension (lower stages): Inconel 600 or stanless-steel [138]
- Nozzle extension (upper stages): carbon-carbon composite [138]
- Combustion chamber: copper-chromium alloy inner wall, low-alloy steel, stainless steel or nichel based outer shell [138]
- Pumps: Inconel 625 or Inconel 718 [138]
- Flow systems (valves): A7075 [138, 81]
- Flow systems (ducts): A625 [138]
- Structure: 4340 steel alloy or 440C high strength stainless steel [138]

For components where multiple material choices can be made, only one of the possible solutions was chosen based on its availability in either ESA or EcoInvent databases [10, 17]. Therefore, the nozzle extension for lower stage engines is considered made of stainless steel 321 [17], the outer shell of the combustion chamber and the pumps are considered made of iron-nickel-chromium alloy [10], and the structure is considered made of 440B steel [17], as 440C is not implemented. The combustion chamber is considered 20% in weight inner wall,

Table 3.42: Level 4 Thrust Engine Mass

Level 4 Thrust	
ENGINE	MASS
Staged Combustion	
LE-7A	1,800 kg [129]
RD-0124	480 kg [83]
YF-115	540 kg [130]
Gas Generator	
CE-20	587 kg [131]
Merlin 1D	490 kg [107]
Merlin 1D Vacuum	550 kg [115]
RD-108A	1,400 kg [132]
YF-40	83 kg [133]
YF-77	2,700 kg [134]
Expander	
LE-5B	269 kg [135]
RL-10C	317 kg [103]
Vinci	280 kg [116]

Table 3.43: Level 4 Thrust Engine Components Mass

Level 4 Thrust				
NOZZLE	COMBUSTION CHAMBER	PUMPS	FLOW SYSTEMS	STRUCTURE
Staged Combustion				
412.94 kg	106.88 kg	286.42 kg	130.47 kg	3.29 kg
Gas Generator				
357.41 kg	187.28 kg	294.95 kg	68.17 kg	60.52 kg
Expander				
90.64 kg	57.94 kg	81.46 kg	11.17 kg	47.46 kg

80% outer shell, while the flow systems are considered 30% in weight valves, and 70% ducts.

The manufacturing processes taken into account are:

- Nozzle: sheet rolling ("Sheet rolling, chromium steel"), TIG (Tungsten Inert Gas) welding ("TIG Welding, Inconel [17]) [17]. All other processes (such as sheet cutting and bending [17]) are considered "Metal working, average for metal product manufacturing" [10] due to the lack of more specific processes. Steel has been used as a proxy for Inconel when no specific process for the latter could be implemented [17].

- Combustion chamber (inner wall): deep drawing [81], for which the proxy "Deep Drawing, steel, 3500 kN press, automode" [10] is used.

- Combustion chamber (outer shell): sheet rolling ("Sheet rolling, chromium

Table 3.44: Level 5 Thrust Engine Mass

Level 5 Thrust	
ENGINE	MASS
Staged Combustion	
RD-191	3,230 kg [136]
RS-25	3,177 kg [137]
Gas Generator	
Vulcain 2.1	2,000 kg [108]
Expander	
-	

Table 3.45: Level 5 Thrust Engine Components Mass

Level 5 Thrust				
NOZZLE	COMBUSTION CHAMBER	PUMPS	FLOW SYSTEMS	STRUCTURE
Staged Combustion				
1,407.3 kg	364.24 kg	976.11 kg	444.65 kg	11.2 kg
Gas Generator				
738.2 kg	386.8 kg	609.2 kg	140.8 kg	125 kg
Expander				
-				

steel" [10]), TIG welding ("TIG Welding, Inconel [17]), CNC (Computer Numerical Control) vertical turning ("Inconel Alloys removed by turning, CNC" [10]) [81]. All other processes (such as expansion forming and horizontal boring [81]) are considered "Metal working, average for metal product manufacturing" [10] due to the lack of more specific processes. Steel has been used as a proxy for Inconel when no specific process for the latter could be implemented [17].

- Pumps: CNC turning ("Inconel Alloys removed by turning, CNC" [10]), milling ("Inconel Alloys by milling" [10]), TIG welding ("TIG Welding, Inconel [17]), heat treatment ("Heat Treatment, Annealing, Inconel alloys", "Heat Treatment, Solution Treatment and Aging, Inconel alloys", "Heat Treatment, Stress Relieving, Inconel alloys" [17]) [81]. All other processes (such as forming/spinning, jig boring, spline cutting, chromium oxide coating, passivation [81]) are considered "Metal working, average for metal product manufacturing" [10] due to the lack of more specific processes. Steel has been used as a proxy for Inconel when no specific process for the latter could be implemented [17].

- Valves: pocket milling ("Aluminium removed by milling, small parts" [10]), EDM (Electrical Discharge Machining) ("Aluminium and Al-Li alloys removed by Spark Erosion, conventional EDM" [17]), CNC turning ("Aluminium and Al-Li alloys removed by turning, CNC" [17]). [81].

- Ducts: machining [81], considered as "Metal working, average for metal product manufacturing" [10].
- Structure: due to lack of more specific information, the manufacturing of the structural elements of the engine is considered "Metal working, average for metal product manufacturing" [10].

Igniters

Solid rocket motor igniters can either be pyrogen or pyrotechnic [125]. These types of igniters could also apply to liquid engines, but the typical architecture used in this case is usually simpler, as they only need to produce a spark to work [125]. Since pyrogen igniters consist of a smaller propellant grain [125], they can be simply modelled considering the necessary amount of solid propellant.

The main parts considered for the pyrotechnic igniter are the ignition controller and pyrotechnic powder, while the casing is assumed to be part of the main structure.

The ignition controller can be modelled as "equipment with low integrated circuit content" [9], based on [123]. Only one controller has been chosen due to the lack of more specific information on other models; this poses some limitations to the accuracy of the model, which has been implemented nonetheless to broadly quantify its environmental impact. According to [9], 2% of the controller weight is made of "Integrated circuit (logic type)", 49% of "Other electronic components", 47% of "Printed Wiring Board, empty" and 2% of "Silicone coating". In Tab. 3.46, the specifications of the modelled controller can be found.

Table 3.46: Ignition Controller Specifications

Ignition Controller	**WEIGHT**	**SURFACE**
IGNITION AND STAGING CONTROLLER	20.41 kg [123]	6,090.31 cm^2 [123]

Based on the aforementioned weight percentage division, the mean amounts of the following quantities can be considered:

- Integrated circuit (logic type): 0.408 kg
- Other electronic components: 10 kg
- Printed wiring board, empty: 9.594 kg
- Silicone Coating: 0.408 kg

The manufacturing processes are "Mounting of PWB (Printed Wiring Board)" (6,090.31 cm^2) and "Electricity for Bake-out" (0.169 kWh, scaled from [9]). The ignition and staging controller can then be implemented in SimaPro.
The pyrotechnic powder taken into consideration is black carbon, which is made of 75% KNO_3, 15% charcoal and 10% sulphur [139]. The average amount of powder is 604.66 g [139].

3.3.3 Power System

The main tasks of the power system involve power storage, conditioning and transfer [42].

Power Storage

The main component taken into account for power storage are batteries (specifically, Lithium-Ion batteries). Other auxiliary components such as capacitors and super-capacitors can already be found in EcoInvent's database [10].

Based on ESA's guidelines, 70% of a battery's weight is made of "Cells, Li-ion (mass)", 25% of "Aluminium casing (mass)" and 5% of "Electronic Unit, low IC (Integrated Circuit) (mass)", to which the aluminium casing surface needs to be added [9]. In Tab. 3.47, the batteries taken into account and their specifications can be found.

Table 3.47: Battery Specifications

BATTERY	MASS	SURFACE
Lithium-Ion FTS Battery 39401	0.68 kg [140]	381.39 cm^2 [140]
Lithium-Ion FTS & TM Battery 39401-1	0.73 kg [141]	360.06 cm^2 [141]
Lithium-Ion Battery 39611 Smart Series	1.68 kg [142]	887.38 cm^2 [142]
Lithium-Ion FTS Battery 39611-1 Smart Series	1.68 kg [143]	887.38 cm^2 [143]
High Power Li-Ion Polymer Battery 39381-4	5.22 kg [144]	1,406.68 cm^2 [144]
Lithium-Ion Battery 39501 Series	1 kg [145]	585.16 cm^2 [145]
High Capacity Li-Ion Battery 39541 Series	4.4 kg [146]	1,245.91 cm^2 [146]
Modular Li-Ion Battery 39521 Series	4.63 kg [147]	1,547.51 cm^2 [147]
LAUNCHER-BATT - S	2.2 kg [148]	1,342 cm^2 [148]
LAUNCHER-BATT - L	9.8 kg [148]	3,368 cm^2 [148]

Based on the aforementioned weight percentage division, the mean amounts of the following quantities can be considered:

- Cells, Li-Ion Mass: 2.24 kg
- Aluminium Casing (Mass): 0.8 kg
- Electronic Unit, low IC (Mass): 0.16 kg
- Aluminium Casing (Surface): 1,201.15 cm^2

The manufacturing processes and their relative quantities taken into account are [9]:

- Sheet Rolling, Aluminium: 0.8 kg
- Anodising, Aluminium: 1,201.15 cm^2
- Cleansing with Solvent: 1,201.15 cm^2
- Electricity Consumption for Processes: 1.7 kWh (scaled from [9])

Based on this data, the generic Lithium-Ion battery can then be modelled.

Power Distribution

Power distribution is handled by components such as wires and cables, which are already implemented in both EcoInvent and ESA's databases [10, 17], and power distribution units. Power distribution units are usually tailored to the launcher's specific features. The generic model for the PDU (Power Distribution Unit) is therefore based on the only off-the-shelf unit available. This poses some limitations to the accuracy of the model, which has been implemented nonetheless to broadly quantify the environmental impact of the launch vehicle electronics.

The power distribution units has been modelled as "PDU / PCDU (Power Conditioning and Distribution Unit)": according to [9], the product "Power supply unit, at plant" can be taken into account. In Tab. 3.48, the specifications of the modelled power distribution unit can be found.

Table 3.48: Power Distribution Unit Specifications

POWER DISTRIBUTION UNIT	MASS
PDU300|POWER DISTRIBUTION	0.79 kg [149]

Therefore, in order to model the PDU, 0.79 kg of the aforementioned process can be taken into account.

Power Conditioning

The devices taken into account for power conditioning are: DC-DC (Direct Current - Direct Current) converters, EMI (Electromagnetic Interference) filters, transformers and fuses. Solid state switches can be considered as toggle switches, in a first approximation. Other components, such as voltage regulators, inverters, low-pass filters, circuit breakers, relays, and monitoring sensors, have not been considered due to the lack of specific information.

DC-DC converters can be considered electronic equipment "with low integrated circuit content" [9]; 2% of their weight is considered made of "Integrated circuit (logic type)", 49% of "Other electronic components", 47 % of "Printed Wiring Board, empty" and 2% of "Silicone coating" [9]. In Tab. 3.49, the DC-DC converters taken into account and their specifications can be found.

Table 3.49: DC-DC Converter Specifications

DC-DC CONVERTER	MASS	SURFACE
SVFL2800D SERIES	88 g [150]	29.18 cm^2 [150]
SVTR2800D SERIES	55 g [151]	20.97 cm^2 [151]
SVSA2800D SERIES	15 g [152]	7.46 cm^2 [152]
SVLFL2800D SERIES	88 g [153]	29.18 cm^2 [153]
SVLTR2800D SERIES	55 g [154]	20.98 cm^2 [154]
SVLHF2800S SERIES	28 g [155]	14.58 cm^2 [155]
SVLSA5000D SERIES	15 g [156]	7.46 cm^2 [156]
SVRFL2800D SERIES	88 g [157]	29.18 cm^2 [157]

Based on the aforementioned weight percentage division, the mean amounts of the following quantities can be considered:

- Integrated Circuit (Logic Type): 1.08 g
- Other Electronic Components: 26.46 g
- Printed Wiring Board, Empty: 25.38 g
- Silicone Coating: 1.08 g

The average surface is 19.87 cm^2.

The manufacturing processes taken into account are "Mounting of PWB" and "Electricity for bake-out" [9]; the energy needed is $4.46 \cdot 10^{-4}$ KWh, scaled from the data available in [9].

Based on this data, the generic DC-DC converter can then be modelled in SimaPro.

EMI filters are electronic components with no integrated circuit content; they can therefore be modelled considering the guidelines for electronic equipment ”with low integrated circuit content”, removing the 2% in weight reserved to "Integrated circuit (logic type)" and assigning it to "Other electronic components" [9]. The weight percentage division is thus: 51% "Other electronic components", 47% "Printed Wiring Board, empty", and 2% "Silicon Coating" [9]. In Tab. 3.50, the EMI filters taken into account and their specifications can be found.

Based on the aforementioned weight percentage division, the mean amounts of the following quantities can be considered:

Table 3.50: EMI Filter Specifications

EMI FILTER	WEIGHT	SURFACE
VSCF20-28	75 g [158]	23.5 cm^2 [158]
DVME28 Series	83 g [159]	29.18 cm^2 [159]
VSCF10-28	40 g [160]	16.11 cm^2 [160]
DVMD28 Series	57 g [161]	16.98 cm^2 [161]
SVRMC28 Series	55 g [162]	20.98 cm^2 [162]
DVMD50 Series	57 g [163]	16.98 cm^2 [163]
VSCF1-28	21 g [164]	7.95 cm^2 [164]
DVMH50 Series	29 g [165]	10.64 cm^2 [165]

- Other Electronic Components: 26.58 g
- Printed Wiring Board, Empty: 24.5 g
- Silicone Coating: 1.04 g

The average surface is 17.79 cm^2, which is needed for the manufacturing process "Mounting of PWB", while $4.305 \cdot 10^{-4}$ KWh of "Electricity for bake-out" are taken into account, scaled from [9].

Based on this data, the generic EMI filter can then be modelled.

Power transformers are electronic components with no integrated circuit content; they can therefore be modelled considering the guidelines for electronic equipment ”with low integrated circuit content”, removing the 2% in weight reserved to "Integrated circuit (logic type)" and assigning it to "Other electronic components" [9]. The weight percentage division is thus: 51% "Other electronic components", 47% "Printed Wiring Board, empty", and 2% "Silicon Coating" [9]. In Tab. 3.51, the power transformers taken into account and their specifications can be found.

Table 3.51: Transformer Specifications

TRANSFORMER	MASS	SURFACE
SGTPL2516T02812x	40 g [166]	5.83 cm^2 [166]
SGTPL2516T02805x	40 g [166]	5.83 cm^2 [166]
SGTPL2511T02812x	28 g [166]	6.79 cm^2 [166]
SGTPL2511T02805x	28 g [166]	6.79 cm^2 [166]
SGTPL-2516	37 g [167]	7.62 cm^2 [167]

Based on the aforementioned weight percentage division, the mean amounts of the following quantities can be considered:

- Other Electronic Components: 17.646 g
- Printed Wiring Board, Empty: 16.262 g

- Silicone Coating: 0.692 g

The average surface is 6.572 cm^2, which is needed for the manufacturing process "Mounting of PWB", while $2.858 \cdot 10^{-4}$ KWh of "Electricity for bake-out" are taken into account, scaled from [9].

Based on this data, the generic power transformer can then be modelled.

Fuses made specifically for space applications can be considered made of of copper alloy terminals in a ceramic casing [168, 169, 170, 171]. It is assumed that 80% of the weight is made of ceramics, and the remaining 20% is made of metals. Since no specific process for fuses is available in databases, the process "Diode, ceramic packaging|Production" [17] is used as a proxy for the insulator part, the process "Copper part for wires|Production" [17] is used to model the copper alloy, to which the "Tin plating, pieces" process [10] is added. In Tab. 3.52, the fuses taken into account and their specifications can be found.

Table 3.52: Fuse Specifications

FUSE	MASS	TERMINAL SURFACE
MGA-S	0.035 g [168]	2.4 mm^2 [168]
HCSF	0.8 g [169]	25 mm^2 [169]
UMT-W	1.42 g [170]	29.96 mm^2 [170]
HCF	0.8 g [171]	25 mm^2 [171]

Based on the aforementioned weight percentage division, the mean amounts of the following quantities can be considered:

- Diode, Ceramic Packaging: 0.611 g
- Copper Part for Wires: 0.153 g
- Tin Plating, Pieces: 20.59 mm^2

As all processes include both the material and the production processes, the generic fuse can be directly implemented in SimaPro.

3.3.4 Communication System

The communication system aboard a launch vehicle mainly consists of radios and antennas [42].

Radios

Radios can be modelled as "Transponders" [9]. Based on ESA's guidelines [9], 20% of the weight is made of "Aluminium Casing", while the remaining 80%

Table 3.53: Radio Specifications

RADIO	MASS	SURFACE
FRONTIER - X	920 g [172]	23552 mm^2 [172]
FRONTIER - S	590 g [173]	22559.46 mm^2 [173]
T-740E S-BAND LAUNCH VEHICLE TRANSMITTER	7.03 kg [174]	2194.83 cm^2 [174]

is made of "Electronic Unit". In Tab. 3.53, the radios taken into account and their specifications can be found.

Based on the aforementioned weight percentage division, the mean amounts of the following quantities can be considered:

- Aluminium Casing: 569.33 g
- Electronic Unit: 2,227.33 g

The manufacturing processes taken into account are [9]:

- Electricity consumption for processes: 1,218.37 kWh (scaled from [9])
- Sheet Rolling, Aluminium: 569.33 g
- Anodising, Aluminium: 746.98 cm^2
- Cleaning with Solvent: 746.98 cm^2

Based on this data, the generic radio can then be modelled.

Antennas

Launch vehicle antennas cover a range of functions, from telemetry to guidance and navigation, transmitting real time information to ground stations and allowing for optimal mission tracking and control [175]. Launcher-specific antennas require particular mechanical properties, as they have to withstand intense vibrations, rapid temperature changes, and high aerodynamic forces [175].

Due to the lack of specific information, a generic antenna made of a metallic frame, an insulator and a heat shield is considered; the insulator is assumed to make up 20% of the weight [9], the metallic frame 70% (lowered from the 80% assumed in [9] to account for the heat shield), and the heat shield the remaining 10%. The antennas and their relative properties that were taken into account, given the availability of their specific data and the similarities to other antennas ([175]), can be found in Tab. 3.54:

The average amounts of materials needed therefore are:

- Aluminium: 462.93 g
- Fused Silica: 132.27 g

Table 3.54: Antenna Materials

ANTENNA	MASS	METALLIC FRAME MATERIAL	INSULATOR MATERIAL	HEAT SHIELD MATERIAL
AS-49090 SINGLE-LAYER MICROSTRIP PATCH ANTENNA	454 g [176]	Aluminium[1]	Fused Silica [176]	-
AS-49034 CAVITY-BACKED HELIX ANTENNA	1140 g [177]	Aluminium[1]	Fused Silica [177]	Resin Quartz Composite [177]
AS-49063 MONOPOLE BLADE ANTENNA	390 g [178]	Aluminium[1]	Fused Silica [178]	Resin Quartz Composite [178]

[1] Assumed based on [9].

- Resin Quartz Composite: 66.13 g

To model the antenna in SimaPro, the metallic frame is considered made of aluminium 6061 that is cast and die forged [9], the insulator is made of "sand (quartz)" that is manufactured as "flat glass" [10], and the heat shield is made of "epoxy resin insulator, SiO_2", which includes both the raw materials and the manufacturing processes.

