Space Launch Vehicle Design

Conceptual design of rocket powered, vertical takeoff, fully expendable and first stage boostback space launch vehicles

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Declaration

I hereby declare that except where specific reference is made to the work of others, this thesis is my own work and contains nothing which is the outcome of work done in collaboration with others, except as specified in the text and Acknowledgements. Of specific note is the MAE 4350/4351 Senior Design class that ran from January 2017 through August 2017 which was tasked with a similar project. The work presented in my thesis did not borrow from their work except for adapting their method for the estimating of the zero-lift drag coefficient of grid fins, which is referenced appropriately in the text. The methodology used to size first stage boostback launch vehicles and simulate a return-to-launch-site trajectory is of my own design and work.

This thesis contains fewer than 51,000 words including appendices, bibliography, footnotes, tables and equations and has fewer than 90 figures.

David Woodward
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For my parents, for being OK with me putting myself through a completely different degree program after you had already helped me through college once. Your support with starting over, all the late nights helping with homework, and your love couldn’t ever be replaced.

Abstract

Engineering design can be broken down into three phases: conceptual design, preliminary design, and detailed design. During the conceptual design phase, several potential configurations are studied towards the identification of the baseline vehicle. Although the least amount of detail is known about the design during the early conceptual design phase, the decisions made during this phase lock in major features effecting life-cycle cost and overall product success.

As the next big space race begins, it is critically important to have a readily available tool for launch vehicle designers that is intuitive to use, easy to modify, cost effective, and provides correct results. This thesis details the creation of such a tool for use during the early conceptual design phase by analyzing existing launch vehicle design software and literature in order to adopt a best-practice approach to launch vehicle sizing. In addition to correctly sizing the vehicle calculating the initial parameters, the tool also determines the vehicle's basic geometric information and runs an ascent-to-orbit trajectory simulation to verify the design's validity. The tool is capable of sizing fully expendable space launch vehicles, fully expendable vehicles whose final stage is to be used for both ascent-to-orbit and additional orbital maneuvering after reaching the parking orbit, and vehicles whose first stage performs a self-recovery through the boostback and vertical landing method.

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Chapter 1. Introduction

On July 16, 1969, five of the most powerful liquid-fueled rocket engines ever designed, the F-1 engine, ignited. These were installed in the first of three stages used in the Saturn V launch vehicle. As the propellant tanks emptied out, the first stage was discarded. Shortly after, the five J-1 engines on the second stage came to life and the vehicle continued its ascent. Soon, those propellant tanks were depleted as well, and the second stage was also left behind. A lone J-2 engine installed on the third stage ignited, carrying the remainder of the vehicle into its parking orbit. A short time later, the third stage would be used again to propel the Apollo 11 Command Service Module and Lunar Module out of Earth's orbit and towards the first ever manned lunar landing.

In the late morning of November 23, 2015, over 45 years after Apollo 11, liquid hydrogen and liquid oxygen flowed into the combustion chamber of a single BE-3 engine and then ejected through the engine's nozzle. This force propelled Blue Origin's sub-orbital launch vehicle New Shepard and its unmanned astronaut module towards space where its trajectory crested at an altitude of 329,839 feet above sea level. At this point, most launch vehicle stages would come crashing to the Earth and be destroyed upon impact with the ground or water. However, this flight was different. New Shepard oriented itself to remain vertical as it re-entered the atmosphere and deployed drag brakes near the top of the rocket to slow itself down and used actuating fins near the base to steer. As it approached the ground, the BE-3 engine came alive once more, slowing the vehicle's descent until it gently touched down on the ground near a designated landing pad. The first sub-orbital launch vehicle stage, and first launch vehicle stage of any type, had launched and made a successful boostback recovery [1], [2], [3].

On the late evening of December 21, 2015, after the sun had set on Cape Canaveral, Florida, a lone launch vehicle stood at the Complex 40 launch pad. As the launch timer approached T-0, a final systems check verified all systems go on SpaceX's Falcon 9 launch. Propellant began to flow to the nine Merlin engines at the base of the first stage, and slowly the Falcon 9 was propelled towards the sky. After several minutes of first stage engine thrusting, the main engines cut off, and the second stage separated from the first. The second stage's lone Merlin engine pushed it away from the first and towards its target orbit. Pressurized gas flew from thrusters located at the top of the first stage, causing it to flip and point back to where it came. Shortly thereafter, it performed the first of three boostback burns to return to the launch site. During the course of this return flight, the stage flipped again and deployed a set of four grid fins to steer itself through the atmosphere. As it approached the ground, four legs lowered from the base of the stage as a slow burn decelerated it down to the ground. The first orbital-class launch vehicle stage had successfully returned from its mission. By the end of the middle of September 2017, SpaceX had successfully landed first-stage boosters sixteen times from various missions, including two of which that had been recovered and reused from previous launches [4], [5], [6], [7], [8].

At some point in the not-too-distant future, SpaceX's massive BFR Spaceship will sit on top of its booster. A large cluster of engines will ignite at the bottom, slowly carrying a number of human passengers and cargo towards a parking orbit. The vehicles will stage, and the booster will land back at the launch site. Once there, it will be refueled, and the BFR Tanker will be mated to its top. Launch will begin again, the vehicles will stage, and the BFR Tanker will rendezvous with the spacecraft to transfer its propellant. This process will be repeated several times to fill the Spaceship's propellant tanks and then, the Spaceship will leave its parking orbit. It won't be heading back towards the Earth, but instead will follow the path taken by several unmanned BFR Spaceships, all loaded with cargo, that have been sent out ahead of it. Their destination: Mars, to establish the first ever human colony on another celestial body. During all of these years of preparation, many hundreds of other launches will occur with various goals: launching new satellites, repairing or refueling existing satellites, transporting humankind to space for either work or pleasure, and studying and capturing near Earth objects for potential mining operations. A dozen companies, perhaps many more, will be performing these launches instead of the handful that regularly launch today.

Or at least, that's the plan. But will it actually happen?

The world of launch vehicles has changed significantly since the first man crossed the Kármán line and entered space. The yearning for space exploration has not changed and the demand for space access increases at a fast pace. As SpaceX attempts to challenge the launch vehicle provider juggernauts for an increasingly larger share of the market whilst other companies such as Blue Origin, Rocket Lab, and Vector Space Systems join the fray, it is critical that vehicle designers and technology forecasters have a tool available that can correctly perform the initial sizing of launch vehicles to ensure the new company's vehicle can competitively perform the mission for which it's designed. Additionally, with the advent of modern reusable launch vehicles a consistent sizing tool provides the ability to generate a variety of reusable vehicles from which the best business case may be selected and an expendable vehicle created to use for reference comparison purposes.

1.1 Phases of Engineering Design

Engineering design is typically broken up into three phases: conceptual design, preliminary design, and detail design.

1. **Conceptual Design**
   Conceptual design is the earliest of the design phases. The goal of this phase is to take promising ideas and expand them into one or more basic designs capable of completing the mission. At the end of the conceptual design phase, these design options are traded against each other and the best vehicle combination concerning performance and cost advances to the preliminary design phase. In the case of an aircraft, developing a basic

design means determining the primary geometric dimensions (length, wing span, size of control surfaces, etc.) as well as the grouping of the components into subsystems whose mass and volume requirements are estimated. The fundamentals to launch vehicle conceptual design are similar: the overall length and diameter of the stages are selected, and the masses and volumes of the payload, structure, and propulsion system requirements (engines, propellant tanks, pressurized gas tanks, reaction control system, etc.) are estimated. The goal is not to have numbers that will accurately match the actual results, but to arrive at the correct starting point in the overall design [9]. In most cases, hardware is not created in this phase. Instead, the design exists purely as numbers on a paper and possibly as a CAD model with only basic geometric information and generic shapes to indicate the volume requirements of subsystems. Exceptions to this include items such as a wind tunnel model, which may be created to confirm basic aerodynamic information, or critical components that are radically different from that which the company has prior experience, which are manufactured for testing. If any trajectory simulations are performed, they are basic simulations with only two or three degrees of freedom [9].

2. **Preliminary Design**
   The option(s) selected for further study move forward into the preliminary design phase. During preliminary design, additional trade studies are done to maximize the performance of the baseline design selected during conceptual design whilst minimizing cost. Estimates on the mass and volume requirements of the subsystems are refined, and individual components of the subsystem are developed into specific parts and part requirements. Basic parts are manufactured and used for testing to ensure they meet said requirements. If the selected design can still complete the mission requirements, it moves on to the detail design phase.

3. **Detailed Design**
   The final phase of design fleshes out all of the fine details of the system: every wire is connected; pipes and tubes are laid out; every system is built, tested, and retested; and entire vehicles are built and flown for certification.

It is important to note that over the course of the design phases there is an inverse relationship between the freedom of design choice versus the level of knowledge and total life-cycle cost. Early in the design process, major components can be in constant flux, and as such the final configuration is unknown. However, as the design leaves the conceptual stage, most of the vehicle’s layout and subsystems become fixed, and trade studies during preliminary design lock into place the remaining unknowns. The work that needs done during the detailed design phase is thus set during the first two phases, and as such the majority of the total life-cycle costs are determined before detailed design begins. In fact, as much as 80% of the total life-cycle costs are defined by the outcome of the conceptual design phase. See Figure 1.1 and Figure 1.2.

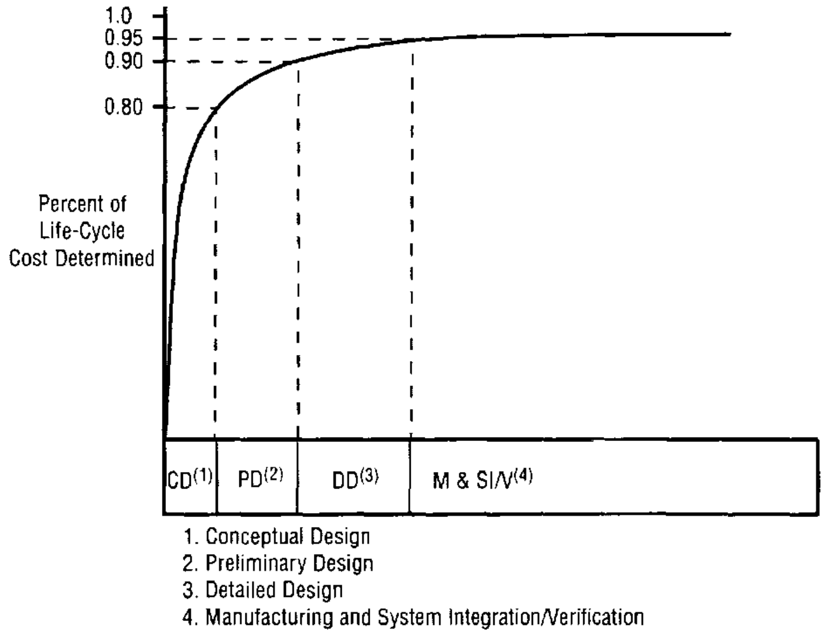

Figure 1.1 Life-Cycle Cost vs. Design Phase Stage [10]

Making major changes in the later design phases carries a large financial penalty: the change requires much of the work done the conceptual and preliminary design phases to be redone. This leads to a delay in the schedule for the detailed design phase, the need to hire additional workers in order to complete the work on time, or both. The larger the change, the more likely it is that other systems are affected by it and require their own rework, compounding the cost problem. As such, it is important that the work done during conceptual design phase eliminates designs that cannot complete the purpose the item is being created for and identifies any potential issues for review during the start of the preliminary design phase.

1.1.1 Discipline of Launch Vehicle Design

During all three design phases, the engineering team is broken into groups which are assigned to handle specific systems, or disciplines, of the design. The number and specific functions of these groups can vary depending not only on what is being designed but also on the company's culture and the person in charge of dividing up the engineering team into these disciplines. In *Integrated Design for Space Transportation Systems*, Suresh and Sivan identify eight primary disciplines in launch vehicle design. See Figure 1.3 [11].

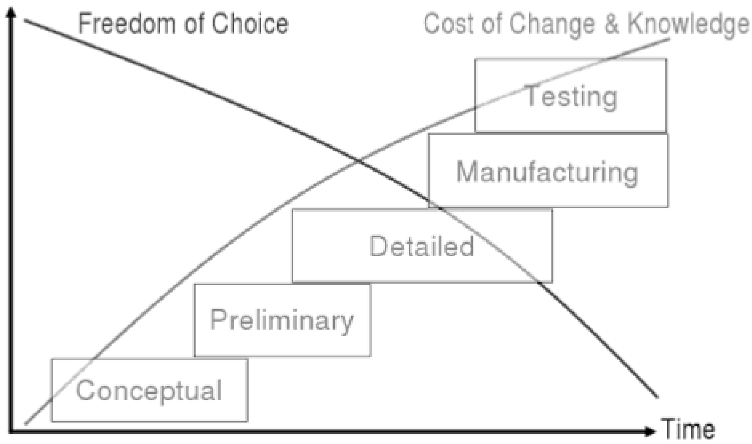

Figure 1.2 Freedom of Design, Cost, and Level of Knowledge vs. Time [12]

In engineering design, all disciplines are interconnected, and this is especially true for the design of launch vehicles. The requirements and performance of the various disciplines are influenced by each other and are driven by the mission for which the launch vehicle is being designed. A major change by one discipline or the mission requirements tends to affect the other disciplines. As such it is crucial to define what disciplines need to be considering during the early design phase and correctly understand how the interrelation between disciplines.

1.2 AVD Laboratory

The Aerospace Vehicle Design (AVD) Laboratory is one of several engineering research facilities at the University of Texas at Arlington, and one of the few academic labs in the country that emphasizes its research focus on the conceptual design phase of aerospace vehicles. The AVD Laboratory focuses on winged hypersonic vehicles and has had several government contracts in that area of research. In recent years, the AVD Laboratory has turned its efforts towards space access. To that end, in 2017 two doctoral graduates from the lab, Dr. Mark "Doug" Coley and Dr. Loveneesh Rana, completed their dissertations “On Space Program Planning” [13] and “Space Access Systems Design” [14], respectively. Additionally, Dr. Rana, Thomas McCall, and James Haley published three papers on winged reusable re-entry vehicles in September 2017 [15] [16], [17].

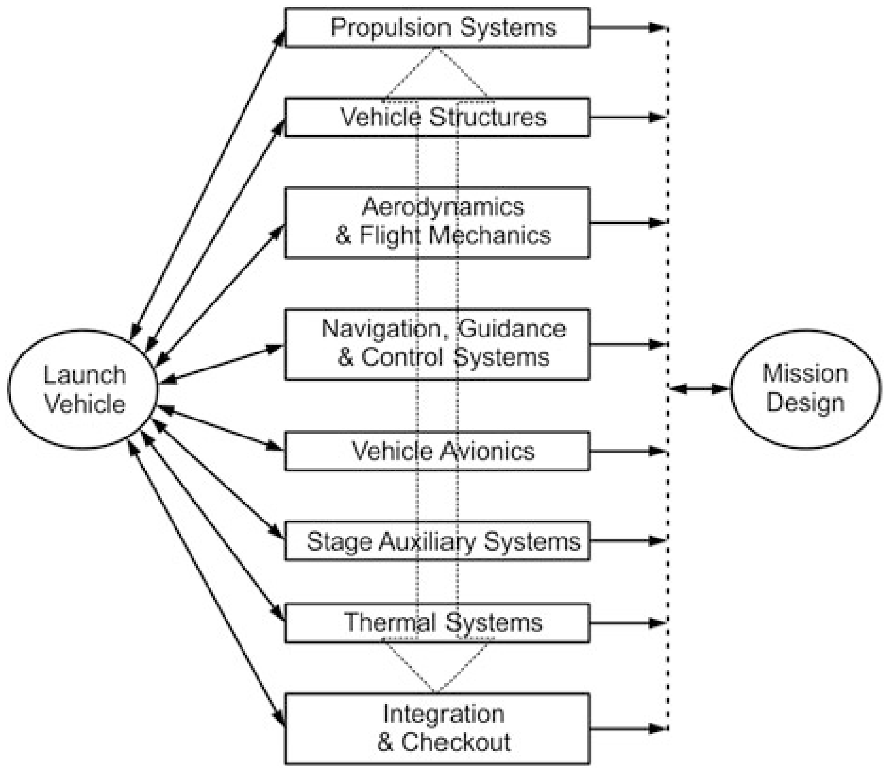

Figure 1.3 Launch Vehicle Major Systems [11]

The AVD Laboratory has a need to compare winged reusable space access vehicles with conventional expendable launch vehicles and the newly-proven boostback-reusable launch vehicles. While it is rather straightforward to obtain the performance data for existing vehicles, vehicles are designed for a specific mission and comparing vehicles with different missions carries risk of drawing inaccurate conclusions. Therefore it is important to a design tool able to consistently size an expendable or boostback-reusable launch vehicle in a way that will enable to AVD Laboratory to make comparisons between these and winged, fully reusable space vehicles designed for the same mission. Thus, this Master's thesis research aim has been from the outset to develop a sizing tool for fully expendable, fully expendable with a transfer stage, and partially-reusable launch vehicles with a first stage boostback. This will be developed such that in the future it can be integrated with existing and upcoming AVD-developed software for the purpose of comparing classical expendable and boostback reusable launch vehicles with their winged counterparts.

1.3 Research Objective

The objectives of this research project are:

- Perform a detailed literature search on space launch vehicle design literature to develop a best-practice approach to creating a new design tool;

- Create the new launch vehicle design software in a manner that it is easy to use and new features can be integrated in without difficulty;

- Verify the software's capability through three case studies.

Chapter 2. Literature Survey

For this thesis, a three part literature review has been performed. The first part focuses on the history of US launch vehicles. The second part covers existing launch vehicle design software and their features. The third and final part investigates existing launch vehicle design literature addressing the overall design methodology.