3.3.5 Command & Data Handling System

The command and data handling system mainly consists of command computers and data processing, storage and distribution electronics [42]. The data processing functions are considered to be carried out by the flight computer itself.

Command Computers

On-board flight computers are usually tailored to the launcher's specific features. The generic model for the command computer is therefore based on the only off-the-shelf flight computer available. This poses some limitations to the accuracy of the model, which has been implemented nonetheless to broadly quantify the environmental impact of the launch vehicle avionics. The computer has been modelled as "SMU (Satellite Management Unit)/ OBC (On-Board Computer)": according to [9], 20% of its weight consists of "Aluminium casing", while the remaining 80% is made of "Electronic Unit, high IC". In Tab. 3.55, the specifications of the modelled flight computer can be found.

Table 3.55: Flight Computer Specifications

FLIGHT COMPUTER	MASS	SURFACE
PHOENIX FLIGHT COMPUTER	5.44 kg [179]	2,171.48 cm^2 [179]

Based on the aforementioned weight percentage division, the mean amounts of the following quantities can be considered:

- Aluminium casing: 1.08 kg
- Electronic Unit, high IC: 4.32 kg

The manufacturing processes taken into account are "Sheet rolling aluminium", "Anodising of aluminium", "Cleaning with solvent" and "Electricity consumption". The electricity consumption is 2.349 kWh, scaled from [9].

Data Storage Units

Data storage is mainly handled through SSDR (Solid State Data Recorders). They can be modelled as "electronic units with high memory type integrated circuit" [9]. According to [9], 10% of the weight is made of "Integrated circuit (memory type)", 56% of "Other electronic components", 32% of "Printed Wiring Board, empty" and 2% of "Silicone coating". In Tab. 3.56, the SSDRs taken into account and their specifications can be found.

Table 3.56: Solid State Data Recorders Specifications

SSD	WEIGHT	SURFACE
RH3440 Solid-State Data Recorder	620 g [180]	16000 mm^2 [180]
RH304T Solid-State Data Recorder	750 g [181]	16000 mm^2 [181]
RH3480 Solid-State Data Recorder	750 g [182]	22000 mm^2 [182]

Based on the aforementioned weight percentage division, the mean amounts of the following quantities can be considered:

- Integrated Circuit (Memory Type): 70.67 g
- Other Electronic Components: 395.73 g
- Printed Wiring Board, Empty: 226.13 g (18000 mm^2, as the required functional unit is mm^2 [10])
- Silicone Coating: 14.13 g

The manufacturing processes taken into account are "Mounting of PWB" and "Electricity for bake-out", based on [9]. The average SSDR surface is 18000 mm^2 (See Table 3.53), while the mean energy needed for bake-out is $5.837 \cdot 10^{-3}$ KWh, scaled from the data available in [9].
Based on this data, the generic SSDR can then be modelled in SimaPro.

Data Distribution Units

Data distribution is mainly carried out by data buses [42]; however, since every rocket architecture is unique, no modular or standardized data bus system is commercially available, thus making it impossible to model a generic data distribution unit; if their specific features were known, they could be modelled as "electronic components with high calculation or data processing needs", according to ESA's guidelines [9].

3.3.6 Guidance, Navigation and Control System

The GN&C (Guidance, Navigation and Control) system contains the attitude control system and the reaction control systems [42]. Their functions can be carried out through thrust vectoring and the use of control surfaces, controlled and guided by navigation sensors such as an INU (Inertial Navigation Unit) and star trackers [42]. Everything is regulated by the flight computer [42].

Thrust Vectoring

The most common TVC (Thrust Vector Control) methods can be found in Tab. 3.57; they can be applied to solid and/or liquid engines.

Table 3.57: Thrust Vectoring Methods [125]

METHOD	LIQUID	SOLID
Gimbal	X	-
Flexible Laminated Bearing	-	X
Flexible Nozzle Joint	-	X
Jet Vanes	X	X
Jetavator	-	X
Jet Tabs	-	X
Side Injection	X	X
Small Control Thrust Chambers (Vernier Thrusters)	X	-

Gimbals, if present, add approximately 3.6% to the engine mass [72]. They can be considered made of the same material as the engine structure, which is 440b stainless steel [138]. Considering a mean engine mass of 1,064 kg [127, 128, 129, 87, 130, 131, 107, 115, 132, 133, 134, 135, 103, 116, 136, 137, 108], the mean mass of a gimbal mechanism is 38.3 kg. The manufacturing process through which it is produced is "Metal working, average for chromium steel product manufacturing" [10].

A nozzle with a flexible laminated bearing is held in place by a ring with alternating layers of elastomer and sheet metal [125]. Assuming that the ring is placed exactly at the nozzle throat, with an average throat diameter of 0.62 m [99, 120], and that the bearing is made of 10 layers (5 elastomer and 5 metal layers) of 0.254 mm thickness [183] and an assumed 7 cm height, a total amount of 0.19 kg of rubber and of 1.36 kg of stainless steel are needed to manufacture it. The process "synthetic rubber, at plant" [10] is taken into account for both the material and the manufacturing, while for the production of the stainless steel sheet the process "sheet rolling, chromium steel" [10] is taken into account. Since no specific data was found for flexible nozzle joints, but they too are made of alternating layers of elastomers and metals [125], in a first approximation the flexible laminated bearing can be considered a proxy for the nozzle joint.

Jet vanes assemblies comprise of the vanes themselves, and a shroud to which they are attached [184].
The shroud outer diameter corresponds to the engine nozzle exit diameter (D) [184]; it is 0.54D tall [184], 21.6 cm thick [184], and it can be considered made out of the same material as the vanes (C-SiC [185]). Considering a mean diameter of 1.72 m [127, 87, 132, 133, 116, 103, 136, 137, 108, 99], the mass of the shroud is 2525.81 kg.
The chord of the vanes is approximately 0.64D [184], which corresponds to 1.1 m for a mean exit diameter of 1.72 m. Jet vanes have a double-wedge airfoil with blunted edges shape [184]; however, they are considered to be a simple parallelepiped (with dimensions 1.13 x 0.73 x 0.18 m) made of C-SiC [185]: the mass of a single jet vane is therefore 307.36 kg.
The manufacturing process taken into account for the production of both the shroud and the vanes is "Heat, industry supply" [17], as the vanes need to go through several heating processes during production [185]. The total amount of heat required to manufacture the shroud is 13.3 MJ [185], while the heat required for the production of one jet vane is 1.61 MJ [185].

The jet tab assembly is comprised of a steel shaft and support, blade and shaft insulation, refractory metal blade, and refractory screw [186]. The shaft and support are made of cast stainless steel, the insulation is made of glass phenolic material which is compression moulded, and the refractory components are machined from molybdenum [186]. Assuming a square 55.8 x 55.8 cm tab [187], with a support that is 2.54 cm in diameter, two 0.64 cm refractory layers [186] and two 0.51 cm insulation layers [186], the total masses of the different elements can be found in Tab. 3.58.

The manufacturing process taken into account for both the support and the shaft is "Casting, steel" [17], while for the refractory parts "Metal working,

Table 3.58: Jet Tabs Element Masses

ELEMENT	MASS
Support	2.19 kg
Shaft	61.29 kg
Refractory	40.74 kg
Insulation	5.88 kg

average for metal product manufacturing" [10] is employed, due to the lack of more specific processes. Lastly, the insulation is modelled by considering "GFRP, mould, RTM (Resin Transfer Moulding)" [17], as it includes both the necessary materials and the manufacturing.

A jetavator is made up of three laminated layers:

- Heat resistant layer: usually made of tungsten, molybdenum or ceramic [188]
- Heat insulating layer: usually made of CFRP, rubber or ceramic [188]
- Reinforcing layer: usually made of steel or aluminium [188]

Considering a mean nozzle exit diameter of 1.4 m [99], based on [188], the thickness of the heat resistant layer and the reinforcing layer is 0.035 m, while the insulating layer is 0.018 m thick. The height of the jetavator is 1.12 m. The first layer is assumed to be made of molybdenum, the second of CFRP and the third of steel. The masses of the three elements can be found in Tab. 3.59.

Table 3.59: Jetavator Element Masses

ELEMENT	MASS
Heat Resistant Layer	1,762 kg
Insulating Layer	146.04 kg
Reinforcing Layer	1,354.97 kg

The considered manufacturing process are "Metal working, average for metal product manufacturing" [10] for the heat resistant layer, as more specific processes are not available and "Steel removed by turning, average, conventional" [188, 10] for the reinforcing layer. The insulation is modelled by considering "CFRP, cured by autoclave Process" [17], as it includes both the necessary materials and the manufacturing.

Vernier thrusters consist in small engines that can be both next to the main engine or along the rocket body [125] and, as such, can be modelled as shown in Subs. 3.3.2. They will be considered as gas generator engines, although most small thrusters are pressure-fed and thus do not fit into any of the categories mentioned in Subs. 3.3.2. In Tab. 3.60, the specifications of the considered Vernier thrusters can be found.

Table 3.60: Vernier Thrusters Masses

THRUSTER	MASS
R-1E	3.70 kg [189]
LR101-11	22 kg [190]

The mean mass is therefore 12.85 kg, which corresponds to a nozzle extension mass of 4.03 kg, a combustion chamber mass of 2.58 kg, a pumps mass of 3.63 kg, a flow systems mass of 0.5 kg and a structure mass of 2.11 kg. The different elements can then be modelled with the materials and manufacturing processes mentioned in Subs. 3.3.2.

Out of all possible thrust vectoring methods introduced in Tab. 3.57, only the side injection method has not been modelled, as it consists of additional ducts in the engine, which have already been modelled in the Subs. 3.3.2.

Control Surfaces

Control surfaces include fins and wings [42].

The fins are modelled after the four fins on the first stage of Falcon 9 [101], which are made of titanium that is cast in a single piece [191]. The dimensions of the grid fins are approximately 1.2 x 1.5 x 0.25 m [192], considering only 5% of the volume to be actually made of titanium, the total weight of a single grid fin is 101.4 kg. Using the process "Casting, titanium" [17], the generic grid fin can be implemented in SimaPro.

Wings appear on launch vehicles like Blue Origin's New Glenn; however, no public information about their size or mass is publicly available, and, for such reason, they cannot be modelled.

Navigation Sensors

Two types of navigation sensors that can be found on a launch vehicle are INUs and star trackers [42]. However, no available off-the-shelf sensors of either type have been located and, as such, these components have not been implemented. Two possible modelling approximations can be found in Chapter 4.

Computers

The flight computer is assumed to carry out both GN&C and C&DH (Command and Data Handling) functions [179]. The implementation is therefore the one presented in the Subs. 3.3.5.

3.3.7 Emergency Systems

The main types emergency systems on board a launch vehicle are abort systems and fire control systems [42]. There are also redundant systems or components [42], which can simply be modelled in an LCA by choosing more than one unit wherever necessary.

The majority of launch abort system are implemented in manned missions, in order to protect the crew in case of failure; however, they are mostly installed on the capsule rather than the launch vehicle itself and, as such, will not be covered in this document.

Most launch vehicles, whether crewed or not, are equipped with a flight termination system, whose purpose is to destroy the rocket in case of emergency to avoid hurting civilians [193]. The main components of a flight termination system are a receiver (an antenna), a safe and arm device, and a termination system [193]. The antenna has already been modelled in the Subs. 3.3.4, while the safe and arm device can be simplified as a switch. As mentioned in Subs. 3.3.3, the switch is modelled as a toggle switch. The termination system can either comprise of a destruct charge or a linear shaped charge [193].

The main components of a destruct charge are the metal housing, the detonation charge and the transfer lines [194]. Considering a total mass of 1,130 g [194], and assuming that the casing makes up 60% of it, the explosive 30% and the transfer lines 30%, the following quantities and materials can be considered:

- Stainless Steel [194]: 678 g
- PBXN-5 Explosive Load [194]: 339 g
- Transfer Lines [194]: 113 g

Stainless steel 304 has been chosen, while both the transfer lines and the PBXN-5 have been considered as made of 95% RDX (cyclotrimethylenetrinitramine, $C_3H_6N_6O_6$) explosive crystals and 5% of polyurethane, which acts as a binder [195]. Due to the lack of more specific information, the manufacturing process for the stainless steel casing is considered to be "Metal Working, average for Chromium Steel Product manufacturing" [10].

A linear shaped charge consists of an explosive, encased in a metal sheath liner [196]. The metal taken into account is aluminium, while the explosive is RDX [196]. A mean length of 1.3 m can be considered [197], to which corresponds an average explosive mass of 263.25 g [197]. It is assumed that the explosive makes up 30% of the weight, while the casing the remaining 70%; thus, to model the charge, 614.25 g of aluminium are needed. Due to the lack of more specific information, the manufacturing process taken into account for the metal sheath is "Metal Working, average for Aluminium Product manufacturing".

Chapter 4

Ariane 6 LCA Goal, Scope and LCI Definition

The aim of this section is to describe the first two phases of a complete cradle-to-use LCA study of the Ariane 6 launcher, which has been chosen as the real case study due to the attention to sustainability posed on it, from its design to the production and use phases [198].
The first steps to perform an LCA are to define the goal and scope of the study, its functional unit (a quantified performance of a product system for use as a reference unit), and the system boundaries [3]. The following step is to then perform the LCI data collection and the necessary calculations [3]; whenever possible, the use of secondary data from the database developed in Chapter 3 will be implemented.

4.1 Goal and Scope

The objective of this study is to understand the environmental hot spots in the cradle-to-use part of the life cycle of a launcher, and to identify eventual phases with a negligible impact, which can therefore be neglected in a simplified first approach. This type of analysis not only helps characterise the areas on which it is important to focus, but, when compared to other similar studies, allows to draw conclusions on the most sustainable options available, thus contributing to the overall sustainable development of the space sector.

As mentioned in Chapter 2, the lack of specific data poses limitation to the accuracy of the implemented models; moreover, the exclusion of some processes, as explained in Sec. 4.1.2, could influence the overall results and potentially lead to the wrongful omission of some environmental hotspots. Nonetheless, this study can be considered a first approximation in identifying the major contributors to the environmental impact of a launcher and, as such, can be helpful to companies or LCA practitioners.

4.1.1 Functional Unit

This LCA study has been divided into two different parts: the first one deals with analysing the impact testing and other life cycle phases have on the overall program, comparing the processes implemented in Sec. 4.3 to 117 Ariane 6 launches, assuming the launcher completes the same amount as Ariane 5 did [199]. Since the Ariane 62 configuration with the short fairing was used for testing, the same architecture has been chosen to evaluate the production, transportation and launch impacts. The second section deals with comparing the impact of the four different possible architectures. The functional unit chosen for each architecture is "to place a kg of payload into a Low Earth Orbit". This way, the impacts are normalised and can be accurately compared, while a choice of a single launch as the functional unit would automatically lead to smaller architectures being more sustainable, as they require less components.

4.1.2 Boundaries

This section will explain which processes have been included in the system studies [3], and the reasons for which some phases were excluded.

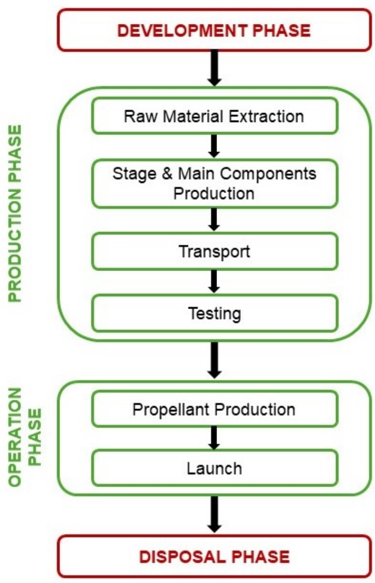

Figure 4.1: Ariane 6 LCA Boundaries

In Fig. 4.1, the implemented processes can be seen. A green border means that the phase has been implemented, while a red border means that it has been neglected. The development phase, which includes office works, business trips and testing and qualification of smaller components [11] has not been implemented due to the lack of data, while the disposal phase falls outside the scope of the study. No infrastructure has been implemented, as it is assumed that its impact is negligible compared to the rest of the program; moreover, buildings and control centres are usually employed for more than one launcher program, and there is no precise rule on how to split their impact over all possible programs. Should their impact be quantified, it would be more sensible to perform a separate LCA study that only includes infrastructure.

4.2 Vehicle Components Inventory

This section will deal with implementing Ariane 6's main components, following the structure implemented in Chapter 3. It is important to note that, while some subsystems can be modelled with a rather high degree of accuracy, the same cannot be said about avionics. However, due to their large use of rare minerals, avionics can be the most impactful components in some impact categories, and, as such, they cannot be neglected in an LCA analysis. Therefore, two different approaches have been undertaken and compared: the first one consists in firstly implementing all the known avionic components on board Ariane 6, and then modelling all the remaining unassigned mass as 20% aluminium casing and 80% electronic units, as in [200]. Subsections 4.2.3, 4.2.4, 4.2.5, 4.2.6 and 4.2.7 will introduce the known components that make up each subsystem, while Subs. 4.2.8 will deal with implementing the unknown avionics with the aforementioned approximation, and compare it to the second approach. The latter consists in considering the overall mass of the avionics, which is then distributed between the main components.

4.2.1 Structure

Main Structure

In Tab. 4.1, the structural components that make up the four different possible architectures can be found.

Table 4.1: Ariane 6 Structural Components

	ARIANE 62 SHORT FAIRING	ARIANE 64 SHORT FAIRING	ARIANE 62 LONG FAIRING	ARIANE 64 LONG FAIRING
COMPONENT	UNITS			
Lower Stage - Heavy Lift Launcher	1	1	1	1
Interstage (GFRP) - Heavy Lift Launcher	1	1	1	1
Upper Stage (Aluminium) - Heavy Lift Launcher	1	1	1	1
Booster, > 1,000 kN Thrust	2	4	2	4
Fairing (GFRP) - Heavy Lift Launcher	1	1	-	-
Long Fairing[1] (GFRP) - Heavy Lift Launcher	-	-	1	1

[1] A new process consisting of 2,600 kg of "GFRP, cured by Autoclave" [17, 117] has been created, as the long fairing option was not implemented in Chapter 3.

Mechanisms

A single unit of frangible joint is taken into account to model the separation mechanisms between the first and second stage [97].

The separation mechanism between the second stage and the payload fairing is made of twelve metallic springs [97]; however, no information on their sizes, composition or manufacturing processes is publicly available and, as such, they cannot be modelled.

4.2.2 Propulsion System

The three components that need to be implemented in the propulsion system are the first stage engine (Vulcain 2.1 [96]), the second stage engine (Vinci [96]) and the engine igniters.

Vulcain 2.1 produces 1,371 kN of thrust [201] and, as such, falls under the category "> 1,000 kN Thrust, Gas Generator" presented in Chapter 3; its corresponding components can therefore be selected.

The engine also features an innovative heat shield design, with thermal tiles made out of Prosial, a silicon based material [202]. Since no in-depth information about the composition of the heat shield is publicly available, it has been modelled as "Silicon product, at plant" [10]. To calculate the necessary amount of silicon to cover the Vulcain 2.1 engine, a surface of 17 m^2 has been considered [17], with an average thickness of 2 cm [202]; the total mass of the heat shield is therefore 792.2 kg.
No ignition mechanism or controller have been considered, as Vulcain 2.1 is ignited from the ground [201]. There is no difference between the Ariane 62 and Ariane 64 architectures.

The Vinci engine produces 180 kN of thrust [201, 96], and, as such, falls under the category "100 - 1,000 Thrust, Expander" presented in Chapter 3, whose corresponding components can be selected to model the engine.

The Vinci engine is also characterised by the use of an APU (Auxiliary Propulsion Unit) [97], which allows for multiple re-ignitions that help the second stage put different payloads in different orbits. Since this type of component normally does not make up rocket engines, it has not been modelled in Chapter 3. As no specific information on its size, material or manufacturing processes is available, it is assumed that it is part of the "Engine Structure" components, although this poses some limitations to the accuracy of the implemented model, as it does not accurately depict all components, and therefore likely underestimates its environmental impact.

On its maiden flight, Ariane 6 carried five PFU (Pyrotechnic Firing Unit)

[203]. It is assumed that three of them are jointly carried by the first and second stage, while each booster carries one. Therefore, the total amount of PFUs needed for Ariane 62 is five, while for Ariane 64 it is seven. The maximum mass of Ariane's PFU is 11 kg [204]; it can therefore be modelled as half of the ignition and staging controller introduced in Chapter 3, for a total amount of 2.5 and 3.5 controllers for the Ariane 62 and Ariane 64 architectures. An equivalent amount of pyrotechnic powder doses is needed to completely model the ignition/staging controllers.

4.2.3 Power System

Power storage in the core stage is assumed to be completely carried out by Li-Ion batteries, as modelled in Chapter 3. Based on [148], a total amount of 30 batteries is assumed to be on board Ariane 6, 20 of which are considered part of the first stage, and the remaining 10 as part of the second stage. Since the batteries are available in two configurations (small and large [148]), it assumed that the batteries of each stage are equally distributed between the two. Therefore, considering an overall battery mass of 120 kg for the first stage, and of 60 kg for the second stage, a total of 57 of the battery units modelled in Chapter 3 are used to perform the LCA. Since the boosters employ a different kind of battery, there is no difference between the Ariane 62 and Ariane 64 architectures.

The P120C boosters use two thermal battery modules each [205]; each module has a mass of 12.3 kg [205], and it needs to be specifically modelled for Ariane 6, as thermal batteries have not been developed among the launch vehicle components in Chapter 3. Based on [206], the necessary materials to model a thermal battery are:

- Steel (11.24 kg), which is hot rolled;
- Filler wire (0.78 kg), which is arc welded;
- Rock wool (0.28 kg), for insulation.

Once the module has been implemented, four units can be added to the Ariane 62 architecture, while eight can be added to Ariane 64.

Power distribution is carried out through wires and cables, and is handled by the power distribution unit. On Ariane 6, the PDU and flight computer are integrated into a single unit called CMFU (Centralized Multi-Functional Unit) [204]. Since the maximum mass of the CMFU is 17.5 kg [204], while the mean masses of the OBC and PDU are, respectively, 5.44 kg and 0.79 kg (as seen in Chapter 3), a single CMFU has been modelled as three flight computers and two power distribution units. Based on [203], it is assumed that a total of five CMFUs are on board the launch vehicle, with no difference between the Ariane 62 and Ariane 64 architectures.

As mentioned in Chapter 3, the total mass of the wiring for the lower and upper liquid stages of Ariane 6 is respectively 1,046.64 kg and 368.92 kg; the total mass of wiring in Ariane 6's core is therefore 1,415.56 kg. As estimated in Section 3.3, the wiring mass in each P120C booster is 156.5 kg; the Ariane 62 architecture therefore has a total amount of 1,728.56 kg of wiring, while the Ariane 64 architecture has 2,041.56 kg. Based on [200], the wiring is completely modelled as "Cable, ribbon cable, 20-pin, with plugs".
The implementation of the remaining components with both the avionics assumptions is discussed in Subs. 4.2.8.

4.2.4 Communication System

As no information on the exact amount of both radios and antennas on board Ariane 6 is available, for the first avionics approximation, one radio has been taken into account for each of the following functions: telemetry, flight termination system and communication with payload. A total of six radios has been implemented, with a double redundancy for each function.

One antenna has been taken into account for each of the following functions: telemetry, command, flight termination system, tracking and communication with payload. The flight termination antenna has already been included in the complete system and, as such, will not be taken into account in this section. Thus, a total of eight antennas has been implemented, with a double redundancy for each function.

4.2.5 Guidance, Navigation and Control System

Both the boosters and the first stage nozzles can be gimballed [207, 208]; as such, three and five units of "Gimbals", for the Ariane 62 and Ariane 64 architectures, respectively, have been considered. No information on eventual thrust vector control systems for the Vinci engine has been found, thus no form of TVC has been implemented for the second stage.

On Ariane 6, neither fins nor wings can be found [97]; no control surfaces have therefore been modelled.

The flight computer has been modelled in Subs. 4.2.3, as it is part of the CMFU. No additional computers need to be modelled for this section.

The implementation of the remaining components, with both the avionics assumptions, is discussed in Subs. 4.2.8.

4.2.6 Command and Data Handling System

The command computer, whose function is considered to be carried out by the flight computer, has been modelled in Subsec. 4.2.3. Therefore, no additional computers need to be modelled for this section.
The implementation of the remaining components,such as data storage or distribution units, with both the avionics assumptions, is discussed in Subsec. 4.2.8.

4.2.7 Emergency Systems

Each Ariane 6 booster is equipped with a linear shaped charged device flight termination system [197]; the Ariane 62 architecture therefore has two, while Ariane 64 has four. No redundancy is assumed for this system.

Due to the lack of more specific information, it is assumed that the lower and upper stages of Ariane 6 employ a different kind of flight termination system, based on commands received from the ground stations to the main flight avionics.

4.2.8 Other Avionics

As mentioned in Sec. 3.3, the estimated mass of avionics on board the core of Ariane 6 is 1,215.3 kg, while each booster carries approximately 156.5 kg.

In order to carry out the first approximation, where the unknown avionics mass is divided into 80% electronic units and 20% aluminium casing, it is necessary to calculate the total mass introduced in the previous sections, and subtract it from the aforementioned quantities. For the Ariane 62 architecture, the total mass of the unknown avionics corresponds to 1,185.25 kg, while for the Ariane 64 architecture it is 1,445.06 kg. In Tab. 4.2, the processes and their respective quantities taken into account can be found.

Table 4.2: Other Avionics Component Masses

	ARIANE 62	**ARIANE 64**
Aluminium Alloy, AlLi [200]	237.05 kg	289.01 kg
Electronics, for Control Units [200]	948.2 kg	1,156.05 kg

To carry out the second approximation, the main subsystems taken into consideration are: power, communication, guidance, navigation and control, command and data handling, propulsion, structure and emergency systems. The total avionics mass for Ariane 62 is 1,528.3 kg, while for Ariane 64 it is

1,841.3 kg. The following components from each subsystem have been taken into account:

- Power System: Li-Ion batteries, thermal batteries, Power Distribution Unit, fuses, transformers, EMI filters and DC-DC converters;
- Communication System: radios and antennas;
- Guidance, Navigation and Control: flight computer;
- Command and Data Handling: Solid State Data Recorder (the flight computer has already been included in the GN&C system);
- Propulsion System: ignition controller;
- Structures: stage separation controller;
- Emergency Systems: linear shaped device flight termination system.

Fourteen components have therefore been considered, each of them having an allocated mass of 109.16 kg in the Ariane 62 architecture, and of 131.52 kg in the Ariane 64 architecture.

In Fig. 4.2, the environmental impact of each of the two assumptions applied to one Ariane 62 launcher can be found. As expected, the highest impact in each case is in the "ADEPLmu (Abiotic Resource Depletion)" [17] category, as avionics contain a high amount of rare materials. However, it can clearly be seen how considering only the main components leads to a far higher impact than considering only the casing and electronic units mass. Since it cannot be determined which assumptions more closely resembles the real impact, a conservative approach is undertaken, considering the main components only assumption for the simulations, even though its higher impact might offset other components or life cycle phases impacts.

4.3 Testing

The main testing steps taken into consideration are those presented in Chapter 3, Sec. 3.1. Since specific information on pressure vessel, bake and environmental tests is not available, Subsections 4.3.2, 4.3.3 and 4.3.4 will be carried out in order to perform a sensitivity analysis, whose goal is to determine the extent of the limitation excluding those processes poses on the overall analysis.

4.3.1 Thrust Measuring Tests/Static Fire Tests

To test the Vulcain 2.1 engine, two demonstration models have been built [209], which, combined, totalled 13,798 seconds of operation [210]. In order to model

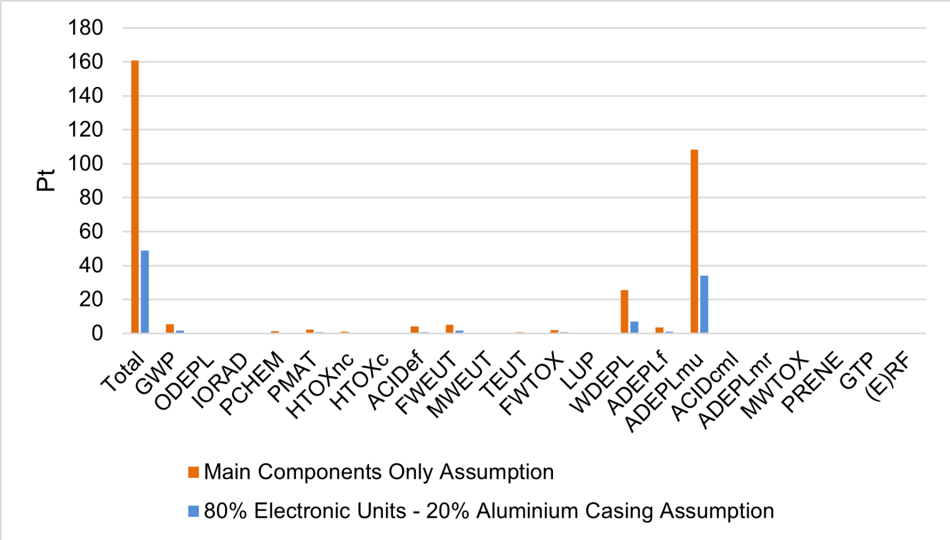

Figure 4.2: Avionics Sensitivity Analysis Results

all impacts due to testing, the production of two ">1,000 kN Thrust, Gas Generator" engines has been taken into account; the amount of LO_x and LH_2 burnt during testing can be calculated through the mass flow rate of the engine (327 kg/s [201]), leading to a total of 4,511,946 kg of propellant being burnt during testing, the production of which is also included in the LCA.

The demonstration model of the Vinci engine has totalled 52,156 seconds of operation during testing [210]. It is assumed that all tests have been carried out by a single model (100 - 1,000 kN Thrust, Expander); however, a model upper stage has also been built to carry out the tests [210], and its production needs to be accounted for in the LCA. During flight, the Vinci engine burns 30,000 kg of LO_x and LH_2 over 900 seconds of operation [201], leading to a mass flow rate of 33.33 kg/s. The total propellant burnt during testing is therefore 1,738,359 kg.

Due to the lack of specific information regarding Ariane 6, it is assumed that its boosters undergo the same number of static fire testing as Ariane 5's. Ten tests of a single boosters are therefore considered [211], with a total amount of 1,436,000 kg of solid propellant being burnt.

4.3.2 Pressure Vessel Testing

Ariane 6's core tanks have the following capacities (See Appendix A):

- Lower stage LO_x Tank: 109.55 m^3
- Lower stage LH_2 Tank: 352.61 m^3
- Upper stage LO_x Tank: 21.91 m^3
- Upper stage LH_2 Tank: 70.52 m^3

The respective volume of each of the tanks has been considered for the following processes [17]: volumetric capacity examination, vibration test, total design loads inspection, proof pressure inspection, pressure cycle inspection, negative pressure inspection, leak test (with water), external leak inspection, expulsion efficiency inspection. To these, the following non-destructive tests need to be added: x-ray/radiographic inspection, acoustic [LEAF (Large European Acoustic Facility)] and ultrasonic inspection [17].

The functional unit of the x-ray inspection is m^2; it is therefore necessary to know the total surface being tested. As a first approximation, 580 m^2 can be considered for the lower stage, while 203.6 m^2 can be taken into account for the upper stage, as introduced in Chapter 3, for a total of 783.6 m^2.

Five acoustic tests have been carried out on the upper stage [212], whose duration is assumed to be equal to the first stage burning time, which corresponds to 470 seconds [97]. A total of 39.17 minutes of the corresponding process can therefore be considered.

Lastly, as the functional unit for the ultrasonic inspection is m^3, the same volumes considered for the destructive tests can be taken into account.

4.3.3 Shake and Bake Tests

As no data regarding Ariane 6's vibration test is available, SLS's testing schedule has been used as a proxy; thus, 50 hours of vibration testing can be considered [213].

Since the vibration test spanned over a week, it can be assumed that the vacuum tests, both in hot and cold environments, lasted five days each.

4.3.4 Environment Tests

As no information is available on neither the duration of the tests or the size of the facility, this testing step will be excluded from the simulations, thus posing some limits on the accuracy of the results related to the testing phase.

4.3.5 Roll Out Tests

The impact of the roll out test is equivalent to the one of the actual roll out, which can be found in Subs. 4.4.3.

4.3.6 Full Rehearsal and Flight Tests

A full model of the Ariane 62 launcher has been built and transported to the launch pad. The only difference from the real launch vehicle lies in the boosters not being fuelled. Therefore, the production of a complete launcher needs to be taken into account for testing. The most environmentally relevant tests performed on the full-scale model are the following [214]:

- Four second Vulcain firing test, 5 September 2023;
- Full scale rehearsal with seven minute Vulcain firing, 23 November 2023;
- Ignition of Vulcain in degraded conditions (assumed to last four seconds), 15 December 2023.

The total propellant needed to run these tests is 139,956 kg.

4.3.7 Testing Sensitivity Analysis

In Fig. 4.3, the results of the sensitivity analysis, performed by including and excluding the testing assumptions presented in Subsections 4.3.2 and 4.3.3, can be seen.

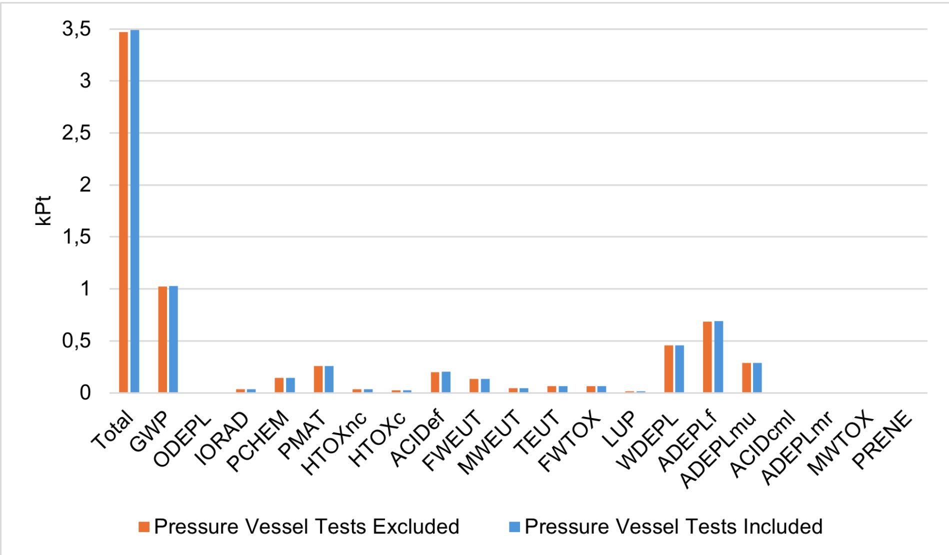

Figure 4.3: Testing Sensitivity Analysis Results

It is clear how the tests carried out on the tanks have a minimal impact compared to the rest of the testing steps; therefore, although their amount could be higher than the one assumed, it can be concluded that neglecting them does not pose a great limitation to the accuracy of the results related to this phase.

4.4 Transportation

The three main transport segments taken into account are:

- Transport of the avionics and other small scale components from the production centre to the integration centre;

- Transport of the main components (liquid propulsion stages, boosters and fairing) from the production centre to the vehicle assembly building;
- Transport of the fully stacked vehicle from the assembly building to the launch pad.

4.4.1 Avionics & Small Scale Components

This section deals with the transportation of the components implemented in Subs. 4.2.8, alongside the interstage structure. The known production centres are:

- Madrid, Spain: CMFU (power distribution unit and flight computer) [204], radios [215], PFU (ignition controller and staging controller) [204];
- Barcelona, Spain: antennas [216];
- Les Mureaux, France: flight termination system [197];
- Getafe, Spain: interstage structure;
- Toulouse, France: batteries [148]. Assumed as production centre for avionics whose origin is unknown.

The integration centres are [217]:

- Le Havre, France: lower stage;
- Bremen, Germany: upper stage;
- Bordeaux, France: boosters.

In Tab. 4.3, all the possible transportation routes starting and ending points, alongside the total distance travelled, can be found. Whenever more than one route is available, the shortest one is chosen, as it usually corresponds to the lowest propellant consumption and thus to the lowest environmental impact. Trucks are assumed to be the main mean of transport; since no specific information is available, the process "Transport, freight, lorry, unspecified" [10] is taken into consideration.

In the Ariane 62 architecture, booster avionics make up 20.5% of the total avionics mass, while the upper stage is responsible for 28.6% and the lower stage for the remaining 50.9%. In the Ariane 64 case, boosters take up a higher percentage (34%), with the upper stage making up 23.8% of the avionics mass, and the lower stage the remaining 42.2%. For each of the fourteen components that have been implemented, it is then assumed that the corresponding aforementioned percentage is destined to its respective main components; for

Table 4.3: Ariane 6 Avionics Transportation Routes Characteristics

PRODUCTION CENTRE	INTEGRATION CENTRE	DISTANCE TRAVELLED
Madrid, Spain	Le Havre, France	1,338 km [218]
Madrid, Spain	Bremen, Germany	2,054 km [218]
Madrid, Spain	Bordeaux, France	682 km [218]
Barcelona, Spain	Le Havre, France	1,197 km [218]
Barcelona, Spain	Bremen, Germany	1,657 km [218]
Barcelona, Spain	Bordeaux, France	569 km [218]
Les Mureaux, France	Le Havre, France	163 km [218]
Les Mureaux, France	Bremen, Germany	824 km [218]
Les Mureaux, France	Bordeaux, France	595 km [218]
Getafe, Spain	Le Havre, France	1,386 km [218]
Toulouse, France	Le Havre, France	825 km [218]
Toulouse, France	Bremen, Germany	1,472 km [218]
Toulouse, France	Bordeaux, France	595 km [218]

example, 50.9% of the flight computer mass, which corresponds to 55.56 kg in the Ariane 62 case, will be transported from Madrid (its production centre) to Le Havre (lower stage integration centre).

By applying this assumption , a total of 4,029.98 tkm (transportation of 1 metric ton across the distance of 1 km [10]) of transport by truck need to be considered for Ariane 62, while 4,226.25 tkm are applied to Ariane 64.