2.1 History of US Launch Vehicles

A timeline of the first successful launch of major US launch vehicles is presented in Figure 2.1. It contains launches starting with Redstone in 1953 to planned launch dates for new launch vehicles through 2020. As indicated by the timeline, the history of US launch vehicles can be divided into separate eras:

1. **Era of Ballistic Missile Launch Vehicles**
   The first launch vehicles in the United States, and in the former Soviet Union, were not initially intended for space access. These early launch vehicles were originally intercontinental ballistic missiles (ICBMs) and/or intermediate-range ballistic missiles (IRBMs) which were modified to launch a payload into space. In many cases, a single missile was not sufficient to create the launch vehicle; instead, combinations of ICBMs and IRBMs stages put together created the launch vehicle. It wasn't until the Saturn IB that a launch vehicle was designed from scratch with space access as the primary design objective. [18], [19], [20]

2. **Era of Saturn Launch Vehicles**
   The second era is marked by the successful launch of the first member of the Saturn launch vehicle family, the Saturn IB, in February of 1966. The Saturn IB, and the entire Saturn family, is unique in that it is the first clean-sheet design of an expendable space launch vehicle: it was designed from the start to be used for space access. The Saturn rockets were used for the Apollo program, the Skylab program, and the joint US-USSR Apollo-Soyuz Test Project. As can be seen in the timeline in Figure 2.1, only one other space launch vehicle, the Titan IIID, was introduced during this time as the existing ICBM/IRBM launch vehicles and Saturn family were able to handle the overwhelming majority of space access needs during this time. [18], [19], [21]

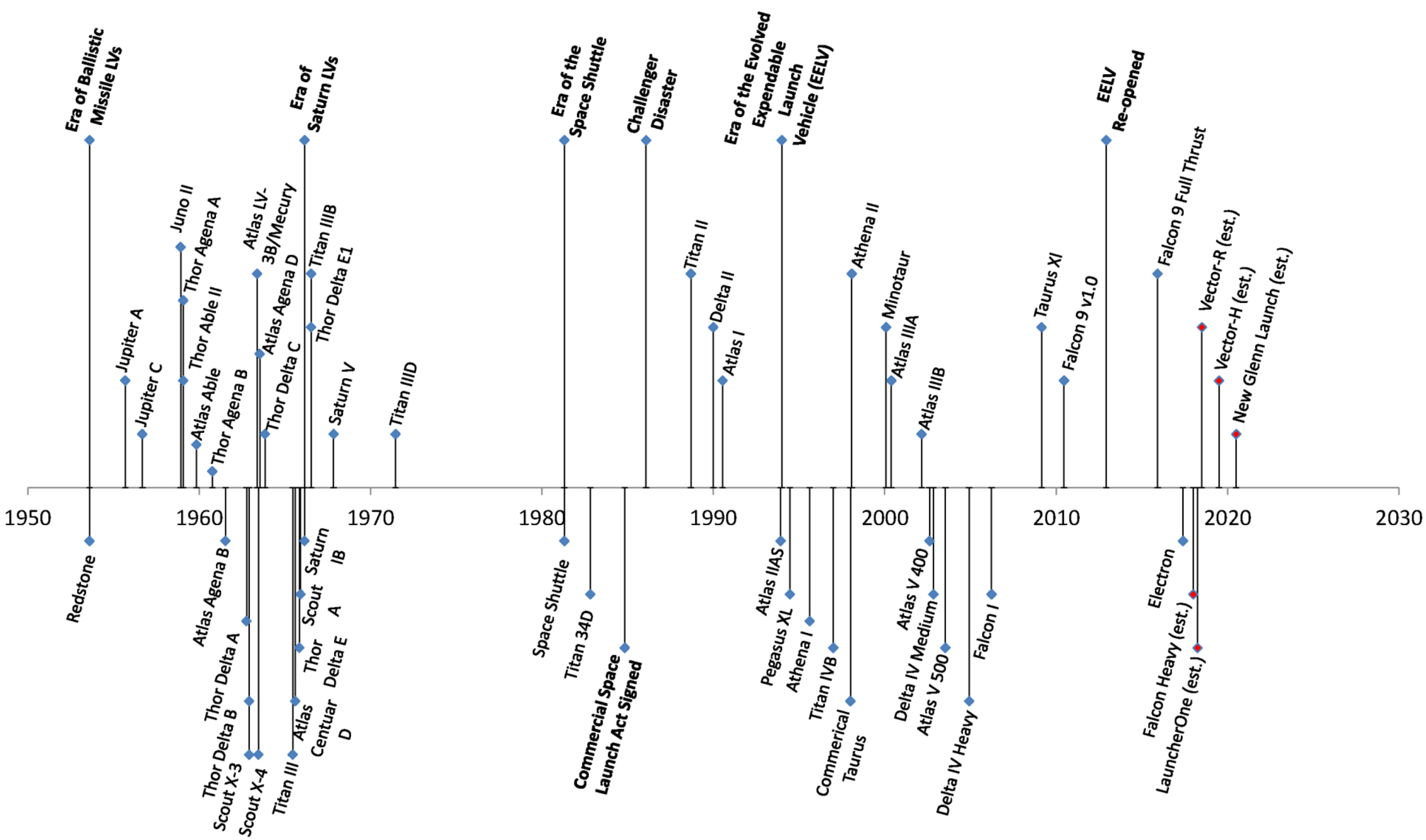

Figure 2.1. Timeline of the First Successful Launch of US Launch Vehicles

3. Era of the Space Shuttle

The third era began in 1981 with the first flight of the Space Shuttle. With it came the promise of significant reductions in the cost of space access through partial reusability, significantly more frequent flights, and the operational advantage of the Space Shuttle's aircraft-like horizontal runway landing. Such promise resulted in overwhelming support for the Space Shuttle, reaching levels such that early on in the design process the US government formally adopted it as the primary launch vehicle service for all payloads to be launched from the country. It was thought that any other launch vehicles, such as the new Titan 34D, would only supplement the Shuttle's capability or fill in when the Shuttle was unavailable.

Unfortunately, this confidence was misguided. While the Space Shuttle was a technological success, it was an operational failure. Launch rates were well below what was expected and the refurbishment effort required between flights drastically exceeded cost projection. As a consequence, the cost to launch a payload to space failed to decrease overall. After the Challenger disaster in 1986, the US government shifted in policy away from using the Space Shuttle as the primary tool for space access, and a resurgence of expendable launch vehicles began with the Titan II, Delta II, and Atlas I launching in the following years. However, these vehicles were derivatives of launch vehicles that were originally modified ICBMs, and no new clean sheet designs emerged [18], [19], [22], [23].

4. Era of the Evolved Expendable Launch Vehicle (EELV): Return of the Ballistic Missile Evolution

Although the Challenger disaster started the reintroduction of expendable launch vehicles for regular access to space, the next era of launch vehicle history didn't begin until the start of the EELV competition. Originally conceived in 1994 and revised in 1997, the goal of the EELV was to encourage innovation *"...to design a new, reliable national launch capability and minimize or eliminate the inherent shortcomings…"* of the available launch vehicles and to switch to a more commercial-like procurement of the launch vehicle (i.e., launch vehicles were simply purchased by the government, and they had minimal to no intervention in the design process) [24]. While the recession in the early 2000s significantly curtailed the commercial launch market whilst the government interference in the design process was much larger than originally intended, the program resulted in the creation of the Atlas V and Delta IV launch vehicles, both of which have multiple variants. While these were not clean-sheet designs, they met the requirements of the EELV contract and the USAF purchased over 20 flights at the conclusion of the first EELV competition [24], [25], [19].

While the Atlas V and Delta IV are the only two vehicles that came out of the EELV contract, a number of other launch vehicles were developed during this time. The Pegasus XL and the Falcon family are unique among these because they were developed free of government intervention during the design process, making them the first purely commercial ventures into space launch vehicles. Additionally, the Pegasus XL and Falcon family have been clean-sheet designs from the outset, a noticeable difference from the other new vehicles from this time period.

In 2012 an extension of the EELV program was announced. The announcement authorized the United States Air Force to purchase up to thirty-six launches from United Launch Alliance (ULA) and up to fourteen to be purchased on a competitive basis. After half of the competitive launches were deferred in 2014, SpaceX sued the Air Force in an attempt to open up the thirty-six launches awarded to ULA to open marketplace competition. The lawsuit was settled in 2015 with an agreement to expedite the certification process for SpaceX's rockets, enabling the company to fly government payloads and offer launch services through 2017 to the Air Force while ULA maintained its thirty-six launch contractor. [26], [27], [28], [29], [30], [31], [32]

5. **Era of the Commercial Launch Vehicle?**

While not labeled on the timeline, a fifth era of a launch vehicles is about to begin. In the next several years, multiple launch vehicle companies are planning on entering or expanding their position in the market with small-, medium-, and heavy-lift launch vehicles. SpaceX plans a launch of the partially reusable Falcon Heavy at the start of 2018 whilst Blue Origin intends to make their debut with the partially reusable New Glenn in 2020. [33], [34] Several small-lift companies, such as Rocket Lab and Vector Space Systems intend to fly payloads into orbit before the end of 2018 [35], [36], [37], [38]. Whether or not these companies will be successful has to be seen. Since the beginning of the space age many proposed ideas and companies having come and gone without success. One such company, XCOR Aerospace, filed for bankruptcy in November 2017 [39].

What makes the era of the commercial launch vehicle unique is the quantity of clean-sheet launch vehicle designs that are proposed and are either already in service or will be performing certification flights in the next few years. If these designs are successful, the fifth era of US Launch Vehicles, an era that promises reduced cost, decreased waiting time to launch, and vehicle reusability, will begin. However, if a company's design is inadequate, they will either be unable to get their vehicle into production or be driven into bankruptcy by more economically efficient vehicles. This emphasizes the need for a launch vehicle design tool which is able to arrive at a correct baseline design to maximize its market success chance.

2.1.1 History of Boostback Reusability

It was clear from the dawn of the space age that discarding an entire launch vehicle after each mission was an expensive proposition: each time someone wanted to place an object into space, they had to manufacture an entire new vehicle. In theory, costs could be brought down significantly if the launch vehicle could be recovered and re-flown. Many of the original designs for a reusable launch vehicle used the same concept behind the Space Shuttle: the vehicle would perform a rocket-powered ascent, then glide to a runway, and perform a horizontal landing. An alternative reusability method that was also proposed was the boostback recovery (also called a "propulsive return" or "tossback recovery").

The idea behind boostback recovery is to use a combination of gravity, aerodynamic forces, and rocket propulsion to slow down and land the vehicle vertically:

- **Gravity** is used to slow the vehicle's ascent and begin accelerating the vehicle back towards the ground without the use of propellants.

- Similarly, **aerodynamic forces** reduce the vehicle's descent velocity to levels that can be canceled out by a small amount of propellants. Movable aerodynamic control surfaces can also be used to steer the vehicle to the desired landing site.

- **Rocket propulsion** is used at landing to slow the vehicle's descent velocity to zero just before landing. Additionally, the engines can be used to provide steering in greater magnitudes than what is provided by aerodynamics as well as provide control where the atmosphere is too thin for control surfaces to be effective through the use of thrust vector control hardware or, if there are multiple engines, firing specific engines to provide asymmetric thrust.

In the 1960s, several of the major players in the aerospace community such as Phil Bono and Dietrich Koelle proposed several different designs for boostback vehicles, such as Rhombus, BETA, and SASSTO. A few commercial companies also ventured into the idea of boostback vehicles, such as McDonnell Douglas's tossback booster in the late 1970s and early 1980s. These one and two stage vehicle designs deviated from the standard cylindrical shape of launch vehicles towards a more short, conical-style body with the larger end at the base. This larger volume at the bottom was used to store landing legs as well as the multitude of engines (the original BETA design called for at least twelve engines and Rombus had thirty-six), and, most notably, was used for aerodynamic purposes. The wide base provided a large surface area for descent to slow the vehicle down with aerodynamic forces while a heat shield protected the primary structure from the aerothermal heating that would occur. [40], [41], [42], [43], [44] Figure 2.2 shows concept art and a preliminary drawing for the four mentioned boostback concepts.

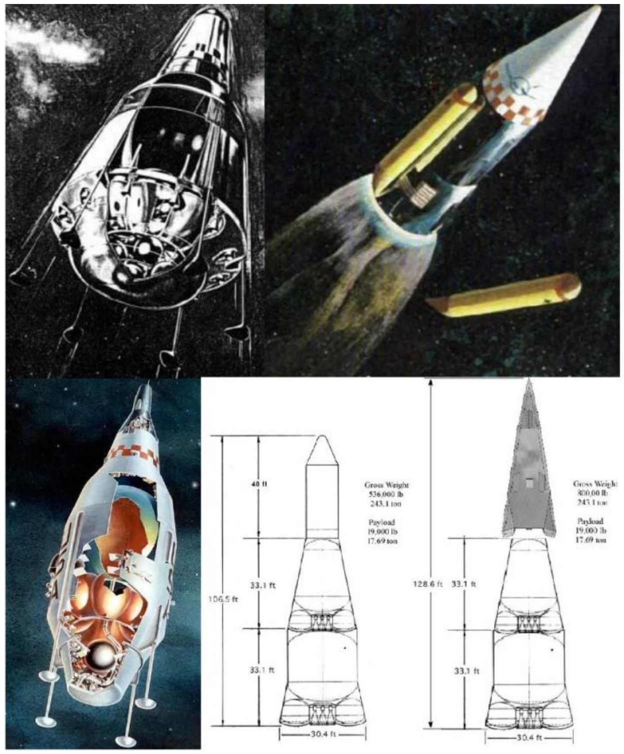

Figure 2.2. Several Boostback Vehicles Concepts. Starting in the Top Left and Going Clockwise: BETA [42], Rombus [43], McDonnell Douglas Tossback Booster [45], and SASSTO [44]

Boostback reusability remained an on-paper idea until the Delta Clipper DC-X. The DC-X, shown in Figure 2.3, was built and first flow in the early-1990s as a small-scale demonstrator to test the feasibility of the single-stage-to-orbit, boostback reusable concept. The vehicle performed well during the low-altitude tests and had eight successful flights demonstrating its ability to ascent, hover, strafe side-to-side, perform a controlled descent, and land. However, the vehicle was destroyed when one of the four landing legs failed to deploy, resulting in the vehicle tipping over when it attempted to land. With the crash destroying the only test vehicle and a lack of support for additional funding to build a second test article, the Delta Clipper program was shelved [46], [47], [48], [49].

Around the same time as the DC-X test flights, Kistler Aerospace (later Rocketplane Kistler) began work on the Kistler K-1. The Kistler K-1 was a two-stage launch vehicle designed to use a combination of boostback and parachute reusability (see Figure 2.4). Kistler Aerospace eventually entered bankruptcy after being unable to provide private financing to match the funding it was receiving from NASA. [50], [51], [52]

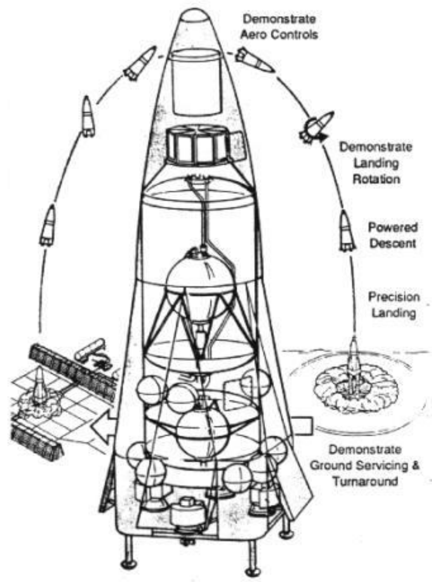

Figure 2.3. DC-X Interior and Trajectory [49]

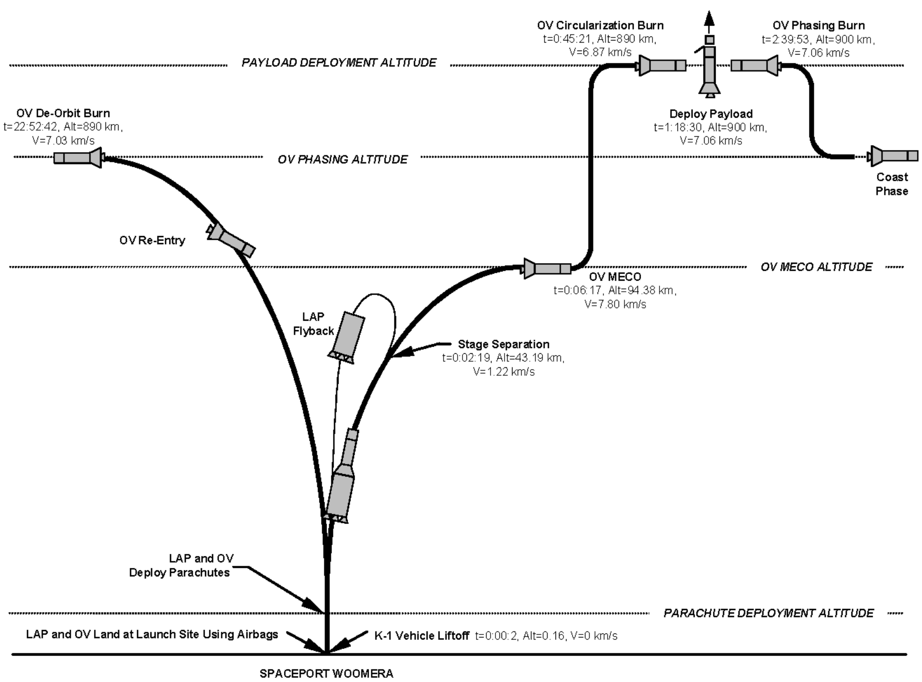

Figure 2.4. Kistler K-1 Flight Profile [53]

Boostback reusability would have faded away once more, but another promising company, Blue Origin, began experimenting with the technology around the same time as the demise of the Kistler K-1. Blue Origin built and flew boostback technology demonstrators in 2005 with Charon, a jet-powered VTVL vehicle, and 2006 with Goddard, a rocket-powered.

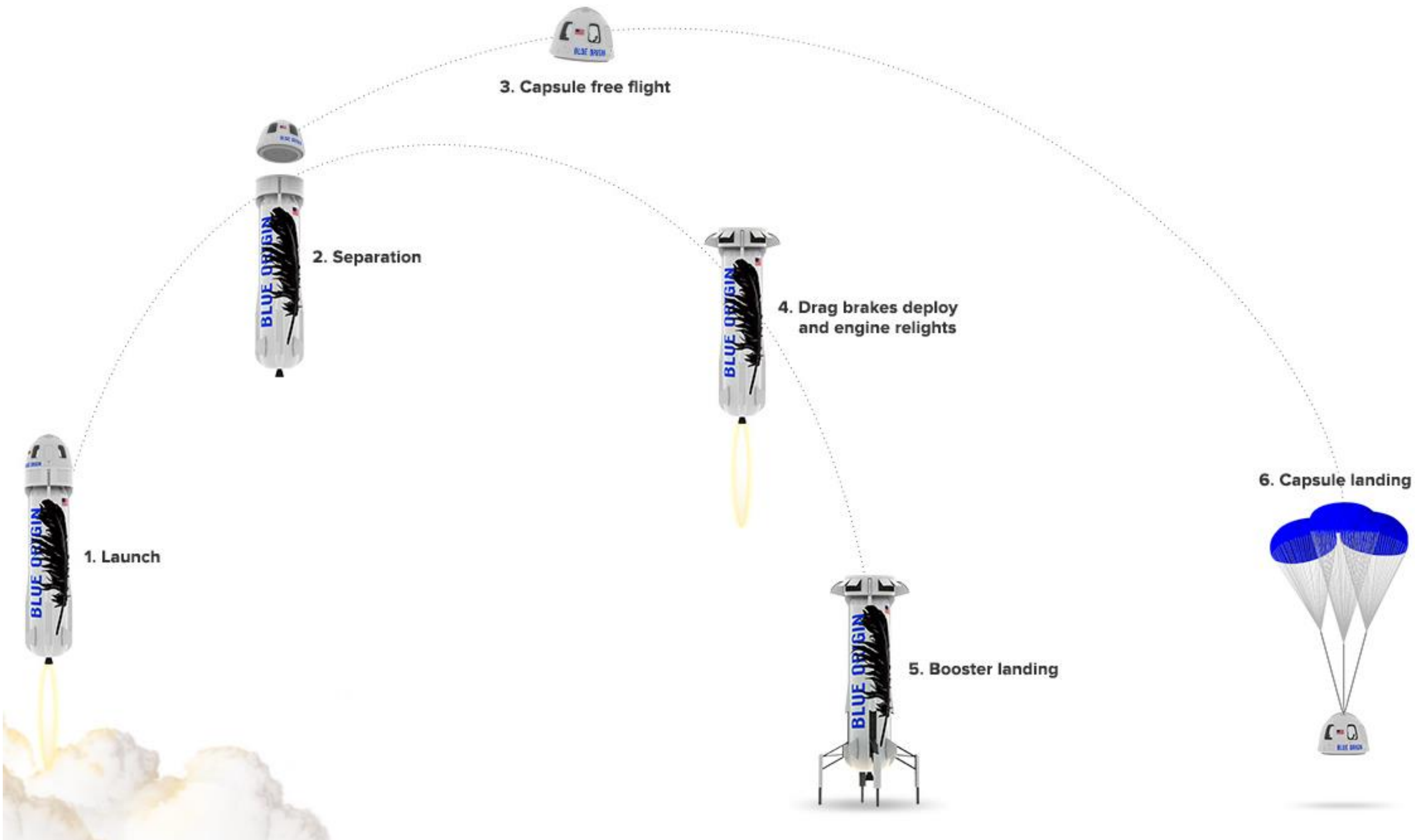

Figure 2.5. New Shepard Trajectory [54]

VTVL vehicle. New Shepard was the culmination of these projects. New Shepard, seen in Figure 2.5, is a sub-orbital vehicle whose mission is to fly scientific payloads and eventually tourists into space.   Blue Origin ran launch and boostback landing tests on New Shepard, and on November 23, 2015, New Shepard successfully flew to the edge of space, released its payload capsule, and then landed safely back at the launch site [1], [55], [56], [57], [58], [59]

Blue Origin was not the only company looking into boostback technology. In 2011, SpaceX announced its intent to build a boostback flight demonstrator called Grasshopper. Grasshopper was a ten-story tall, cylindrical rocket with four landing legs. SpaceX drew heavily on data from the DC-X program, and in fact the test program ran very similar to the DC-X flights: early flights were short hops up and down, and later flights involved hovering, flying side-to-side, and flying with wind gusts all while increasing the peak altitude of each hop. SpaceX wrapped up the program in 2013 and began integrating the necessary design changes into the next iteration of the Falcon 9, and on December 21, 2015, SpaceX successfully landed a first stage of the Falcon 9 after delivering a payload into orbit. As of September 2017 SpaceX has since landed sixteen first stages from missions [59], [60], [61], [62].

Figure 2.6. Grasshopper in Flight [63]

2.2 Launch Vehicle Fundamentals

Encyclopedia Britannica defines a launch vehicle *as "…a rocket-powered vehicle used to transport a spacecraft beyond Earth's atmosphere, either into orbit around Earth or to some* *other destination in outer space…*" [64]. Launch vehicles are systems that are primarily cylindrical in shape and traditionally powered by rocket engines. The rockets create the thrust to move the launch vehicle by combusting an oxidizer and fuel together and accelerating the resulting gases out of nozzle out of the back of the vehicle, see Figure 2.7 [65]. Historically, nearly all launch vehicles are expendable systems that either fall into the ocean or are left in space once their mission has been completed.