4.4.2 Main Components

The main components of Ariane 6 are transported from Europe to French Guiana mostly by using the "Canopée" ship; only the last segment is carried out by trucks. The itinerary followed by the ship can be found in Tab. 4.4. Assuming the ship travels at maximum speed all the time (31 km/h [217]), it takes 293.37 hours to complete the journey. Each of the two diesel engines that power the ship has 3,840 kW of power [217], meaning that $8.11 \cdot 10^6$ MJ of diesel are burnt each time a new rocket is built and transported to the launch pad. This process is modelled through "Diesel, burned in fishing vessel" [10], as it is the closest option.

Table 4.4: Ariane 6 Main Components Transportation Routes Characteristics

COMPONENT	PICK-UP POINT	FOLLOWING STOP	DISTANCE TRAVELLED
Upper Stage	Bremen, Germany [217]	Rotterdam, Netherlands [217]	537.08 km [219]
Fairing	Rotterdam, Netherlands [217]	Le Havre, France [217]	457.44 km [219]
Lower Stage	Le Havre, France [217]	Bordeaux, France [217]	996.38 km [219]
Boosters	Bordeaux, France [217]	Kourou, French Guiana [217]	7,103.69 km [219]

The distance travelled by the trucks from the port in Cayenne to the Guiana Space Centre is 79.2 km [218]. Due to the lack of more specific information, the process chosen to model this transportation segment is "Transport, Freight, lorry, unspecified" [10]. Since the functional unit of the process is "tkm", it is necessary to determine the total mass of the transported components. Tables 4.5, 4.6 and 4.7 respectively present a summary of the unfuelled masses of the different components, the total mass that needs to be transported for all possible combinations, and the total amount of tkm to be implemented for all possible combinations.

Table 4.5: Ariane 6 Main Components Dry Masses

COMPONENT	DRY MASS
Core Stage	23,000 kg [96]
Upper Stage	5,100 kg (See Chapter 3, Subsection "Structure - Heavy Lift Launch Vehicles")
Booster (Single Unit)	11,000 kg [122]
Short Fairing	1,800 kg [117]
Long Fairing	2,600 kg [117]

Table 4.6: Ariane 6 Possible Transport Combinations

	ARIANE 62	ARIANE 64
SHORT FAIRING	51,900 kg	73,900 kg
LONG FAIRING	57,700 kg	74,700 kg

4.4.3 Transportation to Launch Pad

Ariane 6 is transported to the launch pad in two parts: the core (made from the lower and upper stage joined together), which is assembled in the launch vehicle assembly building, and the boosters, which are fuelled inside the booster finishing facility [220, 221]. The fairing is then added to the top of the rocket before the launch [220].
The main buildings that make up the Kourou Space Centre can be seen in Fig. 4.4.
The core is transported by autonomous guided vehicles, which follow a rail on the 800 meter road that separates the assembly building from the launch pad, where the core is then lifted to stand upright [220]. As no process exists to model an AGV (Autonomous Guided Vehicle) in LCAs, "Transport, freight train" [10] has been chosen to model this transport phase; in order to obtain the amount of tkm for each possible architecture, the masses in Tab. 4.7 need to be multiplied by 0.8 km.

The boosters are transported from the Booster Storage Building to the launch pad directly [221]; it is assumed that the distance between the two is 800 m [220]. In order to transport the P120C boosters, a special truck has

Table 4.7: Ariane 6 Total tkm for All Possible Combinations

	ARIANE 62	ARIANE 64
SHORT FAIRING	4,110.480 tkm	5,852.880 tkm
LONG FAIRING	4,569.840 tkm	5,916.240 tkm

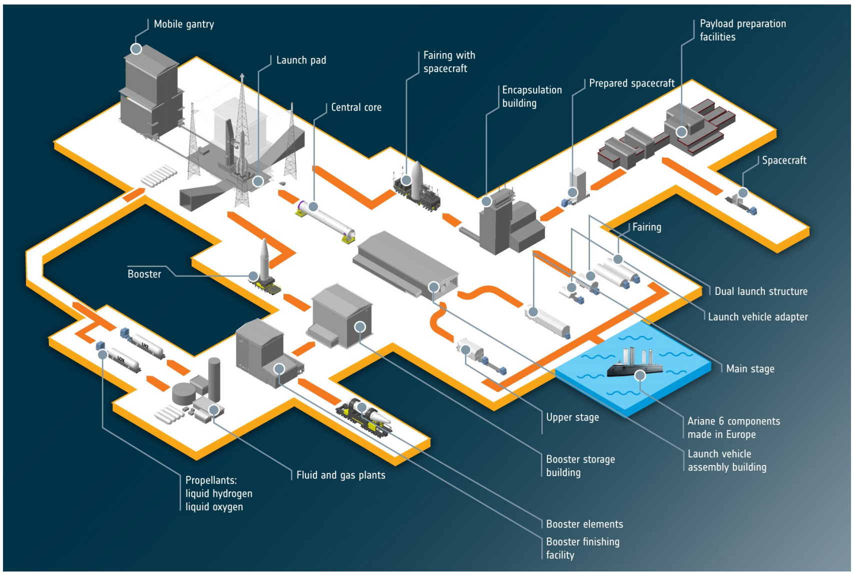

Figure 4.4: Kourou Space Centre Launch Site Assembly [222]

been designed [220]: it is powered by two 235 kW Volvo engines and two 566 kW Scania engines, all of which run on diesel [221]. By knowing that its speed when loaded is 15 km/h, a total transportation time of 192 s for each booster can be calculated. Considering a total truck power of 1,602 kW, 307.584 MJ are necessary to carry each of the boosters from their storage centre to the launch pad. Therefore, a total of 615.168 MJ of the process "Diesel, burned in agricultural machine" [10] can be taken into account for Ariane 62, while 1,230.336 MJ can be taken into account for Ariane 64. The process has been considered as a proxy due to the agricultural machines having the same loaded and unloaded speeds as the booster-carrying truck, even though the latter has been specifically built for this function [221], and cannot be associated to any traditional process.

The fairing is assumed to be transported from the encapsulation centre to the launch pad, which is also assumed to be 800 meters away from the launch pad. Since no information is available on the type of transport employed in this case, the process "Transport, Freight, lorry, unspecified" [10] is chosen. The two possible fairing masses can be found in Tab. 4.6.

4.5 Launch

The launch event is characterised by three main phases:

1. Vulcain 2.1 burn, which consumes 154,000 kg of liquid propellant [201];
2. Boosters burn, parallel to the Vulcain burn, which consumes 143,600 kg of solid propellant for each booster [122];
3. Following the Vulcain shut-off, the Vinci engine burns 30,000 kg of liquid propellant.

Each event can be modelled through the processes implemented in Sec. 3.2; however, the emissions produced by the upper stage have not been included in the LCA, as the combustion happens in the vacuum of space and it does not directly pollute Earth's atmosphere. Although it is not yet clear if this assumption is completely correct, studies tend to follow this approach rather than including wrongly characterised emissions and thus overestimating the impact of the propellant combustion in upper stages [223].

During the launch event, about 700 m^3 of water are employed by water deluge systems [224]. These are modelled through the process "Water, unspecified natural origin" [10], as modelled in Chapter 3.

Chapter 5

Ariane 6 LCA Impact Assessment and Result Interpretation

This chapter will present and discuss the results of the analysis introduced in Chapter 4. The outcomes will be presented in terms of midpoint indicators, as problem-oriented approaches (rather than damage-oriented, like endpoints) are more suitable to space related studies [3].

5.1 Complete Program Impact Analysis

The following analysis will present the results of the LCA, assessing and comparing the impacts in 22 different environmental categories [17]:

- Climate Change (GWP)
- Ozone Depletion (ODEPL)
- Ionising Radiation (IORAD)
- Photochemical Ozone Formation (PCHEM)
- Particulate Matter (PMAT)
- Human Toxicity, Non-Cancer (HTOXnc)
- Human Toxicity, Cancer (HTOXc)
- Acidification (ACIDef)
- Eutrophication of Freshwater (FWEUT)

- Marine Eutrophication (MWEUT)
- Terrestrial Eutrophication (TEUT)
- Ecotoxicity of Freshwater (FWTOX)
- Land Use (LUP)
- Water Use (WDEPL)
- Resource Use, Energy Carriers (ADEPLf)
- Resource Use, Mineral and Metals (ADEPLmu)
- Acidification (ACIDml)
- Resource Use, Mineral and Metals (ADEPLmr)
- Marine Aquatic Ecotoxicity (MWTOX)
- Primary Energy Consumption (PRENE)
- Global Temperature-Change Potential (GTP)
- (Effective) Radiative Forcing ((E)RF)

In Fig. 5.1, the results of the characterisation analysis of the whole program can be seen.

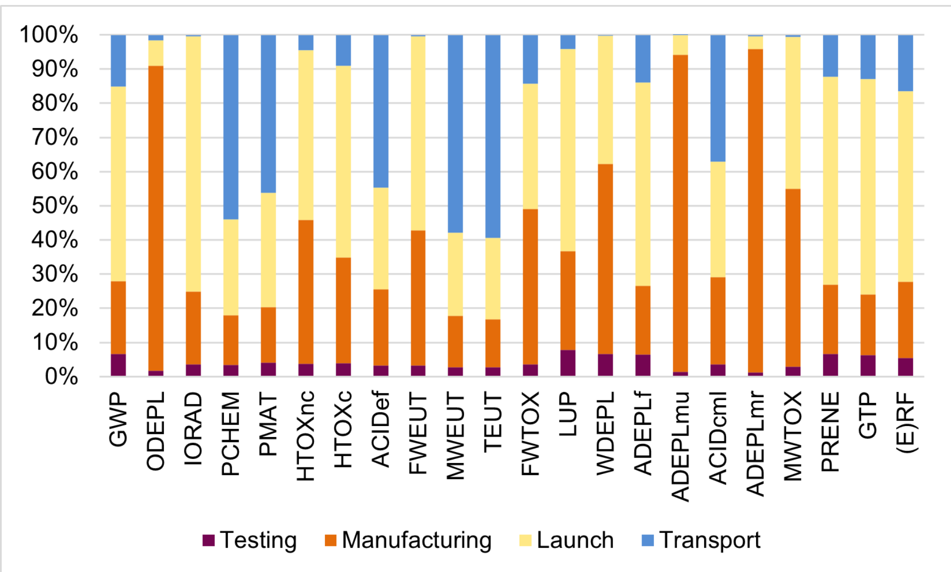

Figure 5.1: Ariane 6 Program LCA Results Characterisation

This type of analysis helps define the environmental hotspots and negligible impacts in all possible categories: if a threshold of 5% is taken into account to consider an impact negligible [9], it can be observed how only the testing and transport fall below this value in some categories, while the manufacturing and launch phases can be considered the major hotspots.

In Fig. 5.2, the most impacted categories from the overall program can be observed.

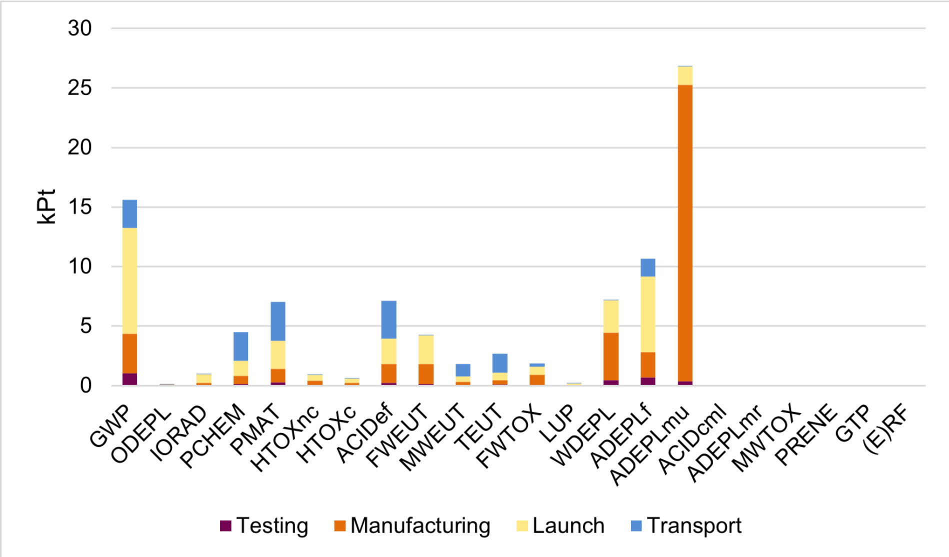

Figure 5.2: Ariane 6 Program LCA Results Weighting

The weighting of the results allows for a comparison between impact categories, therefore helping define not only the major environmental hotspots, but on which ecological category mitigation efforts should focus. The most impacted category is "Resource use, minerals and metals" [17], whose impact the launcher manufacturing is mainly responsible for. The category with the second highest impact is "Global Warming Potential" [17], which is mostly affected by the launch event. Other categories, such as resource use (energy carriers), water use, particulate matter formation and acidification also have a considerable impact, while most of the remaining categories can be considered negligible.

5.1.1 Testing

By looking at Fig. 5.1, it is clear how testing has a minor impact on the overall program: if the aforementioned threshold of 5% is taken into account, it can be seen how testing falls under that value in most categories, and, even when its impact is slightly higher, it never makes up more than 8% of the overall value. It can therefore be said that the impact of this phase is spread out over a high number of launches, and thus does not pose a great environmental concern.

By performing an iterative analysis, the minimum number of launches that are necessary to spread out the impact of testing can be identified: in the Ariane 6 case, after 73 launches, the impact due to testing in most of the categories falls over the 5% threshold (Fig. 5.3). Therefore, at least 74 launches are needed to ensure the testing phase is not one of the biggest environmental drivers in the launcher program.

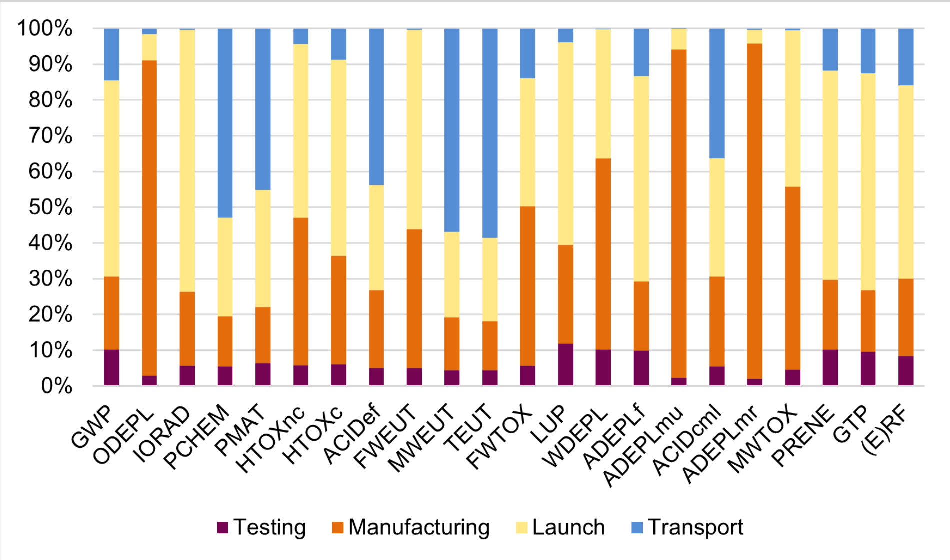

Figure 5.3: Ariane 6 Testing Impact Threshold Analysis

Although the impact of testing can mostly be considered negligible, it is worth noting how the propellants employed in static fire tests make up the majority of the impact (Fig. 5.4); therefore, the considerations introduced in Subs. 2.4.2 on the importance of developing green propellants could help lower the impact of testing even more.

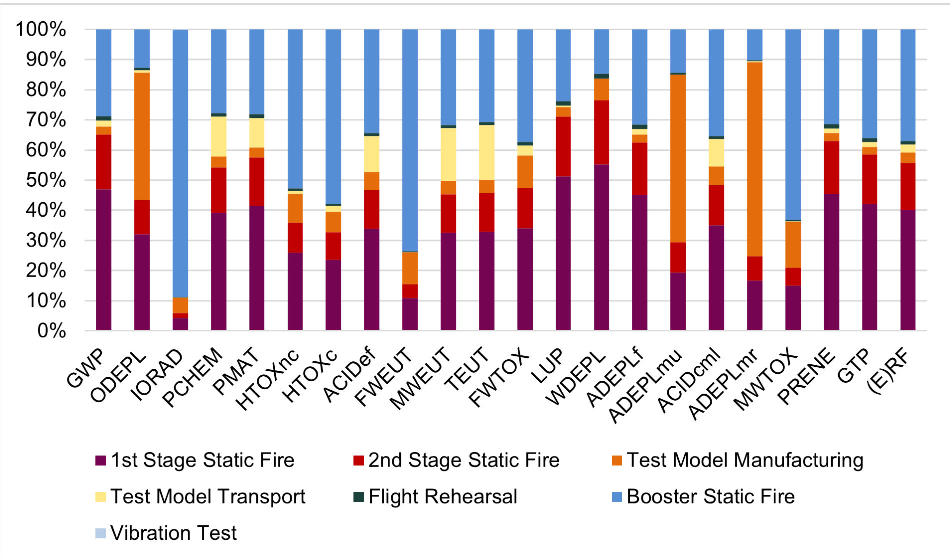

Figure 5.4: Ariane 6 Testing Characterisation

5.1.2 Manufacturing

As mentioned before, the most impacted category overall is "Resource use, minerals and metals" [17], for which the manufacturing is mainly responsible. The production is also the most impactful phase in terms of ozone depletion, freshwater ecotoxicity, water use and marine aquatic ecotoxicity. In Fig. 5.5, the impact share of the production of each component can be found. The

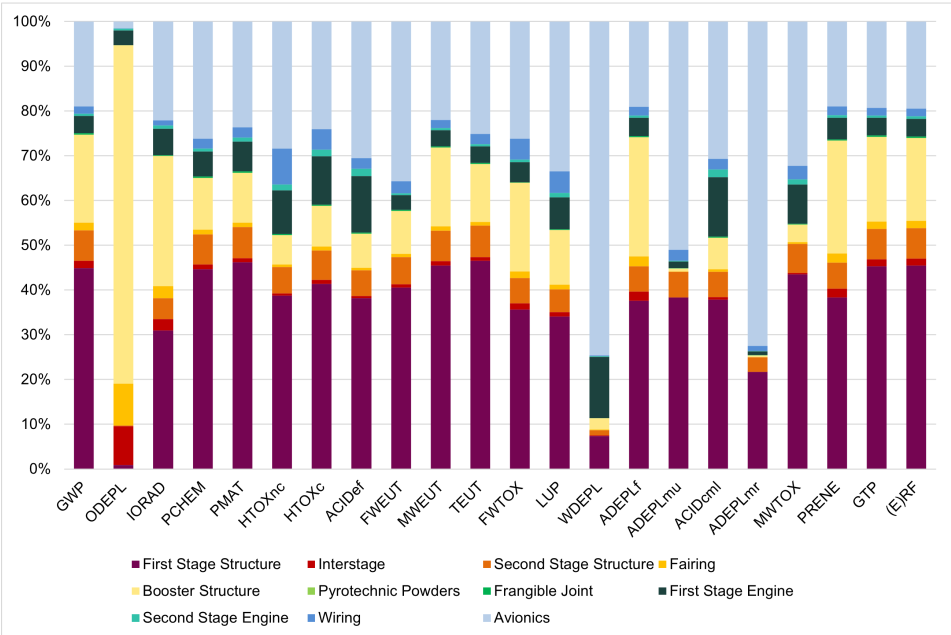

Figure 5.5: Ariane 6 Manufacturing Characterisation

manufacturing of the avionics, as expected, is the main responsible in the resource use category, as seen in Subs. 4.2.8. Avionics, alongside the upper stage and booster structures, are clearly the biggest environmental hotspots in manufacturing: this is likely due to both their mass and the type of material they employ, as smaller structures (like the upper stage one) do not impact as much and do not employ rare materials like avionics do.