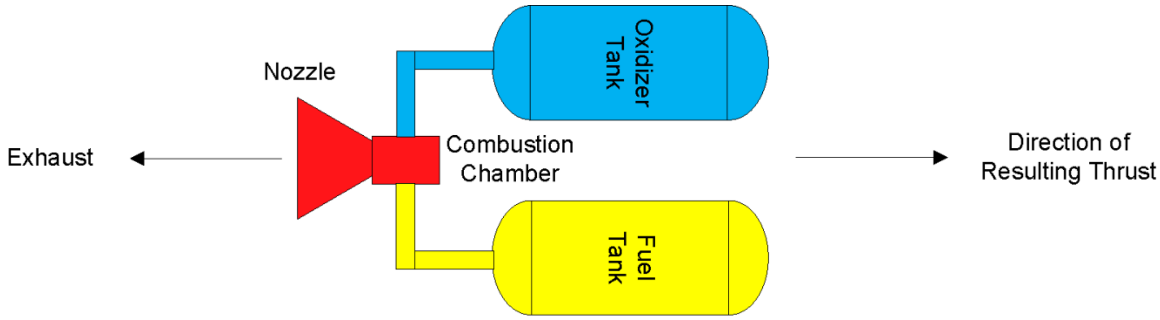

Figure 2.7. Simplified Liquid Rocket Engine

NASA classifies launch vehicles based on its payload capacity to Low Earth Orbit (LEO) [66]. Table 2.1 identifies these 4 categories. [66]

Table 2.1 Classification of Launch Vehicles

Classification	Payload to Leo	
	Tons	Kg
Small-Lift Launch Vehicle (SLLV)	0 - 2	0 - 1,814
Medium-Lift Launch Vehicle (MLLV)	2 - 20	1,814 - 18,144
Heavy-Lift Launch Vehicle (HLLV)	20 - 50	18,143.7 - 45,360
Super Heavy-Lift Launch Vehicle (SHLLV)	50+	45,360

As displayed in Figure 2.8, the components of a launch vehicle are subdivided into three major groups:

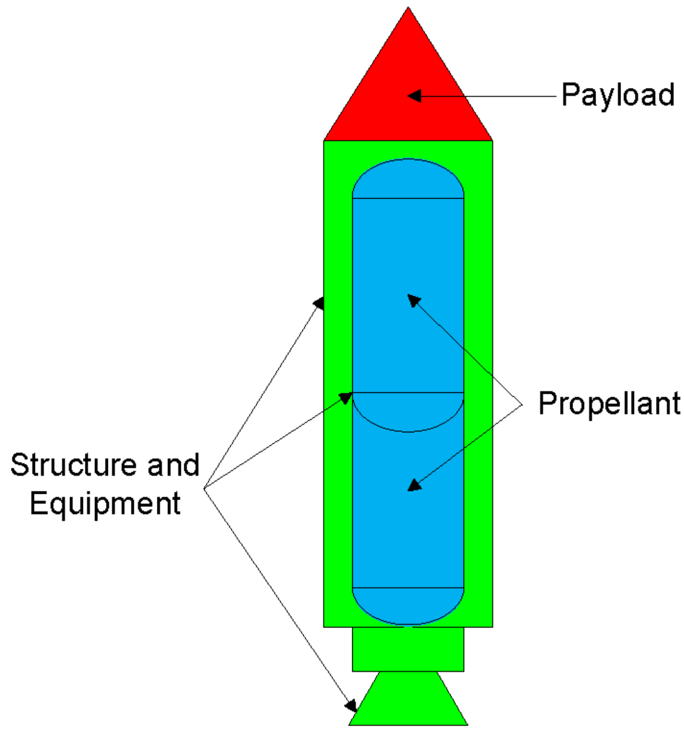

Figure 2.8 Simplified Launch Vehicle Components

1. **Payload**, which is what the launch vehicle is ferrying up into space. The payload is typically provided at the start of the launch vehicle design process as estimated mass and geometric dimensions.

2. **Propellant**, which is the fuel and oxidizer used by the engines to propel the vehicle and takes up the majority of the launch vehicle's total mass and volume.

3. **Structures and Equipment**. This includes everything that is not the payload or propellant: the propellant tanks; engines; vehicle structure; avionics and other on-board computers; fairing for the payload; etc. Some of these parts can be purchased off-the-shelf while others must be designed for a specific launch vehicle.

In order to place an object into orbit, the launch vehicle must increase its velocity until it is equal to the velocity required to maintain that orbit while overcoming various losses, such as losses from gravity and drag during flight, and reaching the orbital altitude. The rocket equation, first derived by Konstantin Tsiolkovsky, relates this change in velocity to engine performance and vehicle mass with

$$\Delta V = g_o I_{sp} \ln\left(\frac{m_{initial}}{m_{final}}\right) \tag{2.1}$$

where:

ΔV is the change in velocity produced.

g_o is the acceleration due to gravity on Earth at sea level.

I_{sp} is the average specific impulse of the vehicle's engine(s). I_{sp} varies based on the specific oxidizer and fuel combination as well as the overall efficiency of the rocket engine.

$m_{initial}$ is the mass of the entire launch vehicle system at the beginning of the flight and is defined as

$$m_{initial} = m_{payload} + m_{propellant} + m_{structures\ and\ equipment} \tag{2.2}$$

m_{final} is the mass of the launch vehicle system at the end of powered flight. The final mass is defined as

$$m_{final} = m_{payload} + m_{structures\ and\ equipment} \tag{2.3}$$

The ratio of initial to final mass is also referred to as the vehicle's mass ratio.

Table 2.2 shows the total ΔV required to reach LEO, Geosynchronous Transfer Orbit (GTO), Geosynchronous Earth Orbit (GEO), and Low Lunar Orbit from the Earth's surface [67]. As can be seen, the velocity changes are significant: if an aircraft could be increased to and maintain the speeds necessary to reach LEO, it would travel the nearly 4 km air distance from Los Angeles to New York in less than half a second. The energy required to obtain that velocity change is contained in the potential energy of the propellants and must be equally high.

Table 2.2. ΔV Required to Reach Various Orbits

Orbit Name	Total ΔV Required From Earth's Surface (km/s)
Leo Earth Orbit (LEO)	9.7
Geosynchronous Transfer Orbit (GTO)	12.2
Geosynchronous Earth Orbit (GEO)	13.8
Low Lunar Orbit	13.6

Per the rocket equation, there are two methods by which the energy requirements could be reduced: increasing I_{sp} and reducing the mass ratio.

Increasing I_{sp} can be done by increasing engine performance, switching to a fuel and oxidizer combination that produces a higher I_{sp}, or changing propulsion system types to something more effective. Chemical rocket propulsion has an I_{sp} range of 200 to over 450 s, depending type of propellants used (solid, liquid, or hybrid), the specific fuel and oxidizer combination, and the efficiency of the engine itself. More exotic forms of propulsion such as nuclear thermal rocket propulsion can push I_{sp} above 800 s but have their own technical as well as legal complications and have never actually been tested in flight [65], [68], [69], [70].

Reducing mass ratio has its own challenges. The payload mass is a starting requirement, so it cannot be changed and, in fact may increase as was the case with the Mercury, Gemini, and Apollo spacecraft [71]. The propellant mass is derived from the energy required to produce the necessary ΔV, so it cannot be directly changed either. This only leaves the structure and equipment mass as eligible for reduction. While the advances in composites has assisted with reducing structure mass, to date it has not been brought down low enough for a single, large vehicle (referred to as single-stage-to-orbit, SSTO) to reach orbit while carrying a payload.

This creates a problem: with I_{sp} capped by technological and chemical limitations as well as legal complications, and a reduction in the mass ratio also handicapped by technological progress, it appeared early on that space access was a nearly impossible task. However, a rather simple solution was proposed to the problem: break the launch vehicle up into several smaller vehicles, put them together, and when one of these parts was no longer required, get rid of it. This is referred to as staging and was originally suggested independently by Konstantin Tsiolkovsky, Robert Goddard, and Hermann Oberth, the fathers of space launch vehicles, in the early 1900’s [72], [73], [74].

2.2.1 Staging

As propellant is consumed during flight, the empty fuel and oxidizer tank volume as well as any structure that houses that part of the tanks effectively becomes dead weight: it no longer provides any benefit, yet energy must still be expended to carry it to a higher altitude. Additionally, during the course of the flight, the vehicle's flight path angle γ, the angle between the direction the vehicle is flying and the ground, slowly changes from being perpendicular to the ground during takeoff to being parallel to the ground. When γ is high there are significant losses due to gravity which the vehicle must overcome with high thrust, see Figure 2.9. As both the γ and the launch vehicle's mass decrease, the losses due to gravity also decrease. This leads to a reduction in the thrust requirement to continue to the target orbit, and thus some of the engines are no longer needed.

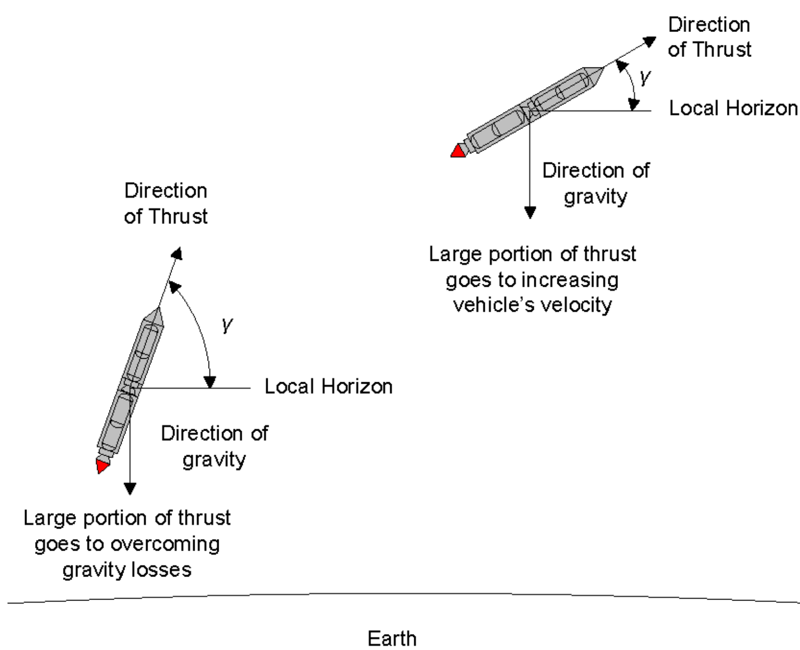

Figure 2.9 Effect of γ on the Losses Due to Gravity

By breaking the vehicle into segments, a spent segment can be discarded at a predetermined time to reduce the dead weight being carried into space and maximize the launch vehicle's energy towards putting its payload into orbit. These segments are called *stages*, and each stage contains some fraction of the propellant required to reach orbit and its own set of propellant tanks, engines, structure, and equipment. A vehicle with multiple stages is called a *multi-stage launch vehicle*. In the case of a multi-stage launch vehicle, when a stage is nearing depletion of its propellants the engines are shut down and the stage separates from the rest of the vehicle. The next stage of the vehicle ignites its engines moments afterwards and continues on its way to orbit while the separated stage is typically burned up in the atmosphere or is discarded into the ocean.

Using a multi-stage launch vehicle changes the way the total ΔV is calculated. Instead of calculating the total ΔV directly, the ΔV of each stage is calculated and summed together. Thus:

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

$$\Delta V_{total} = g_o I_{sp,stage\ 1} \ln\left(\frac{m_{initial,stage\ 1}}{m_{final,stage\ 1}}\right) + g_o I_{sp,stage\ 2} \ln\left(\frac{m_{initial,stage\ 2}}{m_{final,stage\ 2}}\right) + \cdots \tag{2.4}$$

where the definitions for $m_{initial}$ and $m_{structures\ and\ equipment}$ are the same as before but

$$m_{final,\ stage\ 1} = m_{payload,\ stage\ 1} + m_{structures\ and\ equipment,\ stage\ 1}$$

and

$$m_{payload,\ stage\ 1} = m_{intial,\ stage\ 2}$$

The infographic by Tom Logsdon in Figure 2.10 shows how ΔV significantly increases when going from one stage to two stages despite the initial launch weight, payload weight, and I_{sp} remaining the same.

In order to preserve the total ΔV produced, staging should occur when the current stage is out of propellant and the next stage should light its engines almost immediately afterwards. Figure 2.11 shows how the maximum altitude changes when using staging and how a higher altitude is achieved by staging when the active stage is out of propellant rather than when reaching apogee.

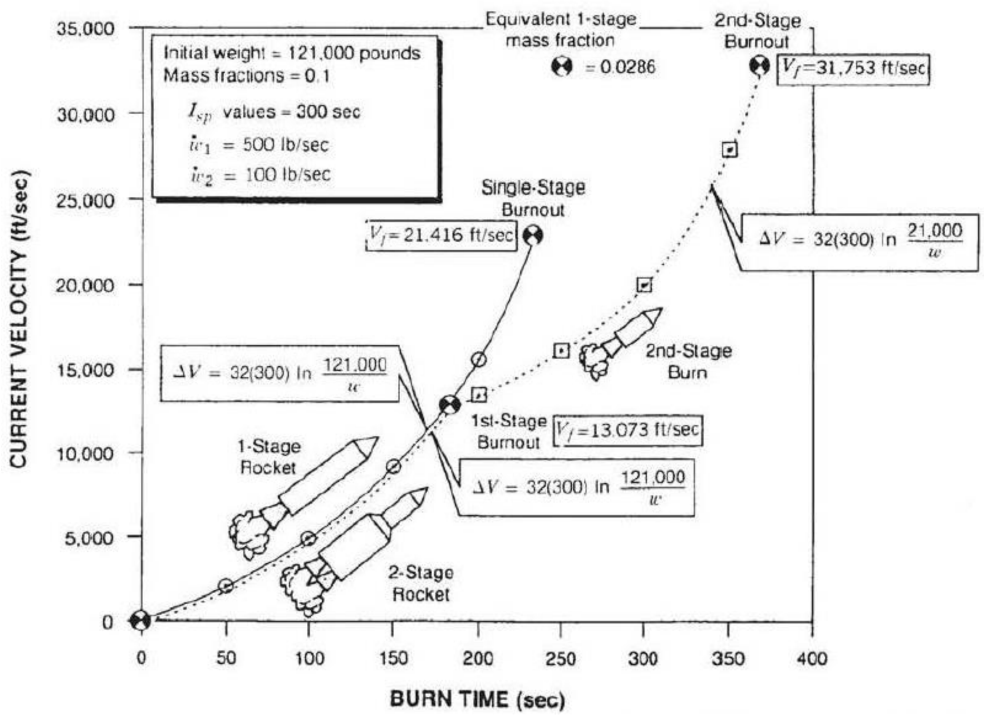

Figure 2.10. Benefits of staging [75]

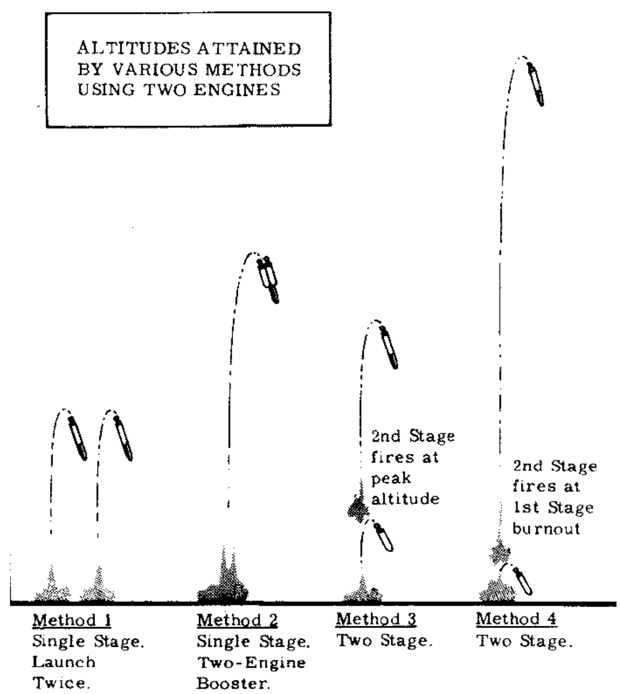

Figure 2.11. Comparison of Altitudes Reached with Staging [76]

As displayed in Figure 2.12, four unique methods of staging exist. They are:

1. **Serial staging**
   In serial staging, the stages are stacked on top of each other in a single line, and only one stage's engines are active at a time. If the diameter of the vehicle changes between stages, the top stages have the smaller diameter while the bottom stages have the larger diameter. This is the most commonly seen type of launch vehicle staging. Examples include the Gemini Launch Vehicle, Saturn V, and Falcon 9.

2. **Droppable engines**
   With this type of staging, engines are discarded during the flight once γ and/or mass have decreased such that the thrust requirement is reduced enough for some of the engines to no longer be necessary. This type of staging was used for the Mercury-Atlas but has not reappeared in US launch vehicles since then [19], [11].

   More recent proposals such as ULA's Sensible Modular Autonomous Return Technology (SMART) concept call for droppable engines as a potential method for reducing cost, but thus far hasn't been implemented [48], [77], [78], [79].

3. **Parallel staging**
   Several strap-on boosters are attached to the sides of the launch vehicle in a pattern that is symmetric across one or more planes. These boosters fire in conjunction with the primary launch vehicle core at launch, but finish burning their propellant before the core finishes and are discarded while the core stage continues its burn. One of the major benefits to parallel staging is the increase to the launch thrust-to-weight ratio by increasing the number of active engines. Parallel staging is less common than serial staging but is still frequently used today, especially in larger launch vehicle such as the Atlas V Delta IV Heavy, and Soyuz.

4. **Piggyback staging**
   Piggyback staging is similar to parallel staging, except only a single stage is strapped onto a larger stage and the payload is carried in the strapped-on stage. Once the main stage reaches burnout, the smaller stage detaches and continues to orbit. In some cases, the larger stage isn't even a rocket. For example, the Space Shuttle used a large external propellant tank as part of its staging method, and this external tank carried the propellant for use in the engines on the Space Shuttle. Once the external tank was depleted, it was discarded from the orbiter.

   Piggyback staging from another vehicle moving at transonic, supersonic, or hypersonic speeds has been proposed as a means of partial or full reusability, although this is often referred to as an air-launch and considered either a half stage or not a stage at all. An example of piggyback/air-launch staging is the Pegasus XL which is launched from an L-1011 Stargazer instead of from a launch site (see Figure 2.13) [80]. A second example includes the air-launch platform Stratolaunch which completed engine testing in September 2017 and low-speed runaway in December 2017 [81], [82].

Like droppable engines, this type of staging is rarely used.

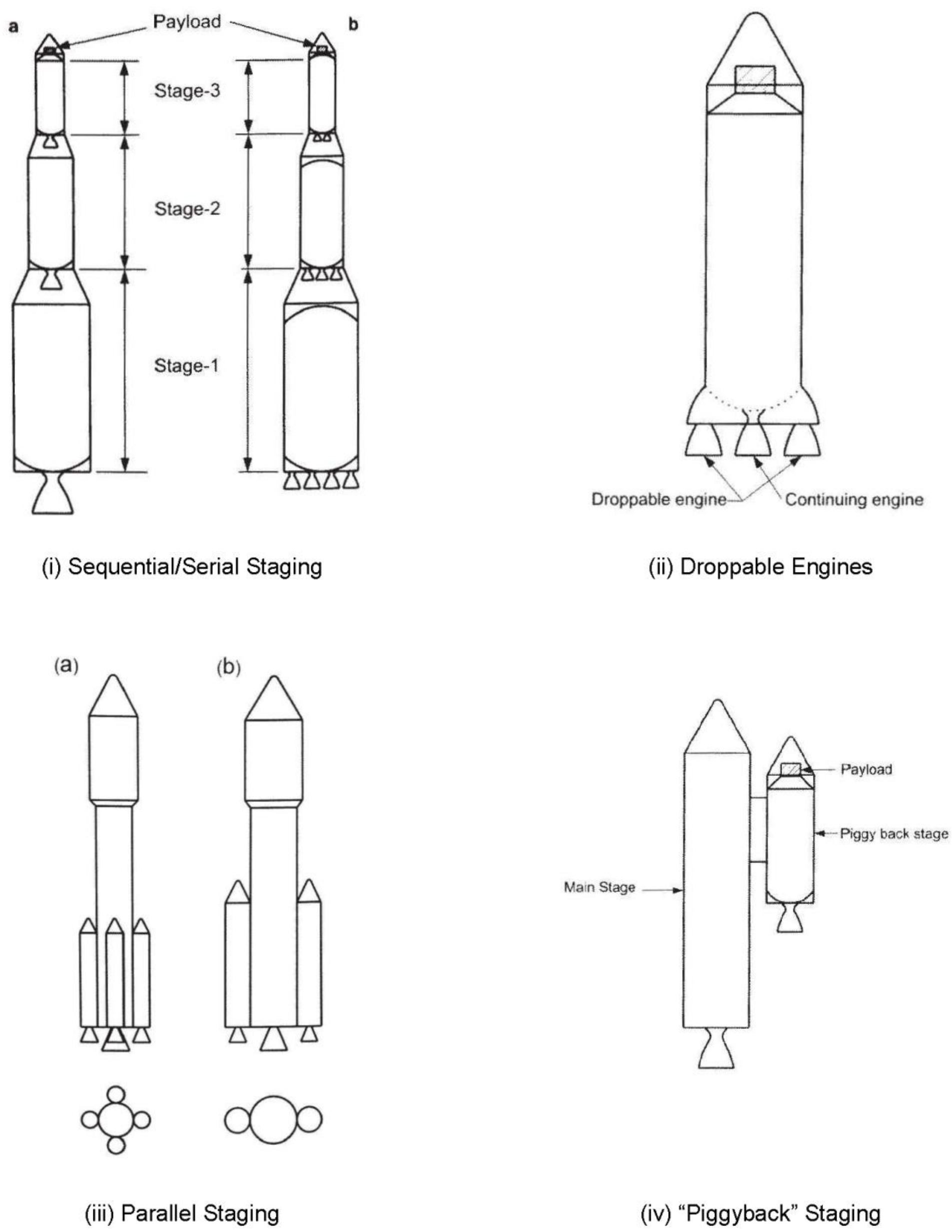

Figure 2.12. Methods of Staging. Adapted From [11]

The various methods of staging can be used in combination with each other. For example, the Space Shuttle used parallel and piggyback staging; the Pegasus XL uses serial and piggyback staging; and both the Delta IV Heavy and Ariane 5 use serial and parallel staging.

Determining exactly how many stages to use and the propellant distribution amongst these stages is critical to maximize the payload capacity of the launch vehicle. It can be a complicated process and is part of the designer's job during the conceptual design stage. However, as a general rule of thumb, space launch vehicles do not typically use more than three stages because the performance gain for each additional stage decreases due to the structure and equipment mass that must be added. Furthermore, each time a stage is added the vehicle becomes more complex and the number of failure points increases, and thus there must be an appreciable reduction in the launch mass in order to make the additional stage worth this additional complications and risk [83].

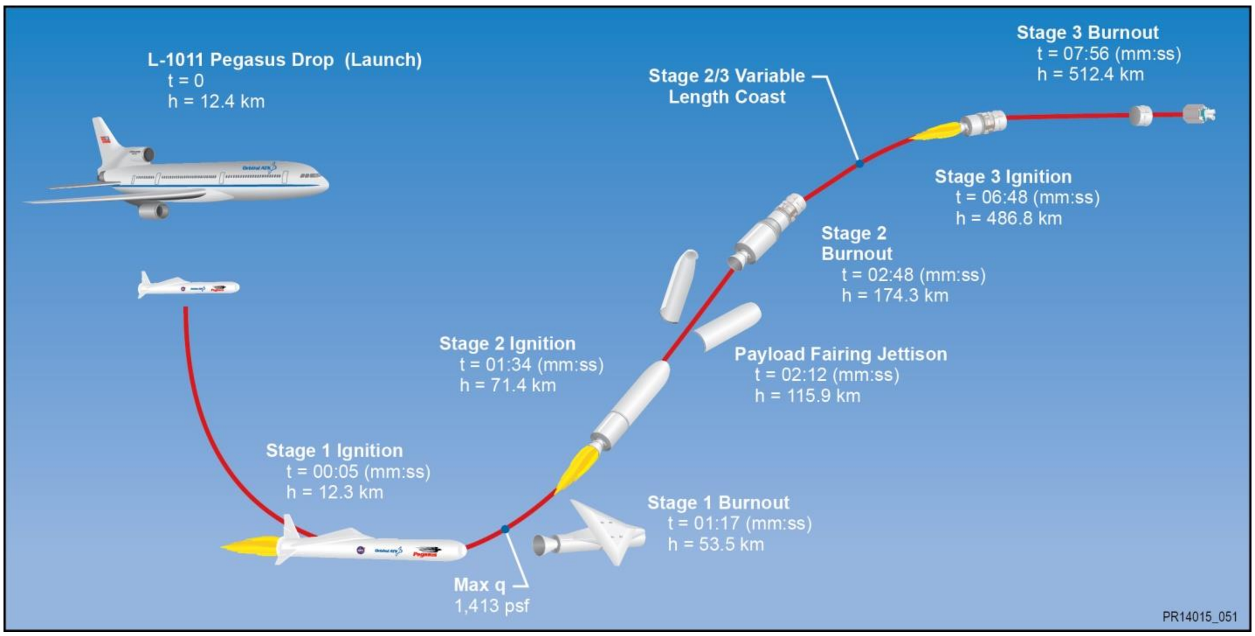

Figure 2.13. Pegasus XL Flight Path [80]

2.2.2 Launch Vehicle Mission Profile

The ascent to orbit trajectory for a two stage vehicle may be seen in Figure 2.14. The ascent process may be split into different phases, and the various literature on launch vehicle ascent trajectories have their own definitions and numbering convention for these phases [11], [84], [85], [86]. For this thesis, the ascent trajectory will be divided into 7 distinct phases as a working definition:

1. **Engine Startup**
   Just before the launch, the engines for the current stage begin their pre-ignition process.

As the countdown to launch approaches zero, the engines ignite and are powered up the full throttle.

2. **Initial Ascent**
   Once the launch window opens, the vehicle begins its ascent at a γ of approximately 90 degrees (straight up).

3. **Pitch-Over**
   Following the initial ascent, the vehicle performs a slight pitch over maneuver to reduce γ to below 90 degrees. As mentioned at the start of Section 2.2, the launch vehicle not only needs to increase its velocity but also is required to angle the velocity vector such that the orbit will maintain the desired shape. By pitching over gradually early into the flight, gravity will change γ into the proper shape to reach and maintain the desired orbit without expending additional energy on steering. This is called a gravity turn.

   Since launch vehicles leaving the Earth's atmosphere experience aerodynamic forces which introduce additional factors such as drag losses and stress on the vehicle from dynamic pressure, a gravity turn trajectory does not provide the most efficient ascent path. In order to optimize the ascent trajectory for minimum energy use, launch vehicles apply a combination of gravity turn and powered steering to reduce γ during the powered ascent phase. This process is explained in detail in reference [85].

4. **Powered Ascent**
   The launch vehicle continues its powered flight through the atmosphere until its propellant is expended. During ascent in the thicker portion of the atmosphere, one or more engines may be throttled down temporarily or shut down to avoid exceeding the maximum dynamic pressure the vehicle's structure can sustain. Similarly, as the vehicle nears burnout the acceleration experienced by the vehicle increases due to the vehicle's rapidly decreasing mass, and thus the engines may be throttled or shut down to avoid exceeding the maximum acceleration limit.

5. **Burnout and Coast**
   Once the propellant is expended, the engines shut down and the entire vehicle coasts. If the launch vehicle has additional stages, as is the case in the example provided in Figure 2.14, the next stage's engines begin their pre-ignition process just before the current stage's engines shutdown.

6. **Staging**
   If the launch vehicle is a multi-stage vehicle and the current stage is not the final stage, staging occurs. The next stage and current stage are separated and both parts of the

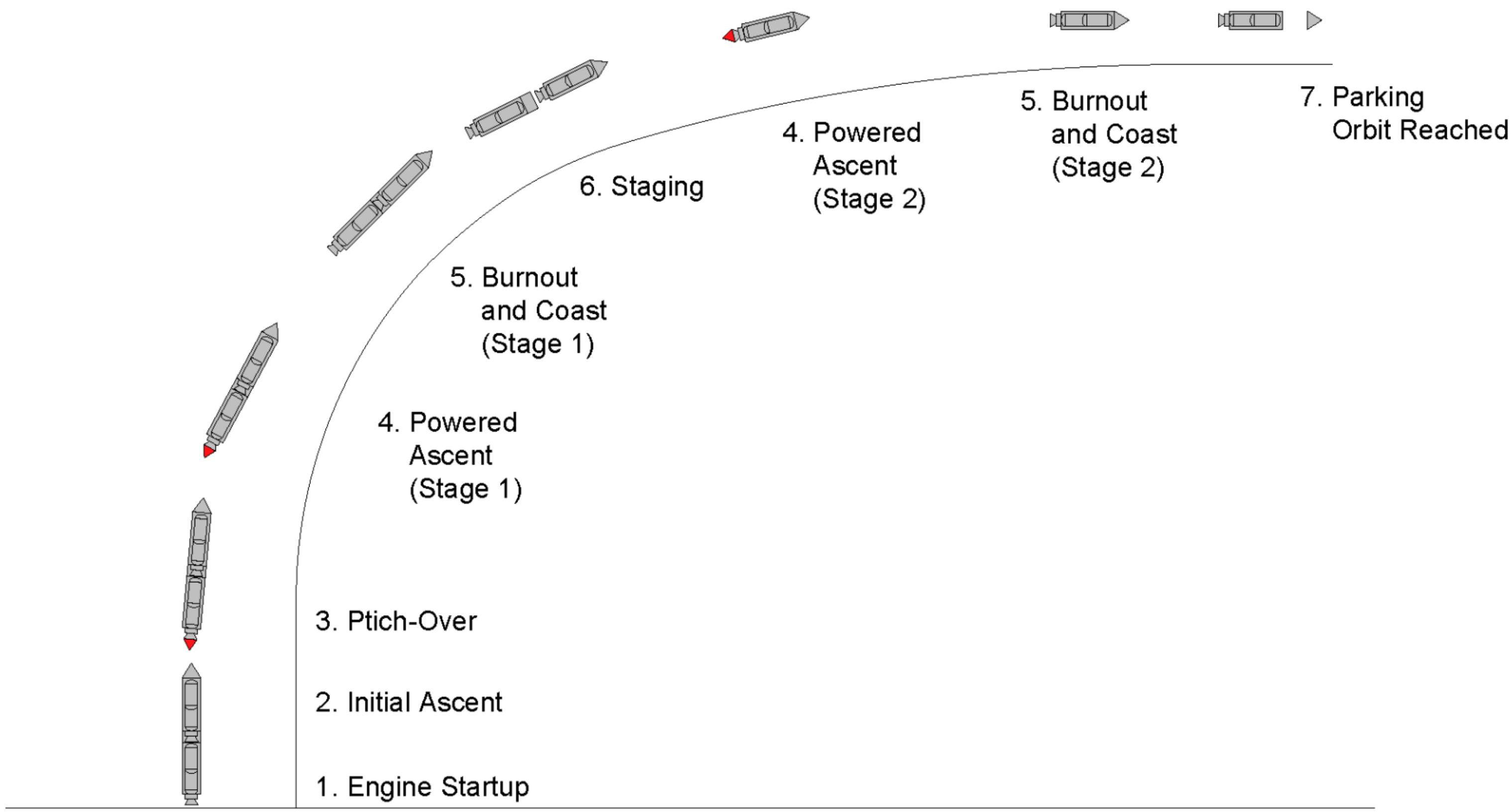

Figure 2.14. Ascent-to-Orbit Trajectory Phases

vehicle coast for another pre-determined period of time. The next stage's engine(s) then ignite and phases four through six repeat until the final stage for ascent is in use and the mission has been complete. For most launch vehicles, that will be once the desired parking orbit has been reached.

7. **Parking Orbit Reached**
The final stage's engines are shutdown once the parking orbit has been reached. In most scenarios this will be the end of the launch vehicle's mission, and the payload is thus ejected from the vehicle, but in some cases the parking orbit is not the final destination for the payload and the final stage will be used for additional maneuvers. Some examples of this include launch vehicles taking a payload beyond LEO and the Saturn V's final stage sending the Apollo spacecraft to lunar orbit.

The duration of the coasting in phases 5 and 6 depends on the vehicle as well as the mission's specific trajectory and can vary significantly. For example, the Gemini Launch Vehicle coasted for 0.7 seconds before staging and 0.2 seconds afterwards while the Falcon 9's user guide sample trajectory times state 5 seconds for the coasting prior to staging and an additional 5 seconds afterwards [87], [88].

2.2.3 Relating Design Disciplines to Launch Vehicles

When the literature review that will be discussed in Section 2.3 was initially started, it became apparent that a method was needed in order to distinguish various references from one another in terms of their usefulness to this research project. Since the goal of this research undertaking has been to develop a new launch vehicle design software, the researcher and author determined the best way to compare the literature was to notate what parts of a launch vehicle's design they cover. In order to do that, a more in depth look at launch vehicles is required.

As introduced in Section 2.2, a launch vehicle's mass is divided into three groups: the payload mass, the propellant mass, and the structure and equipment mass. When the interior of a launch vehicle is examined, the items that makeup the structure and equipment mass can be determined. Cutaways of three such launch vehicles are presented in Figure 2.15 through Figure 2.17. From analyzing these it is determined that launch vehicles consist in general of the following components:

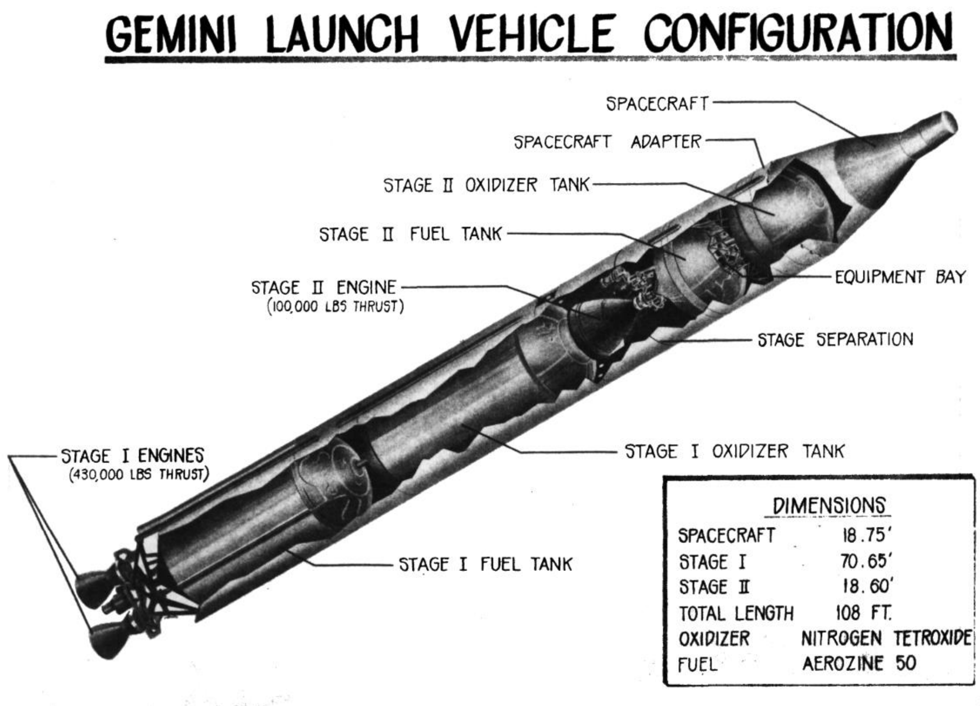

Figure 2.15. Gemini Launch Vehicle Cutaway [89]

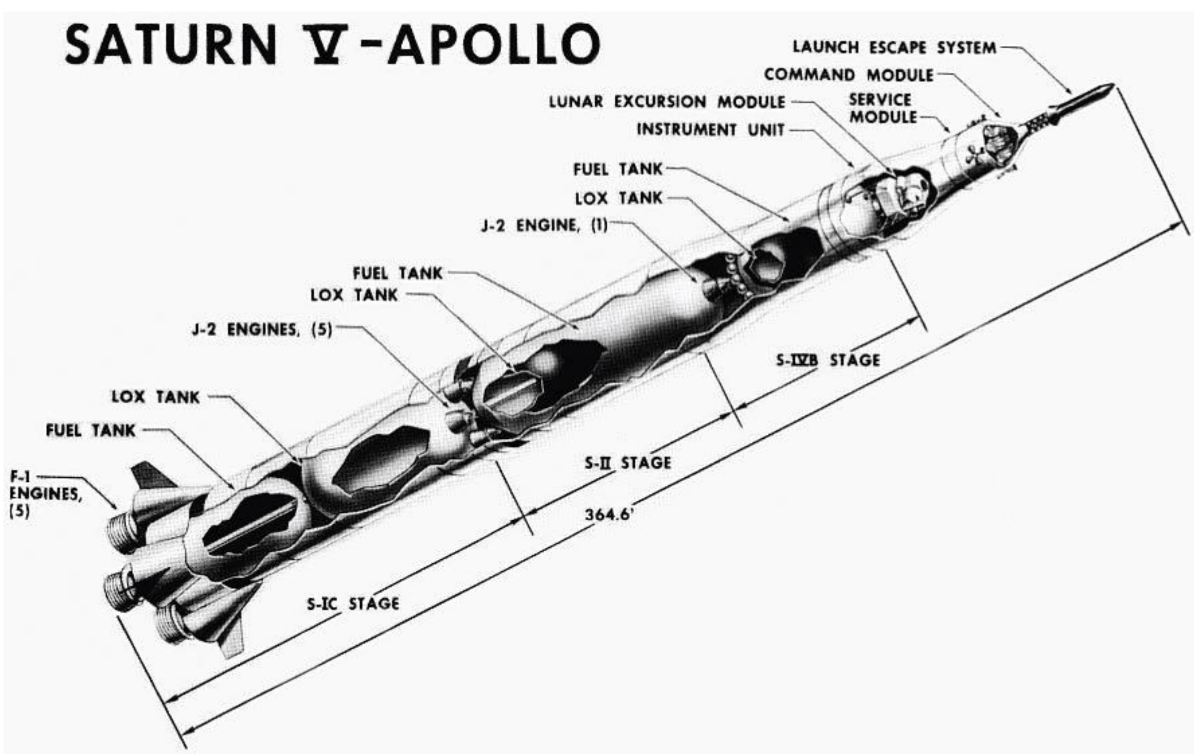

Figure 2.16. Saturn V Cutaway [90]

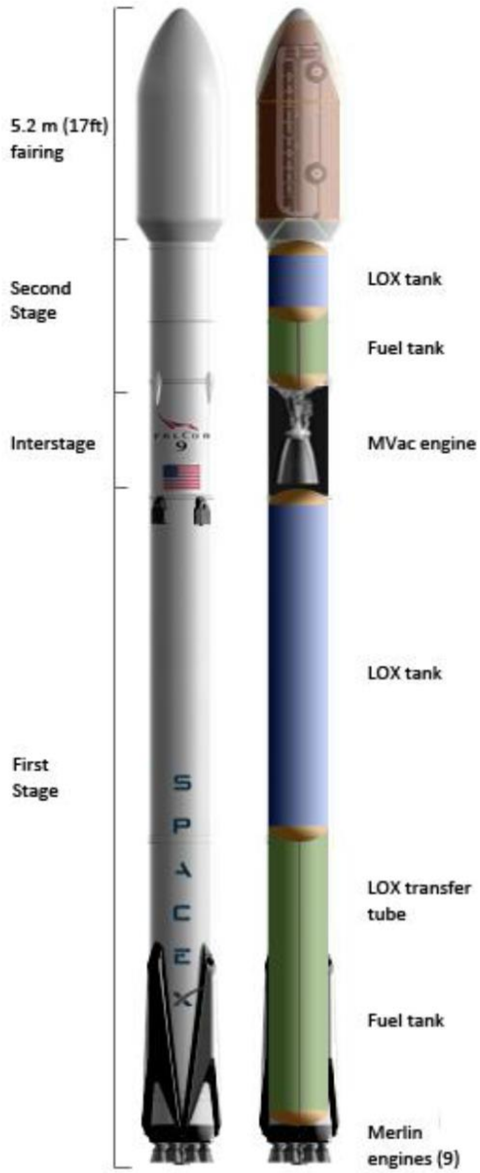

Figure 2.17. Falcon 9 Cutaway [88]

- **Payload fairing**
  During the ascent through the dense atmosphere the launch vehicle is subjected to periods of high stress from aerodynamic forces. While the launch vehicle is designed to withstand these forces, most payloads are designed for operation outside the atmosphere. As such, a payload fairing is required to encapsulate the payload to protect it during ascent.
  Payload fairings are not used when the payload is a capsule as the capsule is designed to experience atmospheric loads.