In Fig. 5.6, the impact characterisation of the avionics production can be seen.

As expected from the approximation carried out in Subs. 4.2.8, the impacts from almost all the components are comparable. However, it can be seen how fuses have a slightly higher impact than the rest, while components such as both thermal and Li-Ion batteries, and the flight termination system, have a negligible impact compared to the rest. Therefore, it can be concluded that components which include memory and IC units, like the PDU and flight computer, or minerals, like fuses, have a much higher impact compared to those made of simpler units.

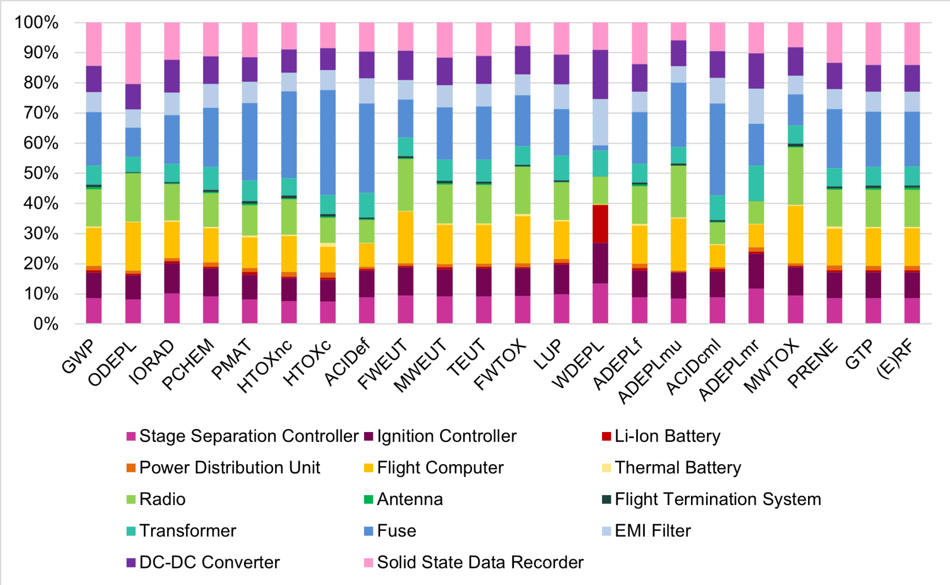

Figure 5.6: Ariane 6 Avionics Manufacturing Characterisation

5.1.3 Transport

The transportation phase is responsible for the highest impacts in the photochemical ozone formation, particulate matter, acidification, marine and terrestrial eutrophication categories. In Fig. 5.7, the impacts related to each transportation phase can be found.

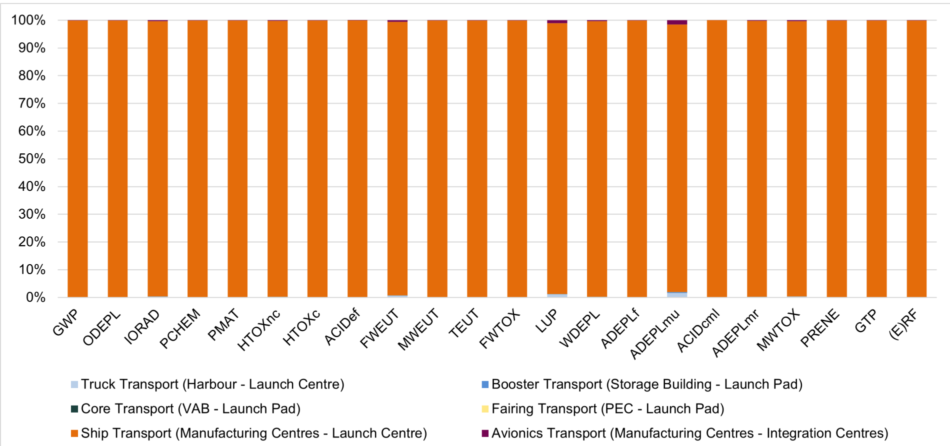

Figure 5.7: Ariane 6 Transport Characterisation

The transport segment with the highest impact is by far the main component transportation by ship: this is likely due to both the high mass of the components being transported and the length of the segment, as the other two phases whose impact appear in the graph involve either high masses or long distances. The analysis confirms how efforts to reduce the ship impact, such as employing sails [217], are a step in the right direction, as impacts from other transportation phases can be considered negligible.

5.1.4 Launch

The launcher use phase, alongside the propellant production, is the most impactful in terms of global warming potential, ionising radiation, human toxicity (both cancer and non-cancer), freshwater eutrophication, land use, resource use (energy carriers), primary energy consumption, global temperature change potential and (effective) radiative forcing. In Fig. 5.8, the subdivision of the impacts can be found.

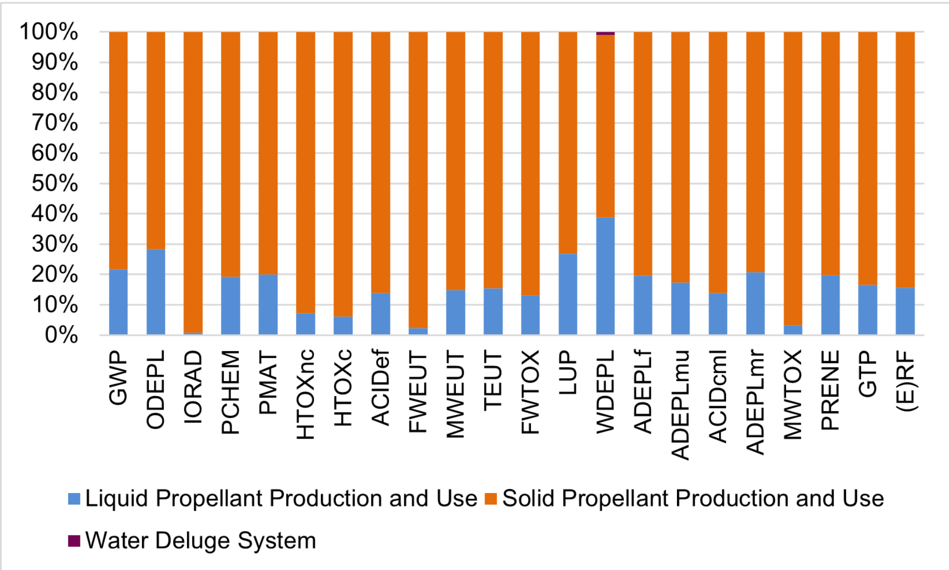

Figure 5.8: Ariane 6 Launch Characterisation

It is clear how the majority of the impacts is related to both the production and the use of the solid propellant, with the liquid propellants still making up a significant share, and the water impact being negligible instead. Therefore, an architecture with more boosters will pollute much more than an equivalent launch vehicle using only LO_x and LH_2; however, as explained in Sec. 5.2, the impact might be offset by the higher amount of deliverable payload. In Figures 5.9 and 5.10, the respective impact shares for the production and combustion of the propellants can be found.

It can clearly be seen how most of the impact related to the propellants comes from their production rather than their combustion. Although the analysis could become more accurate with future research on the dangers of emitted species, the efforts described in Subsec. 2.4.2 to make not only the propellant itself, but also its production line more environmentally sustainable are backed by the results of this analysis.

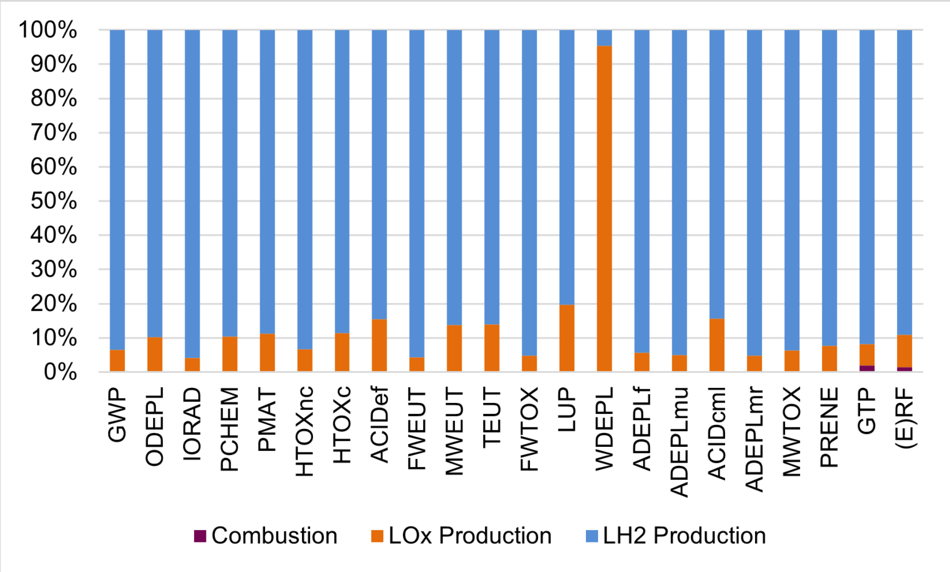

Figure 5.9: LO_x/LH_2 Impacts Characterisation

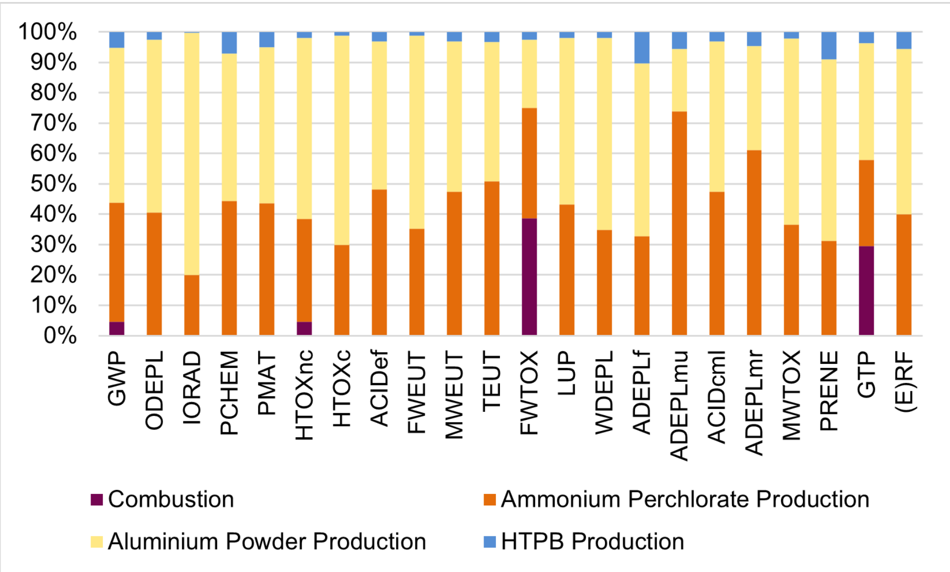

Figure 5.10: NH_4ClO_4/Al/HTPB Impacts Characterisation

5.1.5 Benchmarking

The results presented in Fig. 5.1 can be compared to two similar analyses that have been conducted and published.

The first one concerns a partially reusable mini-launcher developed by MaiaSpace [225]: the study includes the analysis ground activities, non-recurring items (such as testing and R&D (Research and Development)), and the production of the launcher and the propellant over the span of a year of operations [225]. The results have been presented in terms of climate change (GWP category), and resource depletion (ADEPLmu category) [225]. The main drivers in the climate change impact category are the ground activities (which include the transatlantic transportation of the components) and the production of propellants, with manufacturing making up only about 15% of the overall impact, and non-recurring activities having a negligible impact [225]. These results are largely comparable to those presented in Fig. 5.1, considering Ariane 6 ground activities do not include recovery and refurbishment, while the Maiaspace LCA analysis excluded the launch phase.

In terms of resource use, manufacturing is the biggest driver in both analyses, with propellant production being mostly responsible for the remaining impacts, and transport and non-recurring activities having a negligible impact.

The second study taken into consideration models a typical two stage RP-1 expendable launch vehicle [53]. The results of the study, which only focuses on the GWP impact category, attribute 30.4% of the impacts to manufacturing, and 67.6% to propellant production and combustion, while only 0.8% is attributed to logistics [53]. The slight difference when comparing these values to those presented in Fig. 5.1 is probably due to the fact that the analysis presented in [53] does not include testing and small component transportation, while the present document excludes impacts due to the assembly of the launcher, and includes the transportation of avionics from their manufacturing centre to the integration centre. Nevertheless, the main drivers and their respective impact percentages are comparable in both studies.

5.2 Ariane 6 Architectures Comparison

In Chapter 4, four different Ariane 6 architectures have been implemented: Ariane 62 and Ariane 64, which respectively have two and four solid boosters, each one of which can have a short or long payload fairing. In Fig. 5.11, the comparison between the impact per launch of the different architectures can be found. As expected, the Ariane 64 architecture has a higher absolute impact, mostly due to the additional booster structures and propellant; the difference between the short and long fairing architectures is minimal.

It is however mandatory to compare the impacts normalised per kilogram of payload: this is because a higher overall impact might be offset by a greater payload capability. Considering a payload capability in LEO of 10.3 tonnes for Ariane 62, and of 21.6 tonnes for Ariane 64 [97] (there is no change in capability based on the fairing dimensions), the results of the analysis can be found in Fig 5.12.

It can be seen how the higher environmental impact due to the additional components on Ariane 64 is actually offset by the higher payload. This underlines the importance of choosing the correct functional unit when carrying out LCA studies, as a wrong choice could lead to a misinterpretation of the

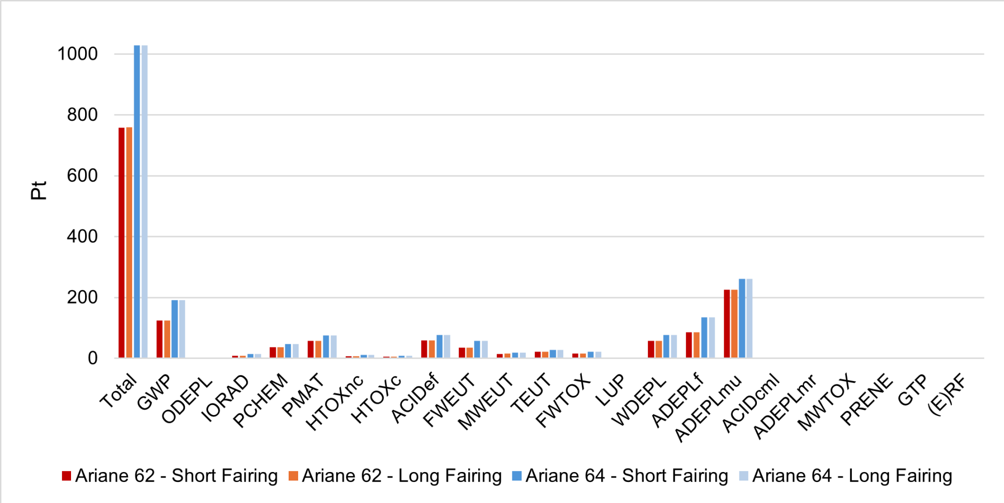

Figure 5.11: Ariane 6 Architectures Comparison

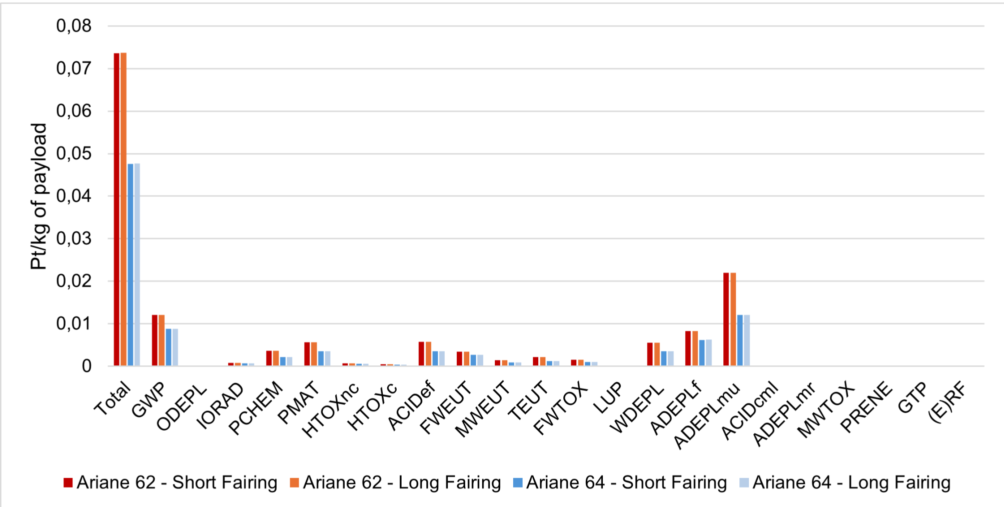

Figure 5.12: Ariane 6 Architectures Comparison Per kg of Payload

results. In this case, the seemingly worse choice is actually more sustainable when carrying the maximum payload possible; should one of the configurations perform a launch not at its full capability, the analysis should be done again, and its results might change.

It is also interesting to compare the numbers of launches it would take for all architectures to carry the same amount of payload in LEO: approximately 2.1 launches of Ariane 62 would be necessary to reach the same capacity as Ariane 64. The results of this comparison can be found in Fig. 5.13.

The difference between the Ariane 62 and Ariane 64 architectures is similar to the one obtained when normalising per kilogram of payload: it can therefore be concluded that, when carrying the maximum payload possible, it is better to have one bigger launch vehicle rather than multiple smaller vehicles. Smaller architectures are therefore useful for lighter payloads, while bigger ones should

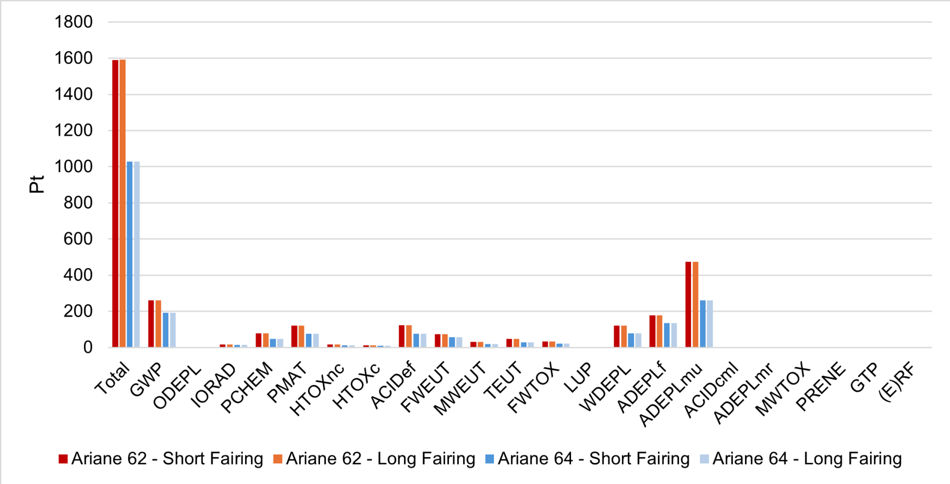

Figure 5.13: Ariane 6 Architectures Comparison (Same Payload)

be the primary choice when carrying heavier cargo, or whenever a satellite ride share is possible.