- **Avionics and other electronics**
  The avionics and other electronics make up the flight computer, various sensors on the vehicle for self-diagnostic and telemetry, power source, wiring, etc.

- **Propellant tanks**
  Each stage of the launch vehicle has its own fuel tank and oxidizer tank. These tanks are cylinders with ellipsoidal end caps. The propellant tanks are separate entities for the Gemini Launch Vehicle and first stage of the Saturn V, but are a single unit separated by a common bulkhead with the remaining Saturn V stages and both stages of the Falcon 9. During the design of the Saturn V it was found that significant weight savings could be achieved by utilizing this common bulkhead design rather than having two completely separate tanks, although this savings is partially or fully negated by the need for additional insulation at the common bulkhead if the temperatures of the fuel and oxidizer are significantly different [75].
  Typically the tank with the heavier mass is on top, as is seen for the Gemini Launch Vehicle, Saturn V first stage, and Falcon 9. This is done in order to pull the CG forward to increase aerodynamic stability; however, it does create a weight penalty due to the additional structure required to keep the heavier tank from crushing the lighter one [91], [92].

- **Primary launch vehicle structure**
  The primary structure for each launch vehicle made up of the propellant tanks, forward and/or aft skirts, and intertank fairings. In the case of the reusable Falcon 9 the vehicle structure also includes landing legs.

- **Interstage fairing**
  While the engines do not require any additional structure surrounding the nozzles while they are in use, an interstage fairing bridges the gap between the stage structures and houses the engine nozzles of inactive stages. When staging occurs, the interstage fairing is also discarded with the prior stage. (I.e., for a two stage rocket, when staging occurs the interstage fairing falls away as part of stage one.)

- **Engines and thrust structure**
  The rear of each stage has the engines, and the engines are held together in their configuration as well as to the vehicle with a thrust structure.

These components are tied back to the primary launch vehicle systems introduced in Section 1.1.1 and shown in Figure 2.18.

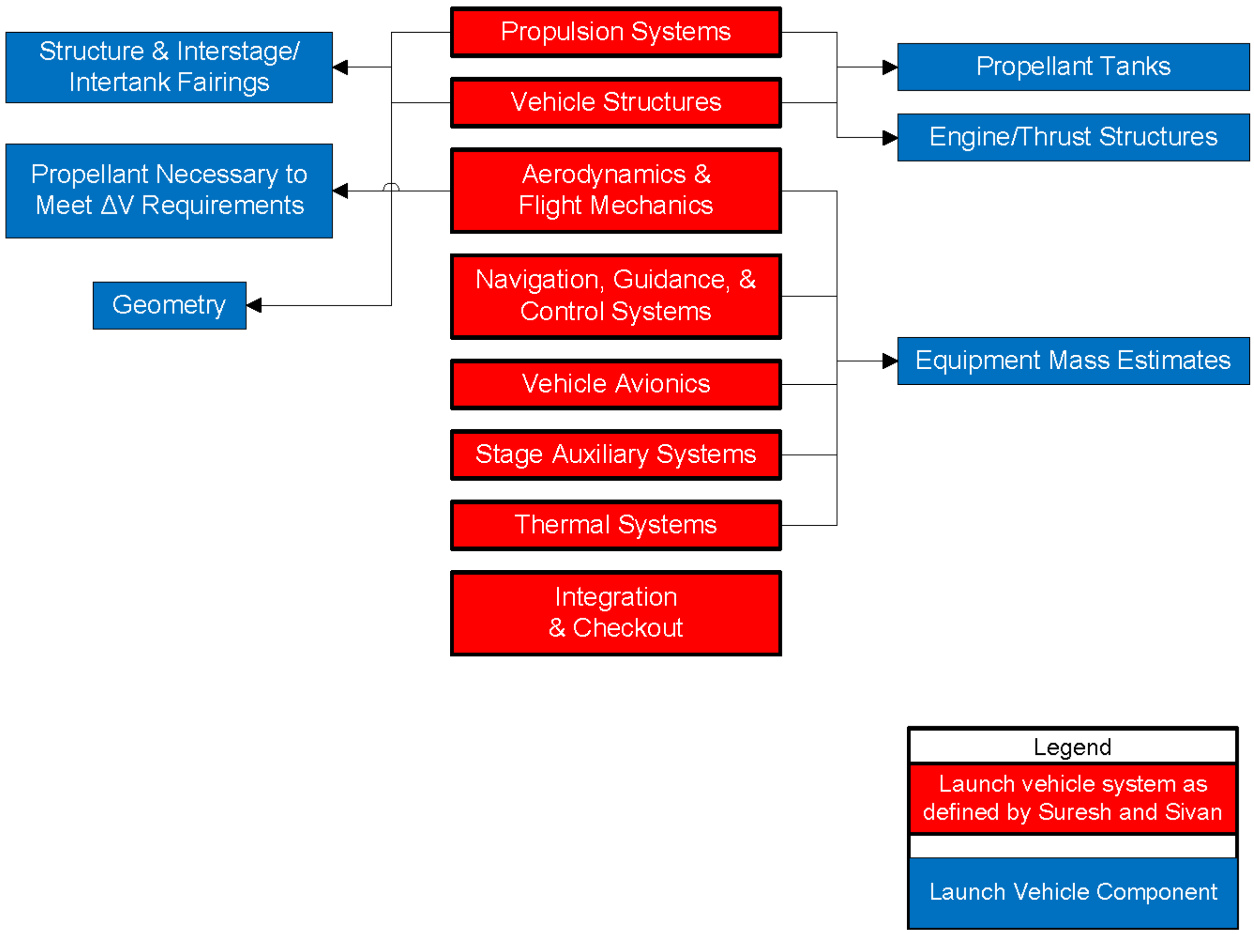

Figure 2.18. Relating Launch Vehicle Components and Design Disciplines

After restarting the literature review, it was realized that this is still insufficient for adequately comparing launch vehicle resources. With the knowledge gained from the literature review up to this point and a more in-depth view into launch vehicles and their mission profile, a series of "elements of design" were derived which better relate the various software and literature to the design of launch vehicles. These elements are:

- $\boldsymbol{\Delta V_{ideal}}$, the ideal velocity required by the payload to maintain orbit.

- $\boldsymbol{\Delta V_{losses}}$, a method to estimate the velocity losses incurred during the ascent to the desired orbit. The ΔV_{losses} must be added to ΔV_{ideal} to determine the total ΔV requirements of the launch vehicle.

- **Launch Vehicle Mass Sizing**, the initial mass estimates for the primary components of a launch vehicle.

- **Propulsion Sizing**, the sizing of the propulsion systems. This includes not only the performance characteristics of the engine (thrust, specific impulse, etc.) but its mass and geometry as well.

- **Component Mass Sizing**, a secondary calculation of the structures and equipment of a launch vehicle in order to confirm this aspect of the initial mass estimate. Specifics that should be included in this secondary calculation are the masses of the propellant tanks, tank insulation, thrust structure, avionics and other electronics, etc.

- **Structural Analysis**, the analysis to ensure certain parts of the vehicle can withstand the expected loads. For example, propellant tank loads directly determine how thick the tanks must be, and this information can be estimated early on based on the propellant densities and volume.

- **Launch and Orbital Mechanics** and **C_D Calculation**, the physics of getting a launch vehicle into orbit that takes into account not only gravity but aerodynamic drag.

- **Boostback Sizing**, **Fly/Glideback Sizing**, and **Parachutes Sizing**, the sizing of partial or full reusability of a launch stage(s).

- **Geometric Sizing**, the physical geometry of the launch vehicle.

- **Cost**, the cost to develop, produce, and then fly a launch vehicle.

Additionally, **Optimization Methods** can be used hone in on an "optimal" design through the use of an objective function and mathematical techniques to determine the sensitivity of the vehicle’s design to specific inputs and modify the inputs accordingly to reach the designed design. The relationship between all of these elements and the design disciplines is seen in Figure 2.19.

It is important to note that "Integration and Checkout" has been removed from the list of launch vehicle systems. While integration and checkout is a critical part of getting a launch vehicle from individual components to a completed vehicle ready to fly, it is not a design discipline but part of operations.

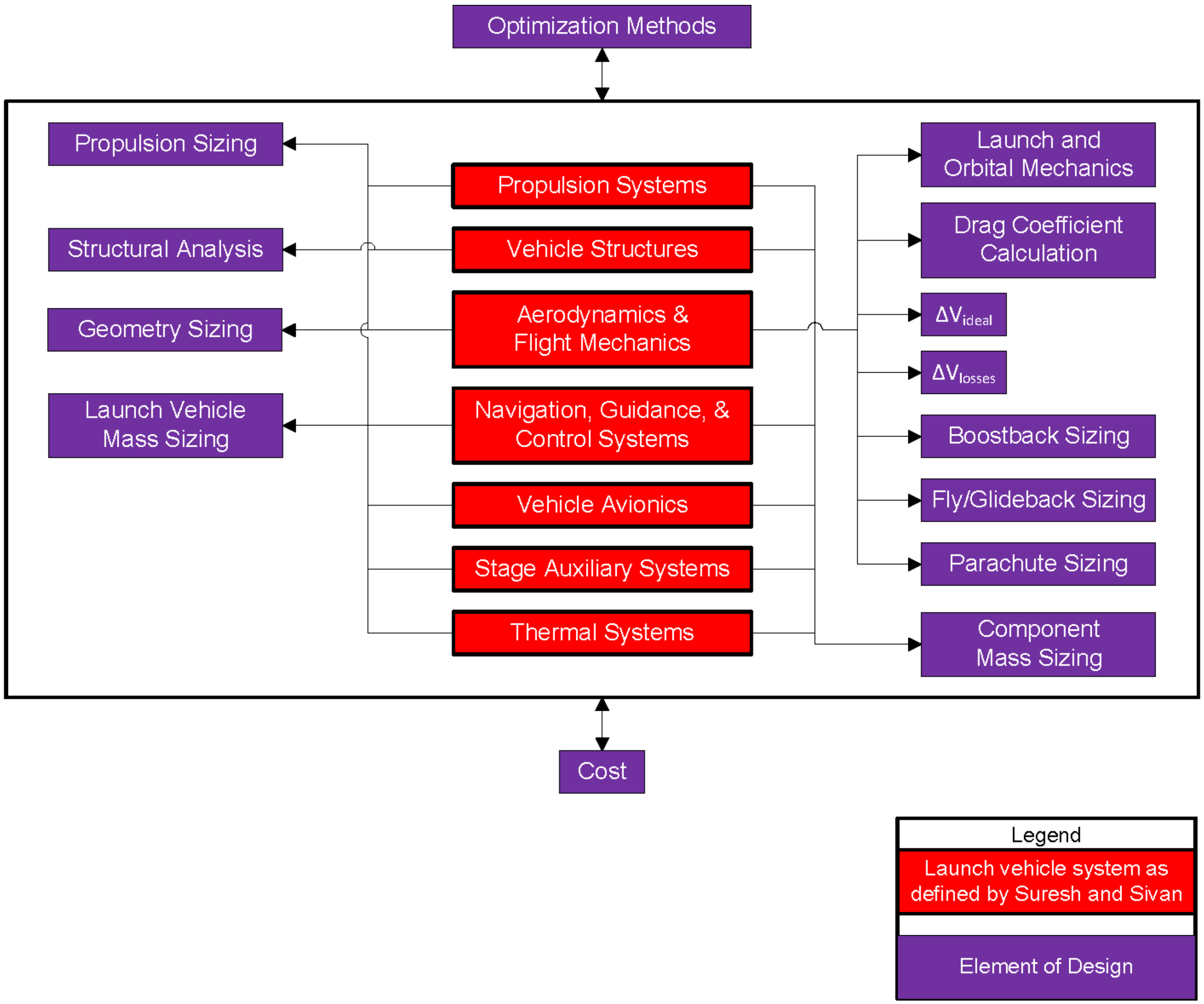

Figure 2.19. Elements of Design

By denoting which of the fourteen elements of design a particular reference contains, the reference's usefulness can be indicated. Furthermore, it allows the scope of the references to be compared and ranked against each other.

2.3 Resources

Existing software, books, technical reports, Master's theses, and PhD dissertations were reviewed in order to determine the capabilities and features of launch vehicle sizing programs, find the necessary equations to develop a new system, and compare the proposed new system's capabilities to what has been done in the past.

Three spreadsheets were generated to capture and compare information on these items. The first contains existing software; the second the various books and technical reports; and the third contains theses and dissertations. In order to appropriately compare resources to each other, each spreadsheet contained columns for identifying if the resource contained one of the elements of design derived in Section 2.2.3. Additionally, the method used by the reference was recorded. The method could be classified as:

- **Analytical**, where the fundamental equations for sizing based on physics are provided;
- **Empirical**, where existing data from a variety of sources has been compiled and equations are derived from the data to size new vehicles;
- **Graphical**, where plots have been generated based off either the physics behind launch vehicles or known data from existing vehicles, but the equations or data used to create the plots is not provided; or
- **a combination of the above**

2.3.1 Existing Software

Data on sixteen existing launch vehicle sizing software was found in the public domain. The master spreadsheet can be found in Appendix C. and a summary of the results is shown in Table 2.3. Each element of sizing was marked with a "Yes" if the program considered that element when sizing a launch vehicle, a "No" if it did not. Due to the proprietary and/or ITAR restricted nature of some of these tools, it was difficult to obtain a complete information on the program or, in some cases, any information at all. Such elements are marked with "Unknown" to indicate that this information was not found during the course of the literature search.

From analyzing the results of this literature survey, it can be seen that none of the software provided a singular tool that included all of the elements of design as specified earlier. All of the tools on which information could be found contain the basic elements required to size a launch vehicle (namely, a calculation of ΔV_{ideal}, ΔV_{losses}, and the initial mass estimates. They also included a check on the structures and equipment mass after the vehicle is sized. Past this the software features vary. For example, some require an outside tool to run a trajectory simulation while others have the feature built-in; others can provide preliminary sizing for a propulsion system specific to the launch vehicle while others require engine data to be provided.

A brief summary of FONSIZE and INTROS are provided below.

FONSIZE

FONSIZE was developed in-house at the Aerospace Corporation by H. G. Nguyen in the early 1990's. The goal of FONSIZE is to combine the launch vehicle sizing with a trajectory simulation in order to automate the process of determining the most efficient trajectory to reach the orbit, using the actual ΔV_{losses} from this trajectory for the vehicle's energy requirements, and trading user-specified variables such as the different types of available propulsion systems in order to minimize the space launch vehicle's mass. It is written in FORTRAN, and the various portions of the sizing process are broken into modules so specific models could be added or replaced as desired. It is capable of sizing not only classical expendable launch vehicles but also winged vehicles for horizontal takeoff or as use for later stages on a vertical-takeoff vehicle [93], [94].

Little documentation is available on FONSIZE outside of two studies published by FONSIZE's creator, Hal Nguyen. In these studies, Nguyen details FONSIZE's ability to simultaneously size a launch vehicle and optimize its trajectory as well as discusses the logic used by the optimization methods. [93], [94] A Nassi-Shneiderman diagram, a visualization of the programming logic used by a process or program, is developed for FONSIZE from these studies and is presented in Figure 2.20. As proprietary software FONSIZE is not available publicly.

Table 2.3 Summary Results of Software Literature Search

Acronym	Developer	Commercial/ Academic	Elements of Sizing		
			# of Y's	# of N's	# of Unknown
FONSIZE	The Aerospace Corporation	Commercial	7	1	6
GTS - Size	The Aerospace Corporation	Commercial	0	0	14
AVID	NASA Langley	Commercial	0	0	14
HAVCD	Boeing	Commercial	1	0	13
FLYIT	Boeing	Commercial	0	0	14
POST	Martin	Commercial	1	0	13
FASTPASS	General Dynamics	Commercial	0	0	14
PREVAIL	The Aerospace Corporation	Commercial	0	1	13
BP	The Aerospace Corporation	Commercial	0	0	14
CONSIZ	NASA Langley	Commercial	3	3	8
ASTOS	Astos Solutions	Commercial	7	0	7
INTROS	NASA	Commercial	8	6	0
N/A	SpaceWorks; AFRL; Wright-Patterson	Commercial	0	0	14
HySIDE SpaceSIDE	Astrox Corporation	Commercial	6	0	8
STAGEX ROKOPT	University of New South Wales at Australian Defense Force Academy	Academic	3	0	10
SIZE	DeBlizan and Pickett (The Aerospace Corporation)	Commercial	1	0	13

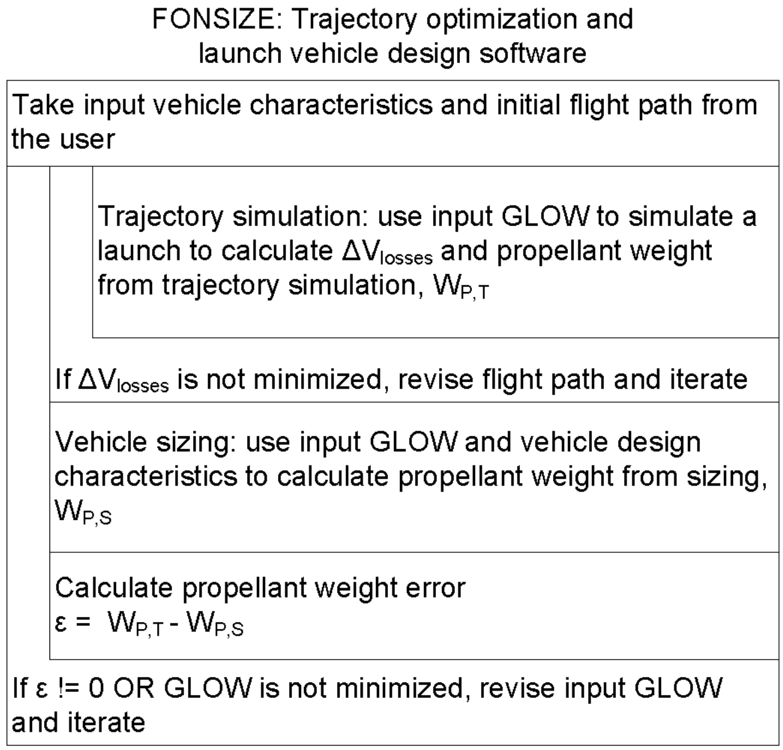

Figure 2.20 NS Diagram for FONSIZE

INTROS

INTROS stands for "INTegrated ROcket Sizing Model" and was developed by Emory Lynn at NASA Marshall Space Flight Center. It is written in Visual Basic and uses Excel for the front end to receive inputs and display outputs. The tool provides a large wealth of different design options in nearly every aspect that would be desired. The propulsion system can be set for either a solid, liquid, hybrid, or rocket-based combined cycle engine, and a variety of different fuels and oxidizers can be used by selecting a combination of fuels and oxidizers from a list with preset densities. If the user desires a different density of their selected propellants or selects "other" to use propellants not include in the list, the user can manually enter densities to use. The geometric shapes to be used for the overall stage body, propellant tanks, etc., can be defined. Wings, canards, and tails can be added to the design for either takeoff and/or landing purposes [95], [96]. A NS diagram of INTROS based off of the user manual is provided in Figure 2.21.

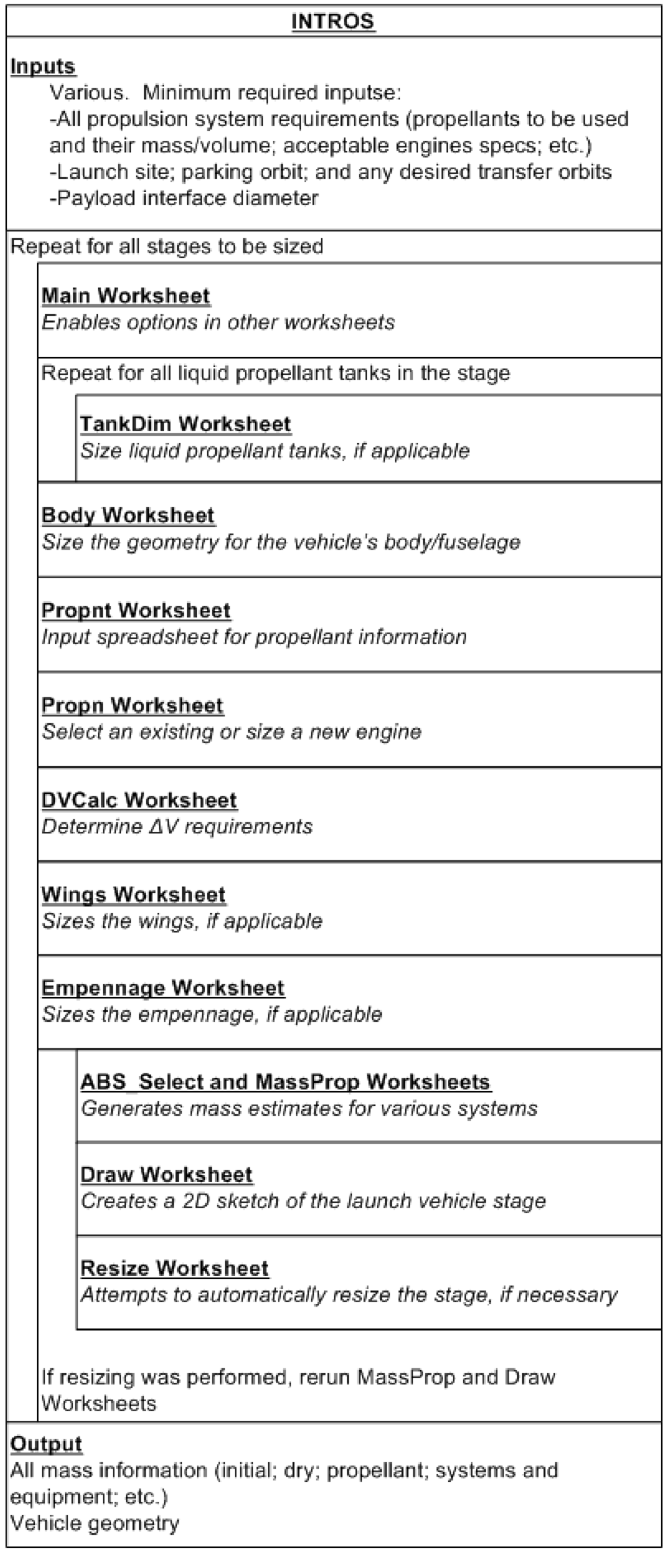

Figure 2.21. INTROS NS Diagram

Mass estimating relationship (MER) equations are a critical component of INTROS. MERs "calculate the masses of the various stage systems, subsystems, propellants, fluids and other consumable items" [95]. These complex equations are created based on empirical data and are part of the reason INTROS is regulated under The International Traffic in Arms Regulations (ITAR) [97]. Because of this regulation INTROS is not available for use outside of government approved contracts and cannot be retained after the project is complete. As this research project is not tied to a government contract INTROS is unavailable for comparison purposes.

INTROS's features and implementation result in a tool which can size a broad range of space launch vehicles but requires significant work on the part of the user. Each stage of the launch vehicle must be sized manually in a separate INTROS file, and then combined together. INTROS does not contain a built in trajectory simulation program to verify the launch vehicle can actually complete its mission, and thus the designer must port the launch vehicle design to a secondary program, such as NASA's POST (Program to Optimize Simulated Trajectories), to simulate a launch and verify whether or not the mission is successful. If the launch vehicle cannot complete the mission, the user must return to the INTROS files and resize the stages as necessary in order to improve performance enough to complete the objective. The Resize Worksheet inside INTROS can be used to decrease the time this step takes, but it still requires the user to manually go back into each INTROS file, resize each stage as they believe is required, assemble the complete vehicle, and rerun the trajectory simulation program. This cycle is manually repeated until a feasible solution is found.

Plans to overhaul INTROS are in progress. Dr. Adam Irvine from the Advanced Concepts Office at NASA MSFC is currently in charge of this project. Dr. Irvine plans to create a standalone program in C++ and later use the software Qt to create a GUI. Ideally, the program will be setup in such a way that information for the subsystems can be calculated either using MERs, an analytic method, or with another program. As the MERs will be decoupled from INTROS into their own subsystem it is hoped that INTROS will be available publicly after the update. [97]

Software Review Results

Nearly all launch vehicle design software is proprietary. The remaining programs have restrictions placed on who can receive the software and what it can be used for. In the case of INTROS, only US citizens with a need to use the software for a government project can use it, and per the user agreement INTROS must be deleted once the project is complete. All of the software provide the basic features required to size an expendable and, with FONSIZE and INTROS, winged reusable launch vehicles. Features past that, such as propulsion sizing, structural and aerodynamic analysis, and a trajectory simulation, vary from tool to tool. Based on the data available on the workings of the software programs, it is very difficult and time consuming to add new features so different types of launch vehicles can be sized.

In order for the software produced by this thesis to be useful compared to these existing programs, it must be readily accessible and include most of the elements of design. The software must also be setup such that any features it does not include can be easily integrated in at a later date without making the system cumbersome to use.

2.3.2 Literature

For this part of the literature review, various books, technical papers, and lectures have been review. When reviewing each reference for what element of sizing it contains, each element of sizing is marked with

- **"Yes"**, if it included the equations necessary to perform that aspect of sizing;
- **"Discussion Only"**, if the item was discussed but equations were not provided; or
- **"No"**, if the element wasn't mentioned at all.

Additionally, the element **Propulsion Sizing** can be marked with "Performance Only" to indicate equations to calculate an engines performance are included but no information is provided on how to determine the engine's mass or geometry.

A total of sixty-three sources including books, technical papers, and lectures were reviewed. All sixty-three references maybe be found in Table C.1 in Appendix C. As is to be expected with their length restriction, most of the technical papers only cover one or two design elements. Books and lectures are far broader in their scope, but the average number of design elements covered is below four. Less than ten of the references included propulsion sizing, structural analysis, a C_D calculation, and a cost analysis. Only one resource included boostback reusability sizing but it did not provide any of the equations necessary to repeat the analysis.

Table 2.4. Summary Results of Book and Technical Paper Literature Search

Source Full Name	Author/ Developer/ Lead Editors	Book/Paper/ Thesis/Other	Year Published	Method (Analytic, Graphical, Empirical)	# of Y's	# of N's	# of Discussion\ Performance
Design Methodologies for Space Transportation Systems [98]	Walter E. Hammond	Book	2001	Analytic	6	4	4
Aerospace Vehicle Design, Volume II: Spacecraft Design [83]	K. D. Wood	Book	1964	Both Analytic and Graphical	7	4	3
Space Planner's Guide [99]	U.S. Air Force (Harney)	Book	1965	Graphical	7	7	0
Handbook of Astronautical Engineering [69]	Heinz Hermann Koelle, et. al.	Book	1961	Both Analytic and Graphical	6	7	1
University of Maryland Lecture Series [100]	David Akin	Lecture	2016	Analytic	7	5	2

After further reviewing of Table C.1, five references are selected as the most comprehensive literature currently available for launch vehicle design: *Design Methodologies for Space Transportation Systems* by Hammond [98], *Aerospace Vehicle Design, Volume II: Spacecraft Design* by Wood [83], *Space Planner's Guide* by Harney [99], the *Handbook of Astronautical Engineering* by Koelle, et. al. [69], and a lecture series from the University of Maryland by Akin [100]. Table 2.4 includes these references and the number of elements they include, and a review of *Aerospace Vehicle Design, Volume II: Spacecraft Design* and the *Space Planner's Guide* are found below.

Aerospace Vehicle Design, Volume II: Spacecraft Design by K. D. Wood

Aerospace Vehicle Design, Volume II: Spacecraft Design was written by K. D. Wood [83] as the second book in a series on the design of various types of aerospace vehicles. The book provides a comprehensive overview of the design of launch vehicles and includes all but three of the elements of design with particular emphasis on the selection of a propulsion system, sizing of the three groups of launch vehicle masses introduced in Section 2.2, and orbital mechanics.

Wood uses a combination of equations and plots for launch vehicle sizing. The plots are used as guides for selecting input values. One such plot is presented in Figure 2.22. This particular figure presents π_{se}, the fraction of total vehicle weight belonging to the structures and equipment, as a function of the payload weight. It includes limits represented by the energy density of available propellants. By starting with a known payload weight from the mission requirements and a propellant energy density from what is available for use in the design, the designer selects an initial value for π_{se} to begin the sizing process with.

Wood takes the reader step-by-step through the sizing process, explaining each calculation as it occurs and any necessary plots required in making assumptions. At the end of the example, the reader will have successfully sized the three mass groups for a two stage launch vehicle, determined the masses of each stage’s fuel and oxidizer, and generated a simplified geometry based that includes the vehicle’s overall and stage length and diameter. A summary of Wood's design process may be found in the NS diagram provided in Figure 2.23 and is utilized in this research project as the core of the vehicle sizing tool.

Space Planner's Guide by E. D. Harney

The *Space Planner's Guide* [99] was created by the Air Force in the mid 1960's and contains a treasure trove of information regarding the design of launch vehicles. The *Guide* was originally restricted to internal use only, but in recent years has made its way online to various auctions sites and has become a valuable technical resource and collectible.

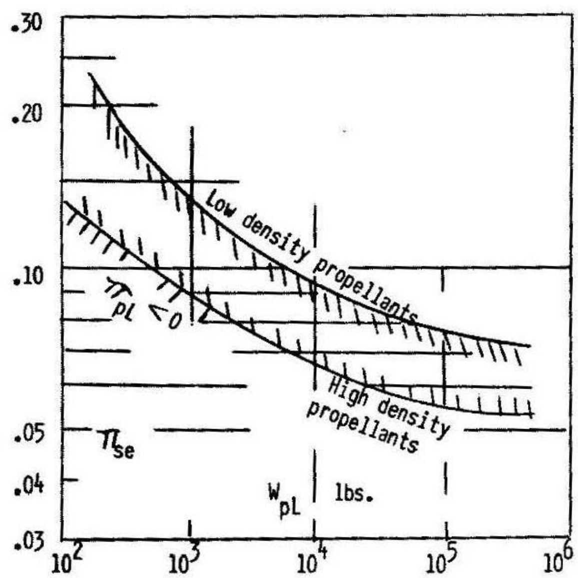

Figure 2.22 Guide to Selecting π_{se} Based on Propellant Density and Payload Weight [83]

Rather than directly using equations which require repetitive hand calculations or the use of computers, which at the time were difficult to get time on, the *Guide* sizes launch vehicles using nomographs. Nomographs are plots of three or more variables which are set up such that the value of one variable can easily be found by drawing straight lines to intersect one of the lines on a plot. An example of a nomograph may be seen in Figure 2.24. This nomograph relates the mass ratio for a stage, the stage's specific impulse, and the ΔV that the stage can produce, $\Delta V_{capability}$. The nomograph works like this: start with a mass ratio for the stage, such as the 3.35 used in the figure. Draw a horizontal line from the value for that mass ratio to the line corresponding to what the stage's specific impulse is. At the intersection of these two straight lines, draw a vertical line down to the x-axis. This value is the $\Delta V_{capability}$ that the stage can produce. The nomograph can be used to find either the mass ratio, the

Aerospace Vehicle Design, Volume II: Spacecraft Design by K. D. Wood
Launch Vehicle Sizing

Required Inputs:
Mpayload; payload geometry; desired orbit; launch location

Determine number of stages:
Using Figure 2-6:4 and design preference

Assume combustion chamber pressure based on desired engine properties

Determine I_{sp} values for each stage and vehicle mean

Estimate ΔV_{ideal}, ΔV_{loss}, and ΔV_{req}, and ΔV split between stages

Calculate effective exhaust velocity and divide it by ΔV_{total} to check ΔV distribution

Calculate the mass ratio and final fractional mass of each stage

Starting at the last stage and working backwards to the first

- Calculate the stage's systems & equipment and payload fractional masses
- Calculate the stage's initial weight

Calculate propellant weight ratio and weight of each stage

Add in estimated ullage of each stage

Select engines and calculate T/W, terminal acceleration, and estimated powerplant weight of each stage

Calculate structure & ullage weight of each stage

Verify systems and equipment fractional weights estimate of each stage

Determine fuel fraction of each stage

Calculate fuel and oxidizer weight, volume, and tank geometry of each stage

Figure 2.23. NS Diagram of the Sizing Process from *Aerospace Vehicle Design, Volume II: Spacecraft Design*

required specific impulse, or the stage's $\Delta V_{capability}$ produced; the user either starts with the values they know or are desired and then work towards the unknown, or iterates over a large number of parameters until the parameters converge on a valid vehicle design.

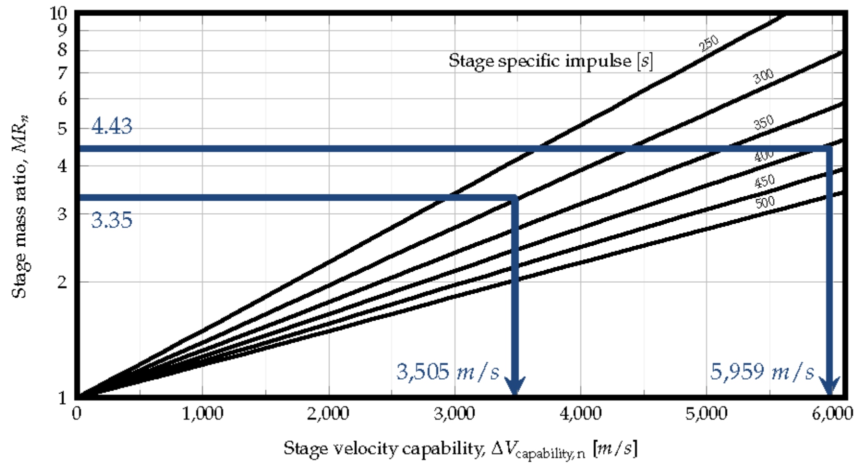

Figure 2.24. Example nomograph recreated from the *Guide* [13]

Coley summarizes the *Guide's* sizing process in the NS diagram seen in Figure 2.25. All of the figures referenced in the NS diagram may be found in reference [13].

While the nomographs provide a quick, simplistic method to sizing launch vehicles by hand, their use introduces a number of problems. First, certain assumptions were made in the generation of some of the nomographs, and in some cases it is not clearly indicated that such assumptions were made. As such there are limitations on the performance of the launch vehicles that can be generated from it. As Coley noted on his use of the nomographs to size launch vehicles as part of the Apollo case study for his dissertation, the nomographs optimize stages for a 185 km orbit and assume no partial burns are used to reach the parking orbit. Because the Saturn V used a partial burn on its third stage to get to its parking orbit, the *Guide* cannot be used to size the Saturn V. (However, note that vehicles similar to the Saturn V that can complete the same mission can be sized using the guide. See Section 4.2.2 of Coley's thesis for additional information [13]).

Second, a nomograph of the specific relationship must exist in order to size that portion of the launch vehicle. In other words, if one wishes to size a launch vehicle to includes features, such as boostback recovery, one must first generate an appropriate

THE SPACE PLANNERS GUIDE
[LAUNCHER SIZING]

Input: number of stages, $m_{payload}$, $m_{escape\ tower}$, ΔV_{req}, propulsion properties of each stage (I_{sp}, propellant density)

Estimate total launch vehicle mass to payload mass ratio – Figure V. C-9: $(1/\lambda_{total})$

Estimate maximum length and diameter – Figure V. C-10: l, d

Estimate stage mass to mass above stage ratio – Figure V. C-11: $(1/\lambda_n)$

First estimate of mass ratio for each stage – Figure V. C-11: MR_n

Calculate first estimate of stage mass and total mass of each stage: $m_{total,n}$, m_n

Estimate stage structure factor –

- First stage
  - Figure V. C-15: ε_1
- Second stage (if applicable)
  - Figure V. C-16: ε_2
- Third stage (if applicable)
  - Figure V. C-17: ε_3

- **Input:** MR_n
- **Estimate mass ratio for next stage** – Figure V. C-22: MR_{n+1}
- **Estimate velocity capability of each stage** – Figure V. C-23: $\Delta V_{capability,n}$
- **Calculate total velocity capability:** $\Delta V_{capability}$

Until $\Delta V_{capability} \approx \Delta V_{req}$

- **Re-estimate stage mass to mass above stage ratio** – Figure V. C-24: $(1/\lambda_n)$
- **Calculate stage masses of launch vehicle:** $m_{total,2}$, m_2, $m_{total,1}$, m_1
- **Re-estimate structure factor for each stage** – Figure V. C-15, C-16, and C-17: $\varepsilon_{n,new}$

Until $\varepsilon_{n,old} \approx \varepsilon_{n,new}$

Calculate final masses of launch vehicle: $m_{total,2}$, m_2, $m_{total,1}$, m_1

Figure 2.25. NS for the Space Planner's Guide [13]

nomograph and insert it into the sizing process at the appropriate point. However, either empirical data or equations must exist in order to generate such a nomograph.

Additionally, computers and the computation power they provide are no longer at a rare, precious commodity. As such sizing launch vehicle through the use of nomographs with previously made assumptions that cannot be changed is no longer necessary.

Literature Review Results

No single piece of literature exists in the public domain which covers all of the various elements of launch vehicle design. Out of the literature listed in Table 2.4, the Wood text is selected to be the primary reference that will be used to create the new launch vehicle design program. Design features which are not covered by the book are sourced from other references.

2.3.3 Other Master and PhD Theses

Other Master and PhD theses were reviewed to get an idea of what other researchers have done in the past. The table used to compare these includes all of the same columns for design elements that are used in the table for sizing books and technical documents. After reviewing several theses, it became apparent that most elements would be marked with an "N" or "Discussion Only". This isn't because the thesis's author didn't do work in this area, but because either the equations required are not provided or existing software, such as NASA's POST or software developed at their university by prior students, is included in their work. As such, if the thesis document included at a minimum a discussion on the element it received a "Y"; if it did not address the topic at all, it received an "N", even if it was clear from the output that the author had to address that element at some point in their work. Additionally, another column was added to indicate whether or not the thesis author included a literature search in their work. Three values could be assigned to this:

- **N**, meaning no literature search information was provided. That doesn't necessarily mean the author did not perform one, but simply that a review of various sizing references is not documented.

- **Basic**, meaning at least one sizing reference was mentioned as being a major contributor to their thesis work.

- **Detailed**, meaning a large number of sizing references were not only mentioned but some sort of comparison was done between them.

When analyzing this table, it is important to remember that the sizing of launch vehicles may not have been the primary focus of that person's thesis. For example, at the start of his thesis Ritter [101] states that the thesis objective is to optimize a heavy lift launch vehicle, and he does so by focusing on the optimization of a few specific elements. Bayley's thesis [102] was on the use of genetic algorithms to find the optimal vehicle design and as such he explores the history behind genetic algorithms. However, a history on launch vehicles is non-existent. Miranda [103] scores an "N" on most elements, but was using existing software developed by the TUDelft that already contained many of those components and as such repeating this information in his thesis was not necessary. Therefore, while a number of these thesis scored a large number of N's, that does not mean they are bad resources. These works simply had a different end goal than the author of this thesis.

Thesis and Dissertation Review Results

There are two major takeaways from portion of the literature review. First, launch vehicle sizing software has been developed on a number of occasions before. However, a thorough review of past-to-present launch vehicle design literature has not been performed up to this point. Second, there are many different aspects to launch vehicle sizing. Solid, liquid, and hybrid rocket motors, or some combination thereof, are all available propulsion options. Several different options for staging exist, and these options can be combined with one another.

While these theses incorporate some or all of these elements of design into their design software, the launch vehicle sizing software is not setup in such a way it would be easy for the user to switch between major design variables. The software is such that a single type of design is done and any drastically different components requires a major overhaul of their work. (An exception to this is Silva Mota's thesis in which he defines his primary research objective as "*...to develop a tool which can be easily reused and extended to model and simulate a launch vehicle…"* . [104] However, there are several elements of design that are needed in today's launch vehicle design software, such as the boostback reusability, which his thesis does not address. Additionally, his thesis does not include all of the steps necessary to replicate his software.)