Chapter 6

Conclusions

This chapter will present the main conclusion and achievements of this work, alongside suggestions to improve future work on the matter.

6.1 Achievements

The major achievements of this work can be divided into three different sections: the state of the art review, the implementation of components and parameters usually missing in common LCA practices, and the Ariane 6 real case study.

Regarding the first section, a thorough review on the current limitations on LCA studies in the space sector was conducted. The main conclusions concern the lack of available data, which limits the accuracy of LCA studies, and details a framework in which companies are encouraged to share their data, with the common goal of achieving a more sustainable industry; this first section also deals with the risks related to the long life cycle of launchers, and it introduces prospective LCAs and risk factors as valuable tools to avoid incurring into material obsolescence while a product is still in its operational phase.

The biggest achievements of the implemented methodology firstly consist in detailing a typical launch vehicle testing routine, together with the correlated testing processes to implement in an LCA study.

Secondly, the methodology has dealt with implementing more than 70 different launch vehicle components, belonging to seven different subsystems. None of these components had previously been implemented in ESA's LCI database [17], and other studies and databases tend to include foreground data, rather than generalised processes. Although not every small scale component was implemented, as commercially available products were not always available, this first implementation can be considered a good result, as it allows users to model a complete, yet simplified, rocket, which can be helpful to both identify hotspots in very early design phases, and to compare different alternatives to ultimately select the most sustainable one. The methodology also dealt with identifying and implementing the production and use of some of the most common propellant combinations, focussing on their combustion products and their characterisation. In order to obtain more complete results, new metrics were added to the analysis method, whose purpose is to better characterise the particulate matter produced by rockets, whose impacts are yet to be completely understood. The addition of usually neglected products of combustion and impact metrics made the analysis as in depth as possible with the currently available knowledge.

Some of the key findings noted when applying everything that was previously introduced to the real case study of Ariane 6 were that testing impact can be considered negligible with respect to the emissions due to manufacturing, transportation and launch of a launcher. Avionics and big structural components are the main drivers in all impact categories relative to the production phase, while ship transport is the sole impact driver of the transport segment. Propellant production, especially concerning solid propellants, is far more impactful than their combustion, although studies on black carbon and alumina particles are yet to give conclusive results. Lastly, by comparing different Ariane 6 architectures, it was observed how a bigger absolute impact does not necessarily lead to a more polluting architecture: if the payload capability is high enough, the higher impacts can be offset by the higher quantity of deliverable cargo, thus making bigger architectures more sustainable. By identifying the main environmental hotspots, the LCA analysis led to the definition of the problem areas, which need further improvements to become more sustainable, and to the phases, such as testing, that can be initially neglected in order to obtain a simplified LCA.

6.2 Future Work

Future work on the matter could mostly be improved in two different ways: firstly, if more data is being openly shared and databases become more complete, the analysis can be carried out more accurately and in depth, giving more reliable results and reducing the usage of approximations. A better characterisation of combustion products could also lead to a reevaluation of the share of impacts related to the launch phase, to which could correspond a burden shift with respect to the analysis carried out in this work.

Secondly, the scope of the work could be expanded to include both the development and disposal phases, introducing new metrics to take the creation of space debris into account, and quantifying the office work and development testing environmental impacts, to give an overall more comprehensive analysis.

Appendix A

Ariane 6 Tank Volume

A.1 Lower Stage Tanks

The lower stage LO_x tank contains 125 tonnes of liquid oxygen at -183°C [97], which has a density of 1,141 kg/m^3. Its volume is therefore 109.55 m^3.
The lower stage LH_2 tank contains 25 tonnes of liquid hydrogen at -253°C [97], which has a density of 70.9 kg/m^3. Its volume is therefore 352.61 m^3.

A.2 Upper Stage Tanks

The upper stage LO_x tank contains 25 tonnes of liquid oxygen at -183°C [97], which has a density of 1,141 kg/m^3. Its volume is therefore 21.91 m^3.
The upper stage LH_2 tank contains 5 tonnes of liquid hydrogen at -253°C [97], which has a density of 70.9 kg/m^3. Its volume is therefore 70.52 m^3.

Bibliography

[1] ESA. «Ecodesign. Reducing Impacts». In: *Clean Space Days*. Oct. 2017 (cit. on pp. 1, 2, 22).

[2] Loïs Miraux. «Environmental limits to the space sector's growth». In: *Science of The Total Environment* 806, Part 4.150862 (2022). ISSN 0048-9697. DOI: `https://doi.org/10.1016/j.scitotenv.2021.150862` (cit. on pp. 1, 2, 5, 8, 9, 22–24, 26).

[3] Wilson, Andrew and Vasile, Massimiliano. «INTEGRATING LIFE CYCLE ASSESSMENT OF SPACE SYSTEMS INTO THE CONCURRENT DESIGN PROCESS». In: *68th International Astronautical Congress (IAC)* (Sept. 2017) (cit. on pp. 1, 2, 65, 66, 81).

[4] Jan-Steffen Fischer , Stefanos Fasoulas, Céline Brun-Buisson, Dr. Encarna del Olmo. «Comparison Study on the Environmental Impact of Different Launcher Architectures». In: Baku, Azerbaijan: 74th International Astronautical Congress (IAC), 2023 (cit. on pp. 1, 2, 15).

[5] ESA. *The ESA Green Agenda*. `https://www.esa.int/About_Us/Climate_and_Sustainability/The_ESA_Green_Agenda`. Accessed: 15.06.2025 (cit. on p. 1).

[6] ESA. *EcoDesign | Empowering Sustainable Space Activities*. `https://blogs.esa.int/cleanspace/2023/12/05/esa-lca-database-empowering-sustainable-space-activities/`. Accessed: 05.04.2025 (cit. on pp. 2, 19).

[7] T. Maury, P. Loubet, S. Morales Serrano, A. Gallice, G. Sonnemann. «Application of environmental life cycle assessment (LCA) within the space sector: A state of the art.» In: *Acta Astronautica* 170 (2020), pp. 122–135. DOI: `10.1016/j.actaastro.2020.01.035` (cit. on pp. 5, 7, 9, 15, 16, 19, 22, 29).

[8] A. R. Wilson. «Estimating the CO2 intensity of the space sector». In: *Nature Astronomy* 6(4) (2022), pp. 417–418. DOI: `https://doi.org/10.1038/s41550-022-01639-6` (cit. on pp. 5, 8, 19).

[9] ESA LCA Working Group. *Space system Life Cycle Assessment (LCA) guidelines*. 1st ed. Available upon request. ESSB-HB-U-005, Oct. 2016 (cit. on pp. 5, 8, 13, 16, 49–58, 82).

[10] ecoinvent Association. *ecoinvent Database, Version 3.9*. Zürich, Switzerland, 2022. URL: `https://www.ecoinvent.org` (cit. on pp. 5, 20, 22, 34–37, 39, 41, 42, 44, 46–51, 54, 56–60, 62, 68, 76–80).

[11] Fischer, Jan-Steffen. «Further development of LCA methodology for reusable and sustainable launchers.» In: Dresden, Germany: Ascension Conference, 2023 (cit. on pp. 5, 7, 13, 15, 16, 19, 22–24, 66).

[12] Scott Kordella, Ph.D. , Ruth Stilwell , John Giles , Christian Zur. «A Space Information Sharing Framework». In: *MITRE* (Nov. 2020) (cit. on p. 5).

[13] Keller, Kirsten M., Douglas Yeung, Dave Baiocchi, and William Welser IV. *Facilitating Information Sharing Across the International Space Community: Lessons from Behavioral Science.* Also available in print form. RAND Corporation. Santa Monica, CA, 2013. URL: `https://www.rand.org/pubs/technical_reports/TR1255.html.` (cit. on p. 6).

[14] European Commission. *European Data Governance Act*. `https://digital-strategy.ec.europa.eu/en/policies/data-governance-act`. Accessed: 21.02.2025. May 2022 (cit. on p. 6).

[15] European Commission. *Data Act*. `https://digital-strategy.ec.europa.eu/en/policies/data-act`. Accessed: 21.02.2025. Dec. 2023 (cit. on p. 6).

[16] European Space Law. *The European Space Law (EUSL)*. `https://www.european-space-law.com/`. Accessed: 21.02.2025. Mar. 2024 (cit. on p. 7).

[17] ESA. *Life Cycle Inventory (LCI) Database*. Available upon request. 2024 (cit. on pp. 7, 8, 11, 12, 16, 17, 20, 21, 23, 24, 27, 29, 31, 32, 34–37, 39–43, 46–49, 51, 54, 59–61, 67, 68, 72, 74, 81, 83, 85, 93).

[18] Sawik, B. «Space Mission Risk, Sustainability and Supply Chain: Review, Multi-Objective Optimization Model and Practical Approach.» In: *Sustainability* 15.11002 (14 2023). DOI: `https://doi.org/10.3390/su151411002` (cit. on p. 8).

[19] R. Sacchi and T. Terlouw and K. Siala and A. Dirnaichner and C. Bauer and B. Cox and C. Mutel and V. Daioglou and G. Luderer. «PRospective EnvironMental Impact asSEment (premise): A streamlined approach to producing databases for prospective life cycle assessment using integrated assessment models». In: *Renewable and Sustainable*

Energy Reviews 160.112311 (2022). ISSN: 1364-0321. DOI: https://doi.org/10.1016/j.rser.2022.112311 (cit. on pp. 8, 13).

[20] K.-S. Moon, H.W. Lee, and H. Kim. «Adaptive Data Selection-Based Machine Learning Algorithm for Prediction of Component Obsolescence.» In: *Sensors* 22.7982 (20 2022). DOI: https://doi.org/10.3390/s22207982 (cit. on pp. 9, 12).

[21] Craig L Josias. «Hedging Future Uncertainty: A Framework For Obsolescence Prediction, Proactive Mitigation And Management». Dissertation. University of Massachusetts Amherst, Feb. 2009. DOI: 10.7275/5645788 (cit. on pp. 9, 12).

[22] European Commission. *REACH Regulation*. https://environment.ec.europa.eu/topics/chemicals/reach-regulation_en. Accessed: 22.02.2025. Dec. 2006 (cit. on p. 9).

[23] European Space Components Information Exchange System (ESCIES). *REACH Background*. https://escies.org/webdocument/showArticle?id=1047&groupid=6. Accessed: 22.02.2025 (cit. on pp. 9–11).

[24] European Chemicals Agency (ECHA). *Search for chemicals*. https://echa.europa.eu/en/information-on-chemicals. Accessed: 22.05.2025 (cit. on pp. 9–11).

[25] P. Janik. «Evolution in Environmental Regulations - Update from ESA REACH Officer». In: *Clean Space Industry Days 2024*. Oct. 2024 (cit. on pp. 10, 12).

[26] Julian Austin. «Clean Space». In: *Clean Space Industry Days 2017*. May 2017 (cit. on p. 11).

[27] Johan Berg Pettersen and Mario Amin Salgado Delgado. «Harmonized space LCA data for efficient eco design. A common eco design basis for the European space industry». In: *Clean Space Industry Days 2018*. ESA ESTEC, Noordwijk, Oct. 2018 (cit. on p. 11).

[28] Cécile Bourrié. «REACH Influence on Surface Treatments». In: *MPTB stakeholder day*. May 2017 (cit. on p. 11).

[29] John Cockerill. *Hexavalent Chromium Ban of 2017 in Aluminium Surface Treatment*. https://surfacetreatment.johncockerill.com/news/as-of-2017-hexavalent-chromium-can-no-longer-be-used-in-aluminium-surface-treatment/. Accessed: 22.05.2025. Dec. 2016 (cit. on p. 11).

[30] ESA - Nebula Public Library. *REACH treatment of the pyrotechnics initiators powder*. https://nebula.esa.int/content/reach-treatment-pyrotechnics-initiators-powder. Accessed: 22.05.2025 (cit. on p. 11).

[31] ECAPS. *Green propellant LMP-103S*. https://www.ecaps.se/rocket-fuel-1. Accessed: 22.05.2025 (cit. on p. 11).

[32] AFRL. *Advanced Spacecraft Energetic Non-Toxic (ASCENT) Propellant*. https://afresearchlab.com/technology/aerospace/successstories/advanced-spacecraft-energetic-non-toxic-ascent-propellant/. Accessed: 22.05.2025 (cit. on p. 11).

[33] NASA. *New Green Propellants Complete Milestones*. https://www.nasa.gov/missions/tech-demonstration/new-green-propellants-complete-milestones/. Accessed: 22.05.2025. Sept. 2015 (cit. on p. 11).

[34] Alessandro Pellis, Fergal P. Byrne, James Sherwood, Marco Vastano, James W. Comerford and Thomas J. Farmer. «Safer bio-based solvents to replace toluene and tetrahydrofuran for the biocatalyzed synthesis of polyesters». In: *Green Chem.* 21 (2019). DOI: 10.1039/c8gc03567a (cit. on p. 11).

[35] ASD-EUROSPACE. «EUROPEAN SPACE SECTOR COMMENTS ON THE ANNEX XV RESTRICTION REPORT FOR PER- AND POLYFLUOROALKYL SUBSTANCES (PFAS)». In: ECHA Public Consultation of 22 March 2023 on the proposed restriction on the manufacture, placing on the market and use of PFASs, 2023 (cit. on p. 11).

[36] Harvard Business Review. *A Refresher on Regression Analysis*. https://hbr.org/2015/11/a-refresher-on-regression-analysis. Accessed: 23.05.2025. Nov. 2015 (cit. on p. 12).

[37] Timescale. *Time-Series Analysis: What Is It and How to Use It*. https://www.timescale.com/blog/time-series-analysis-what-is-it-how-to-use-it. Accessed: 23.05.2025 (cit. on p. 12).

[38] influxdata. *Exponential Smoothing: A Beginner's Guide to Getting Started*. https://www.influxdata.com/blog/exponential-smoothing-beginners-guide/. Accessed: 23.05.2025. May 2023 (cit. on p. 12).

[39] Andreas M. Hein. «How to Assess Heritage Systems in the Early Phases?» In: *6th International Systems and Concurrent Engineering for Space Applications (SECESA)*. Oct. 2014 (cit. on p. 13).

[40] Guillermo J. Dominguez Calabuig, Andrew Wilson, Sifeng Bi, Massimiliano Vasile, Martin Sippel, Martin Tajmar. «Environmental life cycle assessment of reusable launch vehicle fleets: Large climate impact driven by rocket exhaust emissions». In: *Acta Astronautica* 221.ISSN 0094-5765 (2024), pp. 1–11. DOI: `https://doi.org/10.1016/j.actaastro.2024.05.009` (cit. on pp. 14, 15, 29).

[41] Guillermo J. Dominguez Calabuig, Loïs Miraux, Andrew Ross Wilson, Alberto Sarritzu, Angelo Pasini. «Eco-design of future reusable launchers: insight into their life cycle and atmospheric impact». In: *9th EUROPEAN CONFERENCE FOR AERONAUTICS AND SPACE SCIENCES (EUCASS)*. 2022. DOI: `10.13009/EUCASS2022-7353` (cit. on pp. 14–16).

[42] Travis S. Taylor. *Introduction to Rocket Science and Engineering*. CRC Press, Taylor & Francis Group, 2009. ISBN: 978-1-4200-7529-8 (cit. on pp. 15, 19–22, 29, 43, 44, 50, 54, 56, 58, 61, 62).

[43] Heather Henry. «Life Cycle Assessment: Green Rocket Fuel». In: (2009) (cit. on pp. 15, 16).

[44] ESA. *Green hydrogen for Ariane 6 and more*. `https://www.esa.int/Enabling_Support/Space_Transportation/Green_hydrogen_for_Ariane_6_and_more`. Accessed: 23.07.2025. Dec. 2023 (cit. on p. 16).

[45] L3Harris. *Initial RS-25 Certification Campaign of 12 hot-fire tests complete*. `https://www.l3harris.com/newsroom/press-release/2023/06/initial-rs-25-certification-campaign-12-hot-fire-tests-complete`. Accessed: 02.04.2025. June 2023 (cit. on p. 20).

[46] SpaceNews. *Pratt & Whitney Rocketdyne Successfully Completes Hot-Fire Test Series on Second RS-68A Certification Engine*. `https://spacenews.com/pratt-whitney-rocketdyne-successfully-completes-hot-fire-test-series-on-second-rs-68a-certification-engine`. Accessed: 02.04.2025. Nov. 2010 (cit. on p. 20).

[47] NASA. *Appendix K - Space Shuttle Main Engine Testing History*. `https://www.nasa.gov/history/rogersrep/v2appk.htm`. Accessed: 02.04.2025. 1986 (cit. on p. 20).

[48] Rocket Lab. *Rocket Lab reaches 500 Rutherford engine test fires*. `https://www.rocketlabusa.com/updates/rocket-lab-reaches-500-rutherford-engine-test-fires`. Accessed: 06.04.2025. Jan. 2018 (cit. on p. 20).

[49] myRGV. *SpaceX consumes millions of gallons of Brownsville water amid scarcity*. `https://myrgv.com/local-news/2024/11/22/spacex-consumes-millions-of-gallons-of-brownsville-water-amid-scarcity/`. Accessed: 02.04.2025. Nov. 2024 (cit. on p. 20).

[50] NASA. *NASA Stennis Space Center Environmental Resources Document.* 1997 (cit. on p. 20).

[51] ChemEurope. *Supersonic wind tunnel.* `https://www.chemeurope.com/en/encyclopedia/Supersonic_wind_tunnel.html`. Accessed: 03.04.2025 (cit. on p. 21).

[52] Ryan, R. G. and Marais, E. A. and Balhatchet, C. J. and Eastham, S. D. «Impact of rocket launch and space debris air pollutant emissions on stratospheric ozone and global climate.» In: *Earth's Future* 10.e2021EF002612 (2022). DOI: `https://doi.org/10.1029/2021EF002612` (cit. on pp. 22–24, 27, 28).

[53] Bellier, Thomas and Urbano, Annafederica and Morlier, Joseph and Bil, Cees and Pudsey, Adrian. «Impact of Life Cycle Assessment Considerations on Launch Vehicle Design. » In: (2023) (cit. on pp. 22, 89).

[54] Sylvie Durrieu, Ross F. Nelson. «Earth observation from space – The issue of environmental sustainability.» In: *Space Policy* 4 (2013), pp. 238–250. ISSN: 0265-9646. DOI: `https://doi.org/10.1016/j.spacepol.2013.07.003` (cit. on p. 23).

[55] ICCT. *A policy-relevant summary of black carbon climate science and appropriate emission control strategies.* Tech. rep. June 2009 (cit. on p. 24).

[56] Future Learn. *Causes of Climate Change: The Radiative Forcing Concept.* Accessed: 07.05.2025. URL: `https://www.futurelearn.com/info/courses/causes-of-climate-change/0/steps/13583%5C#:~:text=A%20radiative%20forcing%20is%20an%20energy%20imbalance,of%20greenhouse%20gases%2C%20aerosols%2C%20and%20their%20precursors).%5C&text=The%20inclusion%20of%20rapid%20tropospheric%20adjustments%20to,is%20often%20called%20the%20effective%20radiative%20forcing.` (cit. on p. 24).

[57] Carl Russell, Wayne Johnson. *Application of Climate Impact Metrics to Civil Tiltrotor Design.* Moffett Field, CA, 94035 (cit. on p. 24).

[58] Jan-Steffen Fischer, Stefanos Fasoulas. «Review of the environmental impact of space transportation systems towards a full life cycle assessment». In: *73rd International Astronautical Congress.* Paris, France: IAC, 2022 (cit. on pp. 24, 26–28).

[59] Szopa, S., V. Naik, B. Adhikary, P. Artaxo, T. Berntsen, W.D. Collins, S. Fuzzi, L. Gallardo, A. Kiendler-Scharr, Z. Klimont, H. Liao, N. Unger, and P. Zanis. «Climate Change 2021: The Physical Science Basis. Contribution of Working Group I to the Sixth Assessment Report of the Intergovernmental Panel on Climate Change». In: ed. by Masson-Delmotte,

V., P. Zhai, A. Pirani, S.L. Connors, C. Péan, S. Berger, N. Caud, Y. Chen, L. Goldfarb, M.I. Gomis, M. Huang, K. Leitzell, E. Lonnoy, J.B.R. Matthews, T.K. Maycock, T. Waterfield, O. Yelekçi, R. Yu, and B. Zhou. Cambridge University Press, Cambridge, United Kingdom and New York, NY, USA, 2021. Chap. Short-Lived Climate Forcers, pp. 817–922. DOI: 10.1017/9781009157896.008 (cit. on pp. 24, 25).

[60] Nordling, K., Korhonen, H., Räisänen, J., Partanen, A.-I., Samset, B. H., and Merikanto, J. «Understanding the surface temperature response and its uncertainty to CO2, CH4, black carbon, and sulfate». In: *Atmos. Chem. Phys.* 21 (2021), pp. 14941–14958. DOI: https://doi.org/10.5194/acp-21-14941-2021 (cit. on p. 24).

[61] Collins, William and Fry, M.M. and Yu, Hongbin and Fuglestvedt, J.S. and Shindell, D.T. and West, Jason. «Global and regional temperature-change potentials for near-term climate forcers». In: *ATMOSPHERIC CHEMISTRY AND PHYSICS* 13 (2013), pp. 2471–2485. DOI: 10.5194/acp-13-2471-2013 (cit. on p. 25).

[62] Forster, P., T. Storelvmo, K. Armour, W. Collins, J.-L. Dufresne, D. Frame, D.J. Lunt, T. Mauritsen, M.D. Palmer, M. Watanabe, M. Wild, and H. Zhang. «Climate Change 2021: The Physical Science Basis. Contribution of Working Group I to the Sixth Assessment Report of the Intergovernmental Panel on Climate Change». In: ed. by Masson-Delmotte, V., P. Zhai, A. Pirani, S.L. Connors, C. Péan, S. Berger, N. Caud, Y. Chen, L. Goldfarb, M.I. Gomis, M. Huang, K. Leitzell, E. Lonnoy, J.B.R. Matthews, T.K. Maycock, T. Waterfield, O. Yelekçi, R. Yu, and B. Zhou. Cambridge University Press, Cambridge, United Kingdom and New York, NY, USA, 2021. Chap. The Earth's Energy Budget, Climate Feedbacks, and Climate Sensitivity, pp. 923–1054. DOI: 10.1017/9781009157896.009 (cit. on p. 25).

[63] Goita, E.G., Beagle, E.A., Nasta, A.N. et al. «Effect of hydrogen leakage on the life cycle climate impacts of hydrogen supply chains». In: *Communications Earth & Environment* 160 (2025). DOI: https://doi.org/10.1038/s43247-025-02141-3 (cit. on p. 25).

[64] Warwick, N. J. and Archibald, A. T. and Griffiths, P. T. and Keeble, J. and O'Connor, F. M. and Pyle, J. A., and Shine, K. P. «Atmospheric composition and climate impacts of a future hydrogen economy». In: *Atmospheric Chemistry and Physics* 23 (20 2023), pp. 13451–13467. DOI: https://doi.org/10.5194/acp-23-13451-2023 (cit. on p. 25).

[65] Ocko, I. B., and Hamburg, S. P. «Climate consequences of hydrogen emissions». In: *Atmospheric Chemistry and Physics* 22 (14 2022), pp. 9349–

9368. DOI: `https://doi.org/10.5194/acp-22-9349-2022` (cit. on p. 25).

[66] G. Myhre et al. «Climate Change 2013: The Physical Science Basis. Contribution of Working Group I to the Fifth Assessment Report of the Intergovernmental Panel on Climate Change». In: ed. by T.F. Stocker et al. Cambridge University Press, Cambridge, United Kingdom and New York, NY, USA, 2013. Chap. Anthropogenic and Natural Radiative Forcing, pp. 659–740 (cit. on p. 25).

[67] Sanford Gordon and Bonnie J. McBride. *Computer Program for Calculation of Complex Chemical Equilibrium Compositions and Applications, Part I: Analysis*. Tech. rep. NASA RP-1311. NASA Reference Publication 1311. Washington, DC: NASA, 1994 (cit. on pp. 27–29).

[68] Anita Pinalia, Bayu Prianto, Henny Setyaningsih, Prawita Dhewi, Ratnawati Ratnawati. «Design of Propellant Composite Thermodynamic Properties Using Rocket Propulsion Analysis (RPA) Software». In: *Reaktor* 22.1 (2022), pp. 1–6 (cit. on p. 29).

[69] Sand, Maria and Skeie, Ragnhild and Sandstad, Marit and Krishnan, Srinath and Myhre, Gunnar and Bryant, Hannah and Derwent, Richard and Hauglustaine, Didier and Paulot, Fabien and Prather, Michael and Stevenson, David. «A multi-model assessment of the Global Warming Potential of hydrogen». In: *Communications Earth & Environment* 4 (2023). DOI: `10.1038/s43247-023-00857-8` (cit. on p. 29).

[70] Wilson, Andrew and Vasile, Massimiliano and Maddock, Christie and Baker, Keith. «The Strathclyde Space Systems Database: A New Life Cycle Sustainability Assessment Tool for the Design of Next Generation Green Space Systems. » In: *8th International Systems & Concurrent Engineering for Space Applications*. Glasgow, UK, 2018 (cit. on p. 29).

[71] Paul K. McConnaughey, Mark G. Femminineo, Syri J. Koelfgen, Roger A. Lepsch, Richard M. Ryan, and Steven A. Taylor. *DRAFT Launch Propulsion Systems Roadmap - Technology Area 01*. NASA, 2010 (cit. on p. 30).

[72] David L. Akin. *Mass Estimating Relations*. Tech. rep. University of Maryland, 2016 (cit. on pp. 31, 58).

[73] Rocket Lab. *PAYLOAD USER's GUIDE*. Version 7.0. Nov. 2022 (cit. on pp. 31, 32).

[74] Firefly Aerospace. *ALPHA PAYLOAD USER'S GUIDE (V5.1)*. Version 5.1. Feb. 2025 (cit. on pp. 31, 32).

[75] Ryan Woolley. *Flight Club - Interactive rocket launch simulator*. `https://flightclub.io`. Accessed: 06.04.2025. 2024 (cit. on pp. 31, 32).

[76] Firefly Aerospace. *ALPHA PAYLOAD USER'S GUIDE (V2.0)*. Version 2.0. Aug. 2019 (cit. on pp. 31, 32).

[77] Firefly Aerospace. *Tweet on stage tank structure mass*. Accessed: 06.04.2025. 2019. URL: `https://x.com/Firefly_Space/status/1181211551434280963` (cit. on p. 31).

[78] United Launch Alliance. *Atlas V Launch Services User's Guide (V11)*. Mar. 2010 (cit. on pp. 33–37, 43).

[79] ArianeSpace. *Soyuz User's Guide (V2)*. Mar. 2012 (cit. on pp. 33–37, 43).

[80] ISRO. *LVM3-X/CARE MISSION*. 2023 (cit. on pp. 33–35, 37, 43).

[81] Svs, Narayana and Venukuttan, C. and Remesh, G. and Shivaprasad, T.S. and Arumugam, M. «Materials and Manufacturing of Liquid Propulsion Systems for LVM3-M4/Chandrayaan-3 Mission». In: (2024) (cit. on pp. 33–36, 46–49).

[82] Astronautix. *RD-180*. `http://www.astronautix.com/r/rd-180.html`. Accessed: 10.04.2025 (cit. on p. 33).

[83] Astronautix. *RD-108-8D727*. `http://www.astronautix.com/r/rd-108-8d727.html`. Accessed: 22.04.2025 (cit. on pp. 33, 47).

[84] J.E. Fesmire, B.E. Coffman, B.J. Meneghelli, K.W. Heckle. «Spray-on foam insulations for launch vehicle cryogenic tanks». In: *Cryogenics* 52 (4-6 2012), pp. 251–261. ISSN: 0011-2275. DOI: `https://doi.org/10.1016/j.cryogenics.2012.01.018` (cit. on pp. 33, 35, 36, 38, 40, 41).

[85] Astronautix. *Viking 4B*. `http://www.astronautix.com/v/viking4b.html`. Accessed: 22.04.2025 (cit. on p. 33).

[86] SmarterEveryDay. *HOW ROCKETS ARE MADE (Rocket Factory Tour - United Launch Alliance) - Smarter Every Day 231*. `https://www.youtube.com/watch?v=o0fG_lnVhHw&t=384s`. Accessed: 04.03.2025. Feb. 2020 (cit. on pp. 33, 34, 36–39, 41, 42).

[87] Astronautix. *RD-0124*. `http://www.astronautix.com/r/rd-0124.html`. Accessed: 27.04.2025 (cit. on pp. 35, 58, 59).

[88] IasGyan. *CE 20 ENGINES*. Accessed: 27.04.2025. Dec. 2024. URL: `https://www.iasgyan.in/daily-current-affairs/ce-20-engines` (cit. on p. 35).

[89] ISRO. *LVM3-M4/CHANDRAYAAN-3 MOON MISSION*. 2023 (cit. on pp. 35, 36).

[90] Thomas J. Rudman, Kurt L. Austad (Lockheed Martin Space Systems Company). «The Centaur Upper Stage Vehicle». In: *4th International Conference on Launcher Technology "Space Launcher Liquid Propulsion"*. Liege (Belgium): CNES, 3-6 December 2002 (cit. on pp. 37, 42).

[91] United States Space Force. *Centaur upper stage installation recognizes trailblazing space program*. Accessed: 28.04.2025. Nov. 2024. URL: `https://www.spaceforce.mil/News/Article-Display/Article/3955033/centaur-upper-stage-installation-recognizes-trailblazing-space-program/` (cit. on pp. 37, 42).

[92] Beyond Gravity. *Launcher Structures.* `https://www.beyondgravity.com/en/launchers/launcher-structures#Interstage+adapters`. Accessed: 06.03.2025. 2025 (cit. on pp. 37, 39).

[93] Impact Materials. *Carbon Fiber Density: Facts and Insights.* `https://ictfibers.com/carbon-fiber-density-facts-and-insights/`. Accessed: 27.02.2025. Jan. 2024 (cit. on p. 37).

[94] SpaceX. *Falcon 9*. `https://www.spacex.com/vehicles/falcon-9/`. Accessed: 05.03.2025. 2025 (cit. on p. 37).

[95] Wevolver. *Falcon 9 v1.2 or Full Thrust - Block 5*. `https://www.wevolver.com/specs/falcon-9-v12-or-full-thrust-block-5`. Accessed: 04.06.2025 (cit. on pp. 38, 40).

[96] ESA. *Ariane 6: what's it made of?* `https://www.esa.int/Enabling_Support/Space_Transportation/Ariane/Ariane_6_what_s_it_made_of`. Accessed: 10.03.2025. June 2024 (cit. on pp. 38–41, 43, 44, 68, 78).

[97] Ariane Group. *Ariane 6*. `https://ariane.group/en/space-transportation/ariane-6/`. Accessed: 05.03.2025. Mar. 2025 (cit. on pp. 38–41, 68, 70, 74, 89, 95).

[98] SimpleRockets. *ULA Vulcan Centaur Heavy*. `https://www.simplerockets.com/c/500wK2/ULA-Vulcan-Centaur-Heavy`. Accessed: 12.03.2025. 2020 (cit. on pp. 38, 41).

[99] *GEM 63 AND GEM 63XL*. Northrop Grumman. 2020 (cit. on pp. 38, 44, 59, 60).

[100] ULA. *Vulcan Centaur*. Accessed: 06.03.2025. 2022 (cit. on pp. 38–44).

[101] SpaceX. *Falcon User's Guide*. Sept. 2021 (cit. on pp. 38–43, 61).

[102] *Vulcan Launch System's User Guide*. United Launch Alliance (ULA). P.O. Box 3788 Centennial, CO 80112, Oct. 2023 (cit. on pp. 38, 40, 41).

[103] Astronautix. *RL-10C-X*. `http://www.astronautix.com/r/rl-10c-x.html`. Accessed: 08.05.2025 (cit. on pp. 38, 41, 47, 58, 59).

[104] J. Merino, A. Patzelt, A. Steinacher, M. Windisch, G. Heinrich, R. Forster, C. Bauer. «ARIANE 6 – TANKS & STRUCTURES FOR THE NEW EUROPEAN LAUNCHER». In: *MT Aerospace AG, Germany* (2017) (cit. on pp. 38, 39, 41).

[105] M. Wittal, B. McCann, J. Benavides, M. Nazari. «AMBIGUITY REMEDIATION IN SPACE LAUNCH VEHICLES WITH PARAMETER UNCERTAINTIES: A COMPARISON BETWEEN SPECIAL EUCLIDEAN GROUP AND DUAL QUATERNIONS.» In: *AAS 22.821* (2022) (cit. on pp. 38–41).

[106] NASA SPACE FLIGHT. *ULA Vulcan Centaur Heavy*. `https://www.nasaspaceflight.com/2024/01/vulcan-flight-one-preps/`. Accessed: 12.03.2025. Jan. 2024 (cit. on p. 38).

[107] Astronautix. *Merlin 1D*. `http://www.astronautix.com/m/merlin1d.html`. Accessed: 18.03.2025 (cit. on pp. 38, 47, 58).

[108] ESA. *Vulcain 2.1*. `https://www.esa.int/esearch?q=vulcain+2.1`. Accessed: 18.03.2025. Oct. 2017 (cit. on pp. 38, 48, 58, 59).

[109] Everyday Astronaut. *Is SpaceX's Raptor engine the king of rocket engines?* `https://everydayastronaut.com/raptor-engine/`. Accessed: 12.03.2025. May 2019 (cit. on p. 38).

[110] Blue Origin. *BE-4 Engine*. `https://www.blueorigin.com/engines/be-4`. Accessed: 12.03.2025. 2025 (cit. on p. 38).

[111] Ad Astra. *Everything to know about SpaceX's mid-air booster "catch"*. `https://www.adastraspace.com/p/spacex-super-heavy-catch/`. Accessed: 14.03.2025. Oct. 2024 (cit. on p. 38).

[112] Seedhouse, E. «Falcon 9 and Falcon Heavy: Life after the Space Shuttle.» In: Praxis, New York, NY., 2013, pp. 65–84. DOI: `https://doi.org/10.1007/978-1-4614-5514-1_5` (cit. on p. 39).

[113] Teslarati. *Falcon 9 S1 orbital welding (Sam Teller)*. `https://www.teslarati.com/spacex-ceo-elon-musk-says-that-bfr-could-cost-less-to-build-than-falcon-9/falcon-9-s1-orbital-welding-sam-teller-2/`. Accessed: 20.03.2025. Feb. 2019 (cit. on p. 39).

[114] Jack Silvester. *A LOOK INSIDE Space X Falcon 9's Fuel Tank!* `https://www.youtube.com/watch?v=B2296IuK28g`. Accessed: 20.03.2025. Nov. 2019 (cit. on p. 39).

[115] Astronautix. *Merlin 1DVac*. `http://www.astronautix.com/m/merlin1dvac.html`. Accessed: 22.03.2025 (cit. on pp. 40, 47, 58).

[116] Astronautix. *Vinci*. `http://www.astronautix.com/v/vinci.html`. Accessed: 22.03.2025 (cit. on pp. 40, 47, 58, 59).

[117] Beyond Gravity. *Shaping the next generation of European launch vehicles: Contract won for Ariane 6*. `https://www.beyondgravity.com/en/news/shaping-next-generation-european-launch-vehicles-contract-won-ariane-6`. Accessed: 10.06.2025. June 2023 (cit. on pp. 43, 67, 78).

[118] Astronautix. *Zefiro 9*. `http://www.astronautix.com/z/zefiro9.html`. Accessed: 18.05.2025 (cit. on p. 43).

[119] THIOKOL/WASATCH DIVISION. *FINAL REPORT SPACE SHUTTLE SRM INTERIM CONTRACT*. Tech. rep. Thiokol Corporation, 1974 (cit. on pp. 43, 44).

[120] ULA. *Atlas V Cutaway*. 2022 (cit. on pp. 44, 59).

[121] *Epsilon Launch Vehicle User's Guide (Revision A)*. Japan Aerospace Exploration Agency (JAXA). July 2018 (cit. on p. 44).

[122] Avio. *P120 C MOTOR MAIN CHARACTERISTICS*. `https://www.avio.com/p120c`. Accessed: 18.05.2025 (cit. on pp. 44, 78, 80).

[123] *IGNITION AND STAGING CONTROLLER Technical Datasheet*. L3Harris Technologies. 1025 W. NASA Boulevard Melbourne, FL 32919, 2021 (cit. on pp. 43, 49).

[124] Gary N. Harris. *FRANGIBLE JOINT SEPARATION SYSTEM*. U.S. Patent USOO5390606A. Nov. 1991 (cit. on pp. 43, 44).

[125] Prof. Dario Pastrone. *Slides from "Endoreattori" course*. 2023/2024 (cit. on pp. 44, 49, 58–60).

[126] Tizon, Juan & Roman, Alberto. «A Mass Model for Liquid Propellant Rocket Engines». In: *53rd AIAA/SAE/ASEE Joint Propulsion Conference*. 2017. DOI: `10.2514/6.2017-5010` (cit. on pp. 44, 45).

[127] S, Prarthana & Biswal M, Malaya Kumar. «A Short Review on ISRO Rocket Engines». In: *Acceleron Aerospace Journal* 1 (2023), pp. 95–100. DOI: `10.61359/11.2106-2320` (cit. on pp. 46, 58, 59).

[128] Zhang, Nan. «THE DEVELOPMENT OF LOX/LH2 ENGINE IN CHINA». In: *64rd International Astronautical Congress*. Beijing, China, 2013 (cit. on pp. 46, 58).

[129] Astronautix. *LE-7A*. `http://www.astronautix.com/l/le-7a.html`. Accessed: 14.05.2025 (cit. on pp. 47, 58).

[130] huijiwiki. *YF-115*. `https://sat.huijiwiki.com/wiki/YF-115`. Accessed: 14.05.2025. Apr. 2025 (cit. on pp. 47, 58).