Table 2.5. Summary Results of Theses and Dissertation Literature Search

Source Full Name	Author	Thesis/ Dissertation	Year Published	Literature Review	Method (Analytic, Graphical, Empirical)	# of Y's	# of N's
Optimization and Design for Heavy Lift Launch Vehicles [101]	Paul Andreas Ritter	Thesis	2012	N	Analytic	6	8
Design Optimization of Space Launch Vehicles Using a Genetic Algorithm [102]	Douglas James Bayley	Dissertation	2007	N	Analytic	8	6
Design Optimization of Ground and Air-Launched Hybrid Rockets [103]	Francisco Miranda	Thesis	2015	Y*	Analytic	4	10
Conceptual Lay-out of a Small Launcher [105]	Claire Ballard	Thesis	2012	N	Analytic	8	6
Modeling and Simulation of Launch Vehicles Using Object-Oriented Programming [104]	Fabio Antonio da Silva Mota	Thesis	2015	N	Analytic	4	10
A Methodology to Link Cost and Reliability for Launch Vehicle Design [106]	Zachary C. Krevor	Dissertation	2007	Detailed	Analytic	6	8
MULTISTAGE LAUNCH VEHICLE DESIGN WITH THRUST PROFILE AND TRAJECTORY OPTIMIZATION [107]	Ezgi Civek Coskun	Dissertation	2014	N	Analytic	5	9
A Tool for Preliminary Design of Rockets [108]	Diogo Marquest Gaspar	Thesis	2014	Basic	Analytic	6	8

Table 2.5. Summary Results of Theses and Dissertation Literature Search (cont.)

Source Full Name	Author	Thesis/ Dissertation	Year Published	Literature Review	Method (Analytic, Graphical, Empirical)	# of Y's	# of N's
Air Launch versus Ground Launch: a Multidisciplinary Design Optimization Study of Expendable Launch Vehicles on Cost and Performance [109]	M. W. van Kesteren	Thesis	2013	N	Analytic	8	6
Commercial Launch Vehicle Design and Predictive Guidance Development [110]	Matthew R. Tetlow	Thesis	2003	N	Analytic	8	6
Multidisciplinary Design Optimization of Launch Vehicles [111]	Mathieu Balesdent	Dissertation	2012	Basic	Analytic	4	10
Multidisciplinary Design Techniques Applied to Conceptual Aerospace Vehicle Design [112]	John Robert Old	Dissertation	1993	Basic	Analytic	5	9
Performance Study of Two-Stage-To-Orbit Reusable Launch Vehicle Propulsion Alternatives [113]	Marc A. Brock	Thesis	2004	Basic	Analytic	3	11
Analysis of a heavy lift launch vehicle design using small liquid rocket engines [114]	David Phillip Russ	Thesis	1988	N	Analytic	4	10

2.4 Conclusion and Software Specification

A rich history exists for general rocketry and the use of rockets to access space, and a plethora of resources are available which provide instruction on how to size launch vehicles. However, there is a clear need for a program that can size launch vehicles and is easily modifiable in the public domain. Based on the primary design disciplines introduced in Section 1.1.1 and the role of these disciplines in launch vehicles as discussed in Section 2.2.3, modern launch vehicle design software needs the following capabilities:

1. Calculate the energy requirements of the launch vehicle to reach a desired orbit with a pre-determined payload capacity;

2. Find the masses of the components of the launch vehicle;

3. Size the propulsion system to meet the propulsion requirements;

4. Determine the mass and volume of the propellants;

5. Develop the vehicle's basic geometry;

6. Simulate launch-to-orbit trajectory with aerodynamic analysis;

7. Provide a basic stability and control analysis; and

8. Generate a cost estimate for how much the launch vehicle would be to develop, produce, and what would be charged to a customer in order to make a profit.

Due to the limiting time frame of a master's of science research undertaking, the scope of the prototype of the new software is reduced:

- From the types of launch vehicles discussed in Section 2.2.1, it was found that only serial staging and parallel staging are used with regularity. In order to size a parallel stage, the core stage and booster must be sized simultaneously which leads to significant numerical complexities. Therefore parallel staging is not included in the prototype.

- The elements of design included three types of reusability: boostback, fly/glide back, and parachute recovery. Out of these, only boostback reusability is part of the prototype.

- As the goal of this thesis is launch vehicle design and not propulsion design, the software will required a rocket engine to be provided as an input instead of sizing a new engine specific to the launch vehicle.

- Structural analysis plays a limited role in the conceptual design phase, and as such is also excluded from the prototype.

The software will be written in Python to allow it to be run on any system and be easily integrated with existing AVD software. Each part of launch vehicle sizing will be performed in its own module. By dividing the sizing up in this manner modules can be added or removed as desired to change the vehicle's characteristics as desired as well as integrate in new aspects of launch vehicle design, such as parallel staging, with limited modifications required to existing modules.

Chapter 3. Launch Vehicle Design

As mentioned in Section 1.2, the goal of this thesis is to produce a program capable of sizing expendable and boostback-reusable launch vehicles that can be used in conjunction with current and upcoming AVD software. The program must be malleable so it can size a variety of vehicles based on the provided mission objectives and desired launch vehicle properties. The program will be written in Python so it may be directly integrated into the space architecture tool designed by Coley [13].

With the exception of tables, figures, and specific information, this remainder of this chapter after the module introduction defines the primary references used to develop and verify it. The information and equations discussed in the below sections maybe be found in a large variety of texts, including the some of the ones discussed in Chapter 2 and in Appendix C. This chapter will contain most of the equations necessary to code the program.

3.1 Capabilities, Assumptions, and Limitations

Capabilities

The prototype of the system has the following capabilities:

- Sizing of expendable launch vehicles, partially reusable launch vehicles where the first launch vehicle can be recovered via boostback, or expendable launch vehicles whose last stage contains additional propellant for after reaching the parking orbit.

- Producing the basic geometric information for the launch vehicle (lengths and diameters for each stage and the total vehicle).

- Determining mass estimates for specific subsystems.

- Simplicity of use and flexibility of scope, allowing the user(s) to easily modify the program to size launch vehicles that include other features. A user’s guide and a programmer’s guide with instructions on how to run and modify the software may be found in Appendix F. and Appendix G.

Assumptions

The prototype of the system has the following assumptions:

- The payload's mass and geometric data are known and provided by the user.

- All engine data (mass, thrust, I_{sp}, etc.) are known and provided by the user.

Limitations

The prototype of the system has the following limitations:

- **The launch vehicle uses only serial staging.**
  Parallel staging introduces significant difficulties into the sizing process, and as such are excluded from the first version of the program's capabilities. Piggyback and engine staging are rarely-used staging methods and are also excluded with no plans to incorporate them at a future date, but the software is setup such that they can included in the future.

- **The calculated $\Delta V_{required}$ for the parking orbit is for a circular orbit around the Earth.**
  Other types of orbits, including transfer orbits to other celestial bodies, are currently not supported.

- **Only liquid-fueled rocket engines are permitted.**
  It is uncommon for solid rocket motors to be used in serial stages of launch vehicles, and hybrid rocket engines are still relatively new technology that have seem limited applications. As such they are both excluded from the prototype system.

- **The propellant tank diameters are equivalent to the stage diameter.**
  The increase in diameter provided by items the propellant tank, tank insulation, and structure thicknesses have a small effect on the overall size. For example, consider the Falcon 9. From the information in Sections 4.3 and 4.4, the stage diameter is 3.7 meters. If the Falcon 9's propellant tank, tank insulation, and structure thickness were 0.0127 meters (0.5 inches) thick, well above what would be expected, their combined thicknesses would only be 2.05% of the total stage diameter and can be considered negligible during early conceptual design.

3.2 Program Overview

In order to develop an easily modifiable a program, the software has been broken down into modules. Each module contains one portion of the sizing. For example, there is a module called "Propulsion Module (Given Engine)" that calculates the thrust requirements and then the number of engines required based on provided engine information and the determined vehicle launch mass from an earlier module. If the user instead wishes to create a new engine based on the thrust requirements, a new module could be created to estimate engine performance. This new module would then replace the old one. While the input and output modules may also require adjusting, the remainder of the modules used in the sizing process would remain untouched.

Designing the system to calculation specific information through modules also allowed the code to be created in parts and tested individually. Each module has been verified using the examples from the source it was coded from and, as will be explained in Chapter 4, the entire system was verified using point data verification with the Falcon 9 Full Thrust (expendable), Saturn V, and Falcon 9 Full Thrust (Reusable).

To aid with the creation of the program, a Nassi–Shneiderman (NS) diagram was created before beginning work and updated occasional during the coding process. An NS diagram is a visual depiction of the program that details its flow and the logic used. The NS diagram for the system may be seen in Figure 3.1.

The system sizes vehicles through the below process:

1. Request inputs from the user.

2. Calculate the required ΔV needed to reach the destination orbit.

3. Generate the mass of each stage for the vehicle and each stages component masses.

4. Determine the propulsion system information for each stage.

5. Find the masses and volumes for the fuel and oxidizer for each stage.

6. Size the physical geometry of each stage and the total vehicle.

7. Obtain a second estimate for the structure and equipment mass of each stage based on estimates for specific system masses. If the original estimate minus the second estimate is within a specified margin range, then it is a potential design to consider evaluating. If it not, then the vehicle is designated as an invalid design, the sizing process ends, and the user input for π_{se} must be revised.
   The mass module includes the ability to bypass the sizing termination if the user wishes to evaluate designs which will fail this check. If the bypass is enabled, the evaluation will still be performed, but the sizing process will not end and the results of the evaluation will be included in the output file.

8. Perform a cost analysis.

9. Output all the information.

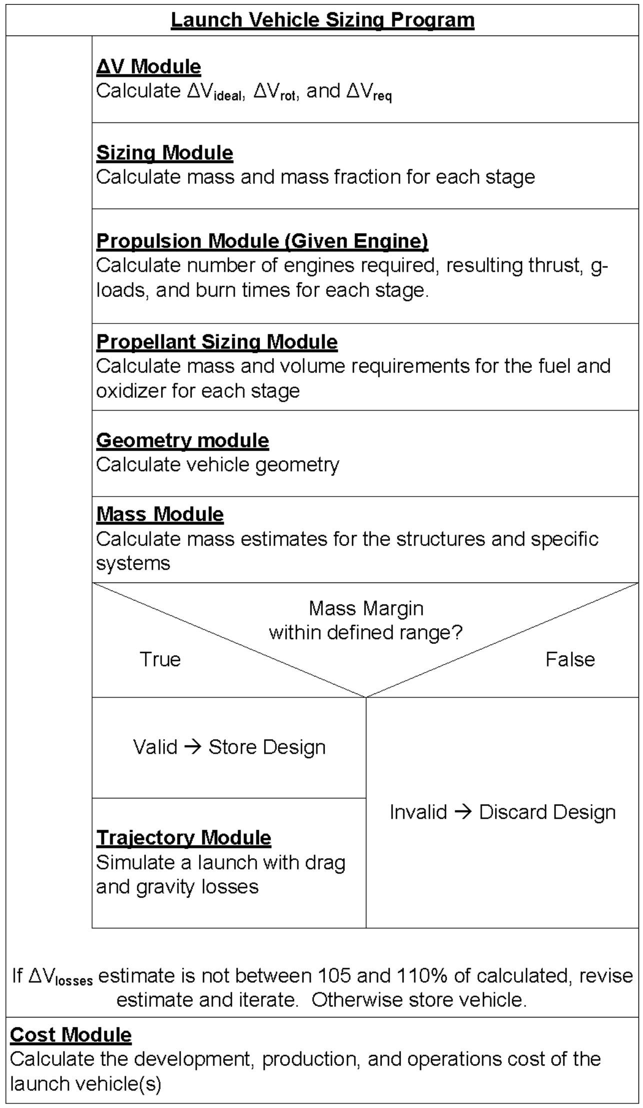

Figure 3.1 Launch Vehicle Sizing Program NS Diagram

3.3 The Main Function

The main function is the primary module that calls and runs all other modules. If the user wishes to make a change to the modules used to size a launch vehicle, it is here where they will make the change.

3.4 Input Module

The input module contains all of the information needed to size a launch vehicle and is created by the user prior to running the software. The User Guide in Appendix F. contains the variables that must be provided by the user and the method to create a new input file.

3.5 *ΔV* Module

The *ΔV* module calculates the required change in velocity the launch vehicle must be able to produce to put its payload into the desired orbit and was created using the information provided in Curtis and Walter [84], [85]. Figure 3.2 contains an NS diagram summarizing the programming flow for this module.

ΔV Module
Inputs Launch location information; estimated velocity losses
Calculate ΔV_{ideal}
Calculate $\Delta V_{rotation}$
Calculate $\Delta V_{required}$
Output Calculate $\Delta V_{required}$

Figure 3.2 NS Diagram of the *ΔV* Module

3.5.1 Circular Orbits

In order to maintain an orbit around a celestial body, the object must have a certain velocity vector and magnitude. This is fairly straightforward: the object in orbit has a velocity vector tangent to its orbit and experiences acceleration due to gravitational attraction towards the celestial object at the center of its orbit. This acceleration due to gravity shifts the resulting velocity vector slightly towards the object, resulting in the object moving in orbit around the celestial body. For a circular orbit, the velocity vector's magnitude and orbit radius remain constant at all times. This is visually displayed in Figure 3.3, and the magnitude of this velocity vector can be expressed mathematically with Equation (3.1). In this equation, G is the gravitational constant, 6.67408E-11 $m^3\ kg^{-1}\ s^{-2}$; M is the mass of the celestial body which the object is orbiting, 5.972E24 kg for the Earth; and r_{orbit} is the radius of the orbit measured from the center of the celestial body to the orbital altitude.

This equation is more commonly displayed as seen in Equation (3.2). Here, $r_{celestial\ body}$ is the radius from the center of the celestial body to its surface, and h_{orbit} is the distance from the celestial body's surface to the orbit. For the Earth, $r_{celestial\ body}$ is equivalent to the radius of the Earth, 6,374E3 m, and h_{orbit} is the altitude of the orbit.

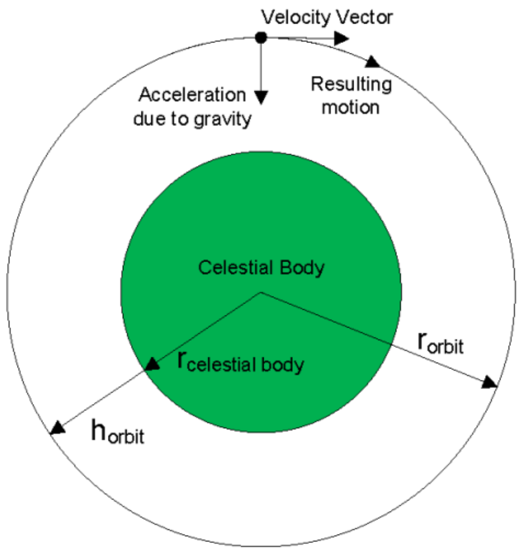

Figure 3.3 Object in a Circular Orbit

$$V_{circular} = \sqrt{\frac{GM}{r_{orbit}}} \tag{3.1}$$

$$V_{circular} = \sqrt{\frac{GM}{r_{celestial\ body} + h_{orbit}}} \tag{3.2}$$

To put a payload into a circular orbit, a launch vehicle must have enough potential energy to reach the desired orbital altitude while simultaneously increasing the payload's velocity until it is equal to the velocity required to maintain that orbital altitude and pitching such that the velocity vector is tangent to the circular orbit at burnout.

3.5.2 Velocity Losses and Benefits

However, the launch vehicle must have the significantly more potential energy than just what is required to reach the orbital velocity because of losses incurred during flight. These losses can occur from a variety of sources, and the two most commonly discussed losses are gravity losses and drag losses. Other losses exist, such as steering losses (also referred to as yaw losses), although they are usually not considered at the conceptual design level because their impact is very small compared to gravity and drag losses. The sum of these losses makeup the total ΔV_{losses} (see Equation (3.3).) The gravity losses tend to makeup the largest portion of ΔV_{losses}.

$$\Delta V_{losses} = \Delta V_{gravity} + \Delta V_{drag} + \Delta V_{other\ losses} \approx \Delta V_{gravity} + \Delta V_{drag} \tag{3.3}$$

In order to properly determine ΔV_{losses}, a trajectory simulation must be run that takes gravity and atmospheric effects into consideration. Since a simulation can't be run before sizing the vehicle, an estimate for the losses is used during initial sizing. Most texts, such as references [83], [85], [115], and [116], recommend an assumption of 1,524 m/s to 1,676.4 m/s.

In additional to the losses discussed above, the rotation of the Earth (or any celestial body that is being launch from) must be taken into consideration. If the payload's orbit will be in the same direction as the rotation of the Earth, the Earth's rotation is beneficial and results in a decrease in the amount of ΔV required. This is referred to as a prograde orbit. If the payload's orbit is in the opposite direction as the Earth's rotation, an increase in ΔV is required and the orbit is called a retrograde orbit. The benefit or loss provided by the Earth's rotation may be calculated with Equation (3.4), in which ω is the latitude of the launch location in radians and the resulting $\Delta V_{rotation}$ is in meters per second. Note that from the equation it can be seen that the largest benefit from the Earth's rotation could be acquired by launching at the equator, and that the potential benefit is reduced as the latitude approaches either pole.

$$\Delta V_{rotation} = \pm 464 * \cos(\omega) \tag{3.4}$$

Summing together the ideal velocity required for a circular orbit, estimated losses incurred, and the benefit or penalty from the launch latitude provides $\Delta V_{required}$, the actual ΔV the launch vehicle must produce in order to put the payload into its target orbit.

$$\Delta V_{required} = \Delta V_{ideal} + \Delta V_{losses} \pm \Delta V_{rotation} \tag{3.5}$$

The final step in the ΔV module calculates ΔV_{stage}, the change in velocity that each stage will be required to provide. The fraction of ΔV that each stage should provide varies based a number of factors, such as the specific the mission requirements, available technology, and number of stages. It is up to the designer to determine what specifically the split should be.

3.6 Sizing Module

The sizing module calculates the various mass information for each stage and the total vehicle and was created using the process and examples provided by Wood [83]. Figure 3.4 contains an NS diagram summarizing the programming flow for this module.

In order to explain this part of the code, several variables and relationships must be defined first. As displayed in Figure 3.5, the mass of a launch vehicle can be broken up into three primary components: the payload; the propellant; and the structures and equipment. Note that the propellant tank masses are included in the structures and equipment mass and thus "propellant mass" refers to just the masses of the fuel and oxidizer.

In the case of a multi-stage launch vehicle, each stage's mass can be broken down in a similar matter: a payload, which is all later stages and the vehicle's primary payload; the propellant used only by the current stage; and the current stage's structure and equipment. See Figure 3.6. This structure and equipment mass may or may not include the interstage fairing that connects the current and the next stage together. For the purposes of this thesis, the interstage fairing is considered part of this mass.

Mathematically, this breakdown is represented by Equation (3.6).

$$m_{initial} = m_{payload} + m_{propellant} + m_{se} \tag{3.6}$$

If both sides are divided by $m_{initial}$, the equation becomes

$$1 = \frac{m_{payload}}{m_{initial}} + \frac{m_{propellant}}{m_{initial}} + \frac{m_{se}}{m_{initial}}$$

$$1 = \pi_{payload} + \pi_{propellant} + \pi_{se} \tag{3.7}$$

Sizing Module	
Inputs Number of stages; $\Delta V_{required}$; ΔV split; payload mass; I_{sp} information; π_{se}	
For each stage	
	Calculate MR
	Calculate $\pi_{burnout}$ (AKA π_{final})
Starting with the last stage and working towards the first stage	
	Calculate $\pi_{payload}$
	Calculate initial mass
	Set the initial mass for the current stage to the payload mass for the prior stage
For each stage	
	Calculate $\pi_{propellant}$
	Calculate mass of the propellant and dry mass
Calculate initial mass, propellant mass, and $\pi_{payload}$ for the total vehicle	
Output Total and fractional mass information for each stage as well as the total vehicle	

Figure 3.4 NS Diagram of the Sizing Module

where

$$\pi_{payload} = \frac{m_{payload}}{m_{initial}} \tag{3.8}$$

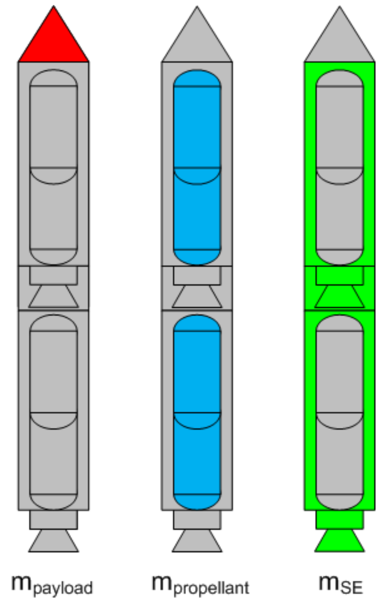

Figure 3.5 Mass components for the total launch vehicle

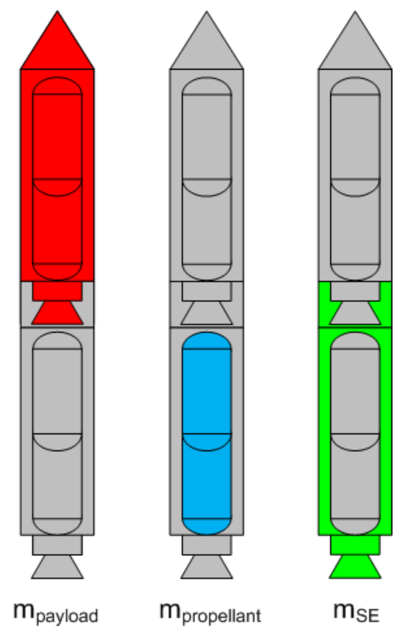

Figure 3.6 Mass components for the first stage of a launch vehicle

$$\pi_{propellant} = \frac{m_{propellant}}{m_{initial}} \tag{3.9}$$

$$\pi_{se} = \frac{m_{se}}{m_{initial}} \tag{3.10}$$

Additionally, the initial mass minus the propellant mass is defined as the final mass or burnout mass. Some texts will also refer to this as the dry mass, although the majority defines "dry mass" to be equivalent to m_{se} (the mass of a single stage without any payload or propellant).

$$m_{final} = m_{burnout} = m_{initial} - m_{propellant} = m_{payload} + m_{se} \tag{3.11}$$

As before, dividing all sides by m_{intial} provides another useful mass fraction, π_{final} or $\pi_{burnout}$.

$$\pi_{final} = \pi_{burnout} = \pi_{payload} + \pi_{se} \tag{3.12}$$

This can be rearranged to

$$\pi_{payload} = \pi_{burnout} - \pi_{se} \tag{3.13}$$

One final ratio is required before beginning the sizing process. This is the mass ratio, MR, which is defined in Equation (3.14).

$$MR = \frac{m_{intial}}{m_{final}} = \frac{1}{\pi_{burnout}} \tag{3.14}$$

Rearranging this equation provides

$$\pi_{burnout} = \frac{1}{MR} = \frac{m_{final}}{m_{initial}} \tag{3.15}$$

By manipulating these relationships and using the rocket equation, a launch vehicle can now be sized.