[131] APTI PLUS. *CE 20 ENGINES*. `https://www.iasgyan.in/daily-current-affairs/ce-20-engines`. Accessed: 14.05.2025. Dec. 2024 (cit. on pp. 47, 58).

[132] Astronautix. *RD-108-11D512*. http://www.astronautix.com/r/rd-108-11d512.html. Accessed: 14.05.2025 (cit. on pp. 47, 58, 59).

[133] Wikiwand. *YF-40*. https://www.wikiwand.com/en/articles/YF-40. Accessed: 14.05.2025 (cit. on pp. 47, 58, 59).

[134] Astronautix. *YF-77*. http://www.astronautix.com/y/yf-77.html. Accessed: 14.05.2025 (cit. on pp. 47, 58).

[135] Astronautix. *LE-5B engine*. http://www.astronautix.com/l/le-5bengine.html. Accessed: 14.05.2025 (cit. on pp. 47, 58).

[136] Astronautix. *RD-191*. http://www.astronautix.com/r/rd-191.html. Accessed: 15.05.2025 (cit. on pp. 48, 58, 59).

[137] Astronautix. *SSME*. http://www.astronautix.com/s/ssme.html. Accessed: 15.05.2025 (cit. on pp. 48, 58, 59).

[138] John A. Halchak, James L. Cannon, and Corey Brown. «Materials for Liquid Propulsion Systems». In: *Liquid Propellant Rocket Engine Performance Testing*. NASA Marshall Space Flight Center, Huntsville, Alabama, 2013 (cit. on pp. 46, 58).

[139] A. Salama et al. «Validation of relations for the determination of pyrotechnic igniter mass». In: *IOP Conf. Ser.: Mater. Sci. Eng.* 973.012004 (2020) (cit. on p. 50).

[140] *Lithium-Ion FTS Battery 39401 Technical Datasheet*. Space Vector Corporation. 20520 Nordhoff Street, Chatsworth CA, 91311, 2023 (cit. on p. 50).

[141] *Lithium-Ion FTS & TM Battery 39401-1 Technical Datasheet*. Space Vector Corporation. 20520 Nordhoff Street, Chatsworth, CA 91311, 2023 (cit. on p. 50).

[142] *Lithium-Ion Battery 39611 Smart Series Technical Datasheet*. Space Vector Corporation. 20520 Nordhoff Street, Chatsworth, CA 91311, 2022 (cit. on p. 50).

[143] *Lithium-Ion FTS Battery 39611-1 Smart Series Technical Datasheet*. Space Vector Corporation. 20520 Nordhoff Street, Chatsworth, CA 91311, 2022 (cit. on p. 50).

[144] *High Power Li-Ion Polymer Battery 39381-4 Technical Datasheet*. Space Vector Corporation. 20520 Nordhoff Street, Chatsworth, CA 91311, 2022 (cit. on p. 50).

[145] *Lithium-Ion Battery 39501 Series Technical Datasheet*. Space Vector Corporation. 20520 Nordhoff Street, Chatsworth, CA 91311, 2022 (cit. on p. 50).

[146] *High Capacity Li-Ion Battery 39541 Series Technical Datasheet*. Space Vector Corporation. 20520 Nordhoff Street, Chatsworth, CA 91311, 2022 (cit. on p. 50).

[147] *Modular Li-Ion Battery 39521 Series Technical Datasheet*. Space Vector Corporation. 20520 Nordhoff Street, Chatsworth, CA 91311, 2022 (cit. on p. 50).

[148] *LAUNCHER-BATT. A battery product line made for Launchers*. AIRBUS. 2025 (cit. on pp. 50, 69, 76).

[149] *PDU300/POWER DISTRIBUTION Technical Datasheet*. MAROTTA. 78 Boonton Ave. P.O. Box 427 Montville, NJ 07045, 2019 (cit. on p. 51).

[150] *SVFL2800D SERIES Technical Datasheet*. VPT Power. 2022 (cit. on p. 52).

[151] *SVTR2800D SERIES Technical Datasheet*. VPT Power. 2022 (cit. on p. 52).

[152] *SVSA2800D SERIES Technical Datasheet*. VPT Power. 2022 (cit. on p. 52).

[153] *SVLFL2800D SERIES Technical Datasheet*. VPT Power. 2021 (cit. on p. 52).

[154] *SVLTR2800D SERIES Technical Datasheet*. VPT Power. 2024 (cit. on p. 52).

[155] *SVLHF2800S SERIES Technical Datasheet*. VPT Power. 2022 (cit. on p. 52).

[156] *SVLSA5000D SERIES Technical Datasheet*. VPT Power. 2024 (cit. on p. 52).

[157] *SVRFL2800D SERIES Technical Datasheet*. VPT Power. 2020 (cit. on p. 52).

[158] *VSCF20-28 Technical Datasheet*. 2nd ed. VPT. Nov. 2024 (cit. on p. 53).

[159] *DVME28 Series Technical Datasheet*. 10th ed. VPT. Oct. 2023 (cit. on p. 53).

[160] *VSCF10-28 Technical Datasheet*. 7th ed. VPT. Nov. 2024 (cit. on p. 53).

[161] *DVMD28 Series Technical Datasheet*. 9th ed. VPT. Oct. 2023 (cit. on p. 53).

[162] *SVRMC28 Series Technical Datasheet*. 9th ed. VPT. Nov. 2020 (cit. on p. 53).

[163] *DVMD50 Series Technical Datasheet*. 5th ed. VPT. Dec. 2023 (cit. on p. 53).

[164] *VSCF1-28 Technical Datasheet*. 4th ed. VPT. Nov. 2024 (cit. on p. 53).

[165] *DVMH50 Series Technical Datasheet*. 2nd ed. VPT. Apr. 2024 (cit. on p. 53).

[166] *SGTPL-28 Technical Datasheet*. Vishay. Apr. 2025 (cit. on p. 53).

[167] *SGTPL-2516 Technical Datasheet*. Vishay. Jan. 2023 (cit. on p. 53).

[168] *Solid State, Thin Film, SMD 1206 Fuse for Space Application, ESCC QPL Listed*. Schurter. Mar. 2025 (cit. on p. 54).

[169] *Solid State, Thin Film, SMD 3220 Fuse for High Current Space Application, ESCC QPL Listed*. Schurter. Mar. 2025 (cit. on p. 54).

[170] *Surface Mount Fuse, 5.3 x 16 mm, Super-Time-Lag TT, 125 VAC / 125 VDC, Breaking Capacity ≤ 1000 A*. Schurter. Mar. 2025 (cit. on p. 54).

[171] *Solid State, Thin Film, SMD 3220 Fuse for High Current Application*. Schurter. Mar. 2025 (cit. on p. 54).

[172] *Frontier-X: A Software Defined, High Data-Rate TT&C Radio Technical Datasheet*. Document Number S300177, Rev 1.1. Rocket Lab. May 2025 (cit. on p. 55).

[173] *Frontier-S: A Software Defined TT&C Radio Technical Datasheet*. Document Number S300179, Rev 1.1. Rocket Lab. May 2025 (cit. on p. 55).

[174] *T-740E S-BAND LAUNCH VEHICLE TRANSMITTER Technical Datasheet*. L3HARRIS. 2025 (cit. on p. 55).

[175] Anywaves. *Launchers Antennas: the backbone of Space Missions*. `https://anywaves.com/resources/blog/launchers-antennas-backbone-space-missions/`. Accessed: 28.02.2025. Nov. 2024 (cit. on p. 55).

[176] *AS-49090 SINGLE-LAYER MICROSTRIP PATCH ANTENNA Technical Datasheet*. L3Harris Technologies. 1025 W. NASA Boulevard Melbourne, FL 32919, 2021 (cit. on p. 56).

[177] *AS-49034 CAVITY-BACKED HELIX ANTENNA Technical Datasheet*. L3Harris Technologies. 1025 W. NASA Boulevard Melbourne, FL 32919, 2021 (cit. on p. 56).

[178] *AS-49063 MONOPOLE BLADE ANTENNA Technical Datasheet*. L3Harris Technologies. 1025 W. NASA Boulevard Melbourne, FL 32919, 2021 (cit. on p. 56).

[179] *PHOENIX FLIGHT COMPUTER Technical Datasheet*. L3Harris Technologies. 1025 W. NASA Boulevard Melbourne, FL 32919, 2020 (cit. on pp. 56, 61).

[180] *RH3440 Solid-State Data Recorder 3U SRIO VPX Radiation Tolerant SSDR Technical Datasheet*. Mercury. 50 Minuteman Road Andover, MA 01810 USA, 2021 (cit. on p. 57).

[181] *RH304T Solid-State Data Recorder 3U SRIO VPX, 4.5 TB Radiation Tolerant SSDR Technical Datasheet*. Mercury. 50 Minuteman Road Andover, MA 01810 USA, 2021 (cit. on p. 57).

[182] *RH3480 Solid-State Data Recorder 3U SRIO VPX Radiation Tolerant SSDR Technical Datasheet*. Mercury. 50 Minuteman Road Andover, MA 01810 USA, 2021 (cit. on p. 57).

[183] Brian Simmons. *TAPE WRAPPED, FABRIC REINFORCED, FLEX BEARING*. U.S. Patent US005399309A. Dec. 1985 (cit. on p. 59).

[184] Mohamed G. AbuElkhier, Sameh Shaaban, Mahmoud Y.M. Ahmed. «Numerical investigation of an add-on thrust vector control kit». In: *Advances in Aircraft and Spacecraft Science* 9.1 (2022), pp. 39–57. DOI: https://doi.org/10.12989/aas.2022.9.1.039 (cit. on p. 59).

[185] Ashish Tewari, T. Srinivasulu, A. Ramesh, Anil Kumar, S. SaiKumar. «Development of Manufacturing Technology for C-SiC Jet Vanes». In: *Procedia Materials Science* 5.2211-8128 (2014), pp. 1567–1573. DOI: https://doi.org/10.1016/j.mspro.2014.07.344 (cit. on p. 59).

[186] Richard G. Eatough. *Improved Jet Tab Thrust Vector Control for the BGM-34C Booster*. Tech. rep. TRW275906001RU00. TRW SYSTEMS GROUP, 1977 (cit. on p. 59).

[187] Thomas L. Nickens, John L. Sutton, Lione, H. Erickson, Philip C. Sottosanti. «THRUST VECTOR CONTROL STUDY FOR LARGE (260 IN.) ROCKET MOTOR APPLICATIONS ». In: Dresden, Germany: Thiokol Chemical Corporation, Apr. 1970 (cit. on p. 59).

[188] Shinichi Watanabe and Takashi Miyata. *Jetavator for Rocket Engine*. U.S. Patent US5735463A. Apr. 1998 (cit. on p. 60).

[189] Astronautix. *R-1E engine*. http://www.astronautix.com/r/r-1eengine.html. Accessed: 21.05.2025 (cit. on p. 61).

[190] Astronautix. *LR101-11*. http://www.astronautix.com/l/lr101-11.html. Accessed: 21.05.2025 (cit. on p. 61).

[191] Elon Musk. *Tweet on hypersonic grid fins*. Accessed: 16.05.2025. 2017. URL: https://x.com/elonmusk/status/878821062326198272 (cit. on p. 61).

[192] The Space Techie. *Grid Fins: The Wings for Re-entry!* Accessed: 16.05.2025. July 2021. URL: https://www.thespacetechie.com/grid-fins-the-wings-for-re-entry/ (cit. on p. 61).

[193] RANGE SAFETY GROUP. *FLIGHT TERMINATION SYSTEMS COMMONALITY STANDARD - DOCUMENT 319-19*. June 2019 (cit. on p. 62).

[194] *DESTRUCT CHARGE*. Pacsci EMC. 2021 (cit. on p. 62).

[195] Island Pyrochemical Industries. *PRODUCT SPECIFICATION PBXN-5*. `http://www.islandpyrochemical.com/pbxn-5/`. Accessed: 01.06.2025 (cit. on p. 62).

[196] *FLEXIBLE LINEAR SHAPED CHARGE*. Pacsci EMC. 2021 (cit. on p. 62).

[197] *SPACE LINEAR SHAPED CHARGE DEVICE (LSCD) Technical Datasheet*. Pyroalliance, Ariane Group. 2025 (cit. on pp. 62, 71, 76).

[198] ESA. *Ariane 6: designed with sustainability in mind*. `https://www.esa.int/Enabling_Support/Space_Transportation/Ariane/Ariane_6_designed_with_sustainability_in_mind`. Accessed: 22.05.2025. May 2024 (cit. on p. 65).

[199] ESA. *Ariane 5*. `https://www.esa.int/Enabling_Support/Space_Transportation/Launch_vehicles/Ariane_5`. Accessed: 18.07.2025 (cit. on p. 66).

[200] Pierpaolo De Lazzari. «LCIA of launchers propulsive technologies: Ariane 5 ECA». MA thesis. Politecnico di Milano, 2021/2022 (cit. on pp. 67, 70, 71).

[201] ESA. *The engines of Ariane 6*. `https://www.esa.int/Enabling_Support/Space_Transportation/Ariane/The_engines_of_Ariane_6`. Accessed: 26.05.2025. June 2024 (cit. on pp. 68, 73, 80).

[202] ArianeGroup. *WHITE VULCAIN. How do the launcher parts withstand the incredible temperatures during launch?* Accessed: 14.06.2025. Oct. 2024 (cit. on p. 68).

[203] Linkedin (AIRBUS CRISA). *Ariane 6 : key equipment from Airbus Crisa.* `https://www.linkedin.com/pulse/ariane-6-key-equipment-from-airbus-crisa-airbus-crisa-c4nmf`. Accessed: 05.06.2025. July 2024 (cit. on pp. 69, 70).

[204] AIRBUS CRISA. *Ariane 6 launcher avionics*. `https://crisa.airbus.com/en/airbus-crisas-solutions/launcher-electronics/ariane-6-launcher-avionics`. Accessed: 05.06.2025. 2025 (cit. on pp. 69, 76).

[205] Gael Dée, Tillo Vanthuyne, Alessandro Potini, Ignasi Pardos, Guerric De Crombrugghe. «Electromechanical Thrust Vector Control Systems for the Vega-C launcher». In: *8TH EUROPEAN CONFERENCE FOR AERONAUTICS AND SPACE SCIENCES (EUCASS)* (July 2019) (cit. on p. 69).

[206] Borbala Rebeka David, Sean Spencer, Jeremy Miller, Sulaiman Almahmoud, Hussam Jouhara. «Comparative environmental life cycle assessment of conventional energy storage system and innovative thermal energy storage system». In: *International Journal of Thermofluids* 12.100116 (2021). ISSN: 2666-2027. DOI: `https://doi.org/10.1016/j.ijft.2021.100116` (cit. on p. 69).

[207] ArianeGroup. *Successful final test firing of the P120C solid rocket motor for Ariane 6 and Vega-C*. `https://ariane.group/en/news/successful-final-test-firing-of-the-p120c-solid-rocket-motor-for-ariane-6-and-vega-c/`. Accessed: 27.06.2025. Oct. 2020 (cit. on p. 70).

[208] Icpe. *LLTVAS LAT*. `https://www.icpe.ro/lltvas-lat/about/`. Accessed: 27.06.2025 (cit. on p. 70).

[209] ESA. *Start of the first Ariane 6 launch campaign*. `https://www.esa.int/Enabling_Support/Space_Transportation/Ariane/Start_of_the_first_Ariane_6_launch_campaign`. Accessed: 13.06.2025. Apr. 2024 (cit. on p. 72).

[210] ArianeGroup. *Ariane 6 Vinci engine: successful qualification tests*. `https://ariane.group/en/news/ariane-6-vinci-engine-successful-qualification-tests/`. Accessed: 13.06.2025. Oct. 2018 (cit. on pp. 72, 73).

[211] M. Vedrenne and A. L. Gonzalez. *ARIANE 5 VERIFICATION AND ASSOCIATED TEST FACILITIES*. Tech. rep. ESA, 1988 (cit. on p. 73).

[212] ESA. *Acoustic test for Ariane 6 upper stage*. `https://www.esa.int/ESA_Multimedia/Images/2024/11/Acoustic_test_for_Ariane_6_upper_stage`. Accessed: 07.07.2025. Nov. 2024 (cit. on p. 74).

[213] NASA. *Vibration Tests for Moon Rocket Help Ensure Safe*. `https://www.nasa.gov/missions/artemis/orion/vibration-tests-for-moon-rocket-help-ensure-safe-travels-on-road-to-space/`. Accessed: 07.07.2025. Aug. 2021 (cit. on p. 74).

[214] ESA. *Ariane 6 tests towards first flight*. `https://www.esa.int/ESA_Multimedia/Videos/2024/04/Ariane_6_tests_towards_first_flight`. Accessed: 13.06.2025. Apr. 2024 (cit. on p. 75).

[215] Thales Alenia Space. *Thales Alenia Space signs contract to start production of the Ariane 6 range safeguard system*. https://defence-industry.eu/thales-alenia-space-signs-contract-to-start-production-of-the-ariane-6-range-safeguard-system/. Accessed: 07.06.2025. June 2023 (cit. on p. 76).

[216] Sener. *Sener provides telemetry antenna for European Ariane 6 launcher*. https://www.group.sener/en/noticias/sener-aporta-la-antena-de-telemetria-para-el-lanzador-europeo-ariane-6/. Accessed: 07.06.2025. July 2024 (cit. on p. 76).

[217] ESA. *Shipping rockets: Ariane on board*. https://www.esa.int/Enabling_Support/Space_Transportation/Ariane/Shipping_rockets_Ariane_on_board. Accessed: 07.06.2025 (cit. on pp. 76, 77, 86).

[218] Google. *Maps*. https://maps.google.com. Accessed: 07.06.2025 (cit. on pp. 77, 78).

[219] *Sea Distances*. https://sea-distances.org/. Accessed: 08.06.2025 (cit. on p. 77).

[220] ESA. *Start of the first Ariane 6 launch campaign*. https://www.esa.int/Enabling_Support/Space_Transportation/Ariane/Start_of_the_first_Ariane_6_launch_campaign. Accessed: 10.06.2025. Apr. 2024 (cit. on pp. 78, 79).

[221] European Spaceflight. *CNES Qualifies Speedy New Ariane 6 Booster Transport Vehicle*. https://europeanspaceflight.com/cnes-qualifies-speedy-new-ariane-6-booster-transport-vehicle/. Accessed: 10.06.2025. Apr. 2024 (cit. on pp. 78, 79).

[222] ESA/ArianeGroup/Arianespace/CNES. *Ariane 6 launch complex processing*. Sept. 2021 (cit. on p. 79).

[223] Tyler F. M. Brown, Michele T. Bannister, Laura E. Revell, Timofei Sukhodolov, Eugene Rozanov. «Worldwide Rocket Launch Emissions 2019: An Inventory for Use in Global Models». In: *Earth and Space Science* (2024). DOI: https://doi.org/10.1029/2024EA003668 (cit. on p. 80).

[224] ESA. *Ariane 6 launch pad water deluge system test*. https://www.esa.int/ESA_Multimedia/Videos/2021/05/Ariane_6_launch_pad_water_deluge_system_test. Accessed: 17.07.2025. May 2021 (cit. on p. 80).

[225] MaiaSpace. «A review on the sustainability strategy in the development of a semi-reusable minilauncher». In: *Clean Space Days*. Oct. 2024 (cit. on pp. 88, 89).