3.6.1 Sizing Process

The rocket equation, also called the Tsiolkovsky equation, is

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

$$\Delta V = g_o * I_{sp} * \ln(MR) \quad \textbf{(3.16)}$$

Where

ΔV is the ideal change in velocity;

g_o is the acceleration due to gravity;

$I_{sp,\ mean}$ is the specific impulse of the engine(s) used; and

MR is the previously defined ratio of the initial mass to the final mass.

The rocket equation relates performance, propulsion, and weight and balance together in a single equation. By solving this equation for *MR*, the equation becomes

$$MR = e^{\frac{\Delta V}{g_o * I_{sp,mean}}} \quad \textbf{(3.17)}$$

If there are multiple stages to a vehicle, the MR for an individual stage can be found by replacing ΔV with appropriate ΔV_{stage} that was calculated in the ΔV module and by using that stage's average specific impulse.

$$MR_{stage} = e^{\frac{\Delta V_{stage}}{g_o * I_{sp,mean}}} \quad \textbf{(3.18)}$$

This equation and Equation (3.15) are looped through for each stage to find *MR* and $\pi_{burnout}$ for each stage. With this information now known, the initial launch mass for each stage can now be calculated one stage at a time, starting with the final stage. First, $\pi_{payload}$ is found with Equation (3.13) using the calculated $\pi_{burnout}$ and assumed π_{se}. Second, the initial mass for the stage can then be calculated via Equation (3.8). Third, the payload mass for the previous stage is set as the initial mass for the current stage, and then the process is iterated until $m_{initial}$ for all stages is known.

At this point enough information is known to calculate $\pi_{propellant}$, $m_{propellant}$, and $m_{burnout}$ by rearranging Equations (3.7), (3.9), and (3.11) appropriately. Once all of the information for the individual stages is known, the various masses and mass fractions can also be calculated for the total vehicle as well as m_{se} for each stage by using the appropriate definitions.

3.7 Propulsion Module (Given Engine)

The next step in the launch vehicle sizing process is to determine the propulsion system requirements and was created using the method provided in Wood to find the number of engines required for a launch vehicle based on a selected engine and calculated initial mass [83]. The process for this module is shown in the NS diagram in Figure 3.7.

The first step is to determine the thrust required to meet the minimum launch thrust-to-weight ratio, *T/W*. Once the thrust requirements have been calculated, the number of engines can be found by dividing this thrust requirement by the thrust per engine and rounding up to the nearest integer. The total actual thrust is then found by multiplying the thrust per engine by number of engines used. This is mathematically displayed in Equations (3.19) through (3.21). Note that as the performance of an engine varies with altitude due to the change in atmospheric pressure, this value is actually an estimate of the average total thrust. For the purposes of conceptual design this number is fine to use, although during preliminary design the change in thrust with altitude should be accounted for in the trajectory simulation.

$$T_{total,required} = (T/W)_{min} * m_{initial} * g_o \tag{3.19}$$

$$n_{eng} = \left(\frac{T_{total,required}}{T_{eng}}\right)_{round\ up\ to\ nearest\ integer} \tag{3.20}$$

$$T_{total,} = T_{eng} * n_{eng} \tag{3.21}$$

In the module, two checks now occur. The first ensures the actual *T/W* does not exceed the specified maximum value by rearranging Equation (3.19) and replacing $T_{total,\ required}$ with T_{total} to find $(T/W)_{actual}$. If it is found that $(T/W)_{actual}$ is greater than $(T/W)_{max}$, then a warning is displayed to the user that either a different engine will need selected or the *T/W* range needs modified. The second check looks for what g-load is experienced by the payload when the stage reaches burnout. If it exceeds the max allowed, then the user is informed that either throttling must occur or one of the engine will need shut off during the flight in order to keep from exceeding this limit and potentially damaging the payload.

The entire process is now repeated for each stage until information for all stages has been calculated.

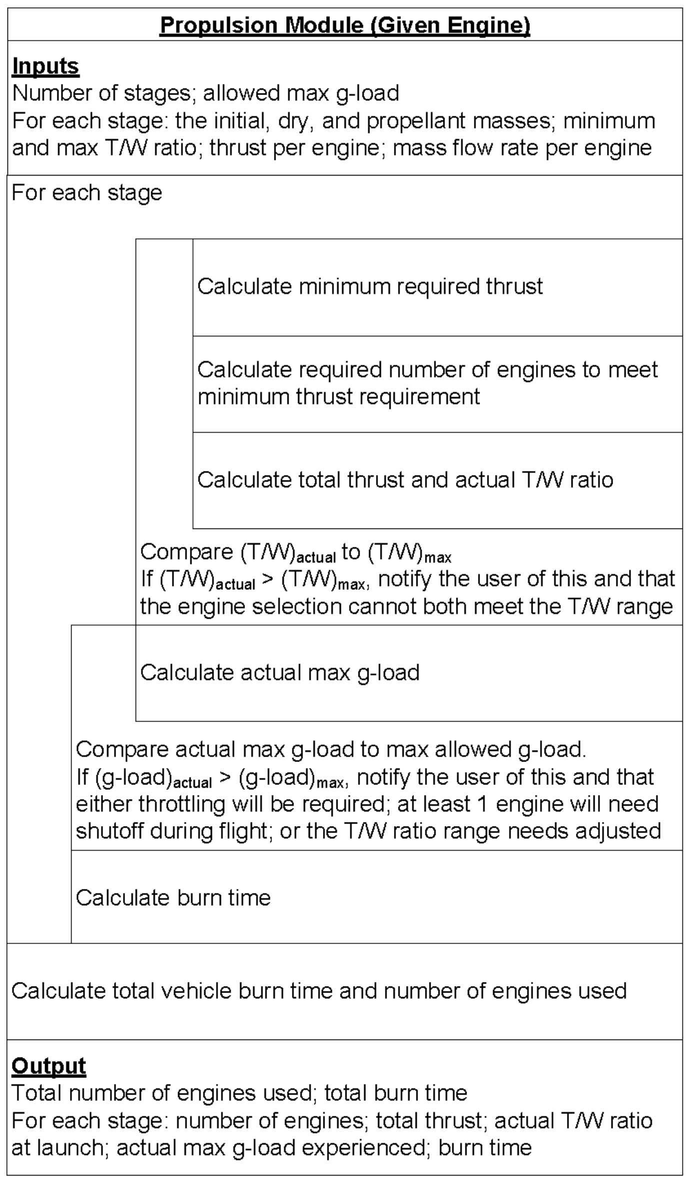

Figure 3.7 NS Diagram for the Propulsion Module

3.8 Propellant Module

The propellant sizing module's process is shown in Figure 3.8. This module is one of the simplest in the program, and was not created with any particular reference but by using the fundamental equations relating mass, volume, and density. By using the known propellant mass, the engine's mixture ratio, and the densities for the fuel and oxidizer, the mass as well as volume for the fuel and oxidizer are calculated with Equations (3.22) through (3.25). The calculations are repeated for each stage, and then the information is summed together to find the total mass and volume for the fuel and oxidizer used for the entire vehicle.

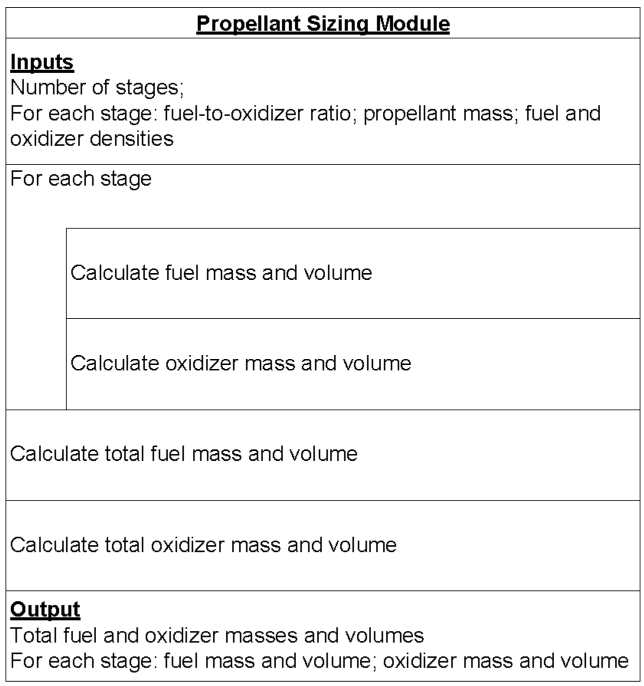

Figure 3.8 NS Diagram of the Propellant Sizing Module

$$m_{fuel} = m_{propellant} * f \tag{3.22}$$

$$v_{fuel} = \frac{m_{fuel}}{\rho_{fuel}} \tag{3.23}$$

$$m_{oxidizer} = m_{propellant} - m_{fuel} \tag{3.24}$$

$$v_{oxidizer} = \frac{m_{oxidizer}}{\rho_{oxidizer}} \tag{3.25}$$

3.9 Geometry Module

With the number of engines and propellant volume for each stage now known, it is possible to make an estimation of the launch vehicle's physical dimensions with the Geometry Module. This module was created using the process described by Wood in his example on determining the geometry of a launch vehicle once its propellant volumes have been determined [83]. This process is summarized with the module's NS diagram which is located in Figure 3.9.

The first step is to size the diameter for each stage. This is done by starting with the final stage and working back to the first stage. The stage's diameter is determined by comparing two values: a diameter required for the payload interface and a diameter required for the number of engines. If the current stage being sized is the last stage, then the payload interface diameter is the interface diameter of the actual payload. If it is not the last stage, then the payload interface diameter is the diameter of the next stage. (I.e., the payload interface diameter for stage one of a multistage rocket is the diameter of stage 2.)

The process for determining the diameter required for the stage's engines is thus: first, the "true" diameter of the engine's nozzle is calculated by adding an engine gap value to the exit diameter of the engine's nozzle. This engine gap is spacing required between engines (see Figure 3.10). This spacing is required for a number of reasons including, but not limited to, preventing damage from acoustic and vibration between engines and for plume expansion. The designer may also want to configure the propulsion system such that the engines are in a specific arrangement. For example, SpaceX uses an what they refer to as an "octo-web" shape for their engine configuration in order to simplify design and assembly [117]. Once the "true" engine nozzle exit diameter has been calculated, this is turned into an area requirement and multiplied by the number of engines to find the total area requirement of the engines. That number is then used to calculate a stage diameter required by the engines.

These two diameter requirements are compared, and whichever value is larger then becomes the diameter for the stage. Something to note here is that this calculated stage diameter is the minimum stage diameter required and may not be the optimal choice. Two examples: first, it may be desirable for a multi-stage launch vehicle to have a single, consistent diameter for manufacturing, transportation, and flight simplicity. Second, for stages with large propellant volumes it may be better to have a much larger diameter than what is required. An increase in diameter can result in a significantly decreased overall length of the vehicle, and this shorter but wider geometry can result in several structural benefits and simplify stage transportation. See Appendix D. for additional details on how vehicle length changes with increasing diameter.

Geometry Module

Inputs
Number of stages; payload interface diameter, $(d_{req})_{payload}$; payload length
For each stage: fuel volume; oxidizer volume; fuel tank thickness and insulation thickness; radius of the pipe that runs through the oxidizer tank; oxidizer tank thickness and insulation thickness; number of engines; length of an engine; diameter of the nozzle exit; spacing gap between engines; length of the gap between stages

Starting with the last stage and working towards the first stage

- Calculate the stage diameter required for the number of engines, $(d_{req})_{engines}$
- Compare $(d_{req})_{payload}$ and $(d_{req})_{engines}$. Set the d_{stage} equal to the larger value
- Set the d_{stage} equal to $(d_{req})_{payload}$ for the prior stage

Calculate the lengths and surface area for the fuel and oxidizer tanks; and the length of the total propellant tank stack

Calculate the length of each stage

Calculate the surface area of the intertank fairing

Output
Total vehicle length
For each stage: stage diameter, lengths and surface areas of the fuel and oxidizer tanks; length of the total tank stack; length of the stage; surface area of the intertank fairing

Figure 3.9 NS Diagram for the Geometry Module

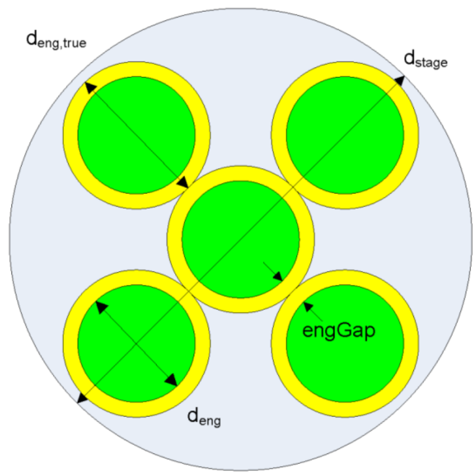

Figure 3.10 Spacing between engines

3.9.1 Propellant Tanks Sizing

The Geometry Module contains a submodule that sizes the propellant tanks. Multiple ways to store a stage's propellant exists, and Figure 3.11 shows five different options used in launch vehicles. There are pros and cons to each of these. As tandem tanks with a common bulkhead and tandem tanks with internal piping (not shown in the figure but nearly identical to the tandem tanks, external piping image) are the most common types of propellant tanks used, submodules have been created to size tanks for either of these options.

The calculations in this submodule are straightforward: each propellant tank is a cylinder with rounded end caps. The volume held by the end caps is calculated first, and then the length of the cylindrical portion is calculated based on the remaining volume to be held. The end cap is a three-dimensional shape called an ellipsoid. An ellipsoid is similar to a sphere, except instead of having one consistent radius each of the three axes may have a unique radius. See Figure 3.12. For the end cap on a cylinder, r_a and r_b are equivalent and, for the purposes of sizing a launch vehicle, are equivalent to the stage's diameter. If the end cap is a hemisphere, then r_c is also equal to r_a. The volume for a single end cap is thus:

$$V_{end\ cap} = 0.5 * \frac{4\pi * r_a * r_b * r_c}{3}$$

$$V_{end\ cap} = \frac{2\pi * r_a^2 * r_b}{3} \tag{3.26}$$

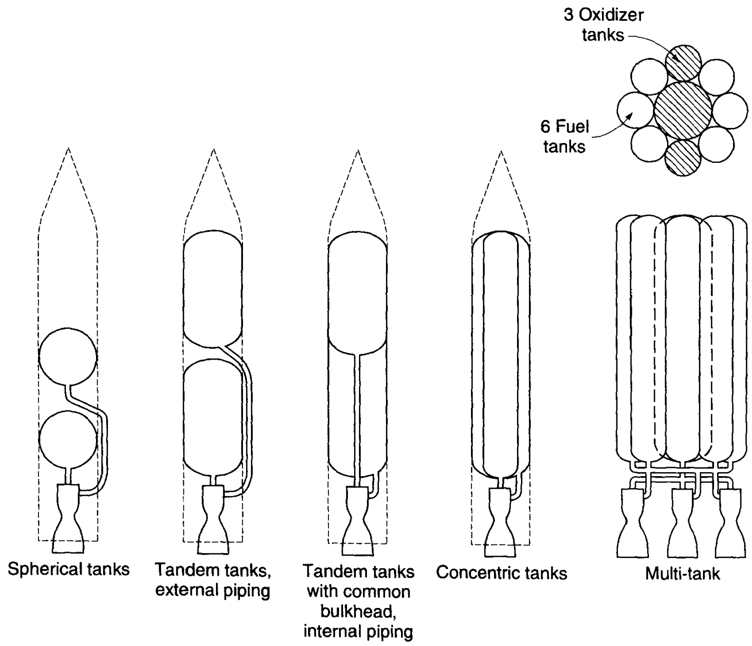

Figure 3.11 Examples of propellant tanks [65]

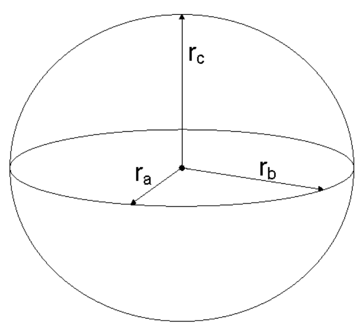

Figure 3.12 Ellipsoid axes

In the case of separate tandem tanks, both the fuel and the oxidizer have two end caps. If the tanks share a common bulkhead, then whichever is on top has two end caps and the other has only one. The volume from the end caps is then subtracted from the total volume required for the fuel or oxidizer. This is the volume the cylinder must hold. As the diameter is fixed at the stage's diameter, what needs to be determined is the length of the cylinder. This is done by rearranging the equation for the volume of a cylinder. Thus:

$$V_{cylinder} = V_{total\ required} - n_{end\ cap} * V_{end\ cap}$$

$$\pi * \frac{d_{stage}^2}{4} * l_{cylinder} = V_{total\ required} - n_{end\ cap} * V_{end\ cap}$$

$$l_{cylinder} = \frac{4\left(V_{total\ required} - n_{end\ cap} * V_{end\ cap}\right)}{\pi * d_{stages}^2} \tag{3.27}$$

This calculation is repeated for both the fuel and the oxidizer tanks. Once the lengths have been found, the total length for the fuel and oxidizer tanks is calculated by adding the cylinder length to the end cap's vertical axis radius, and the total propellant tank stack length is found by summing these two values together. The total surface area of the tanks is also calculated for documentation purposes and for use later in the systems mass module (see Section 3.10).

Once the propellant tank stack length has been calculated, the total stage length can be found by adding this length to the length of the engines used and, if the stage being calculated is not the final stage, the length required between stages. The total vehicle length is then determined by summing together all of the stage lengths and the payload length.

3.10 Systems Mass Check Module

The next module in the system is the System Mass Check Module. The goal of this module is the use equations based on Mass Estimating Relationships, or MERs, to provide estimates for the vehicle's various systems. As previously discussed in the description of INTROS in Section 2.3.1, MERs are developed by building a database of mass information based on existing hardware and creating empirical equations based on this information. The MERs used in this module came from a lecture from Dr. David Akin at the University of Maryland [100]. All of the equations in this lecture have been included in the module; however, as not all of them are needed for the system, a number of calculations commented out. They have been included for future use.

The systems masses for each stage that are calculated by this module are:

- **Mass of the fuel and oxidizer tanks.**

- **Mass of the insulation for the fuel and oxidizer tanks.**

- **Mass of all the engines.**
  Reference [100] includes a MER for calculating the mass of a single engine based

on the thrust and exit area; however, as these are provided inputs, only the total engine mass is calculated.

- **Mass of the thrust structure.**
  The thrust structure is the structure which connects the engines to the rest of the launch vehicle.

The module also calculates three masses for the total vehicle:

- **Mass of the intertank fairing.**
  In the event the propellant tanks are used for the external structure of the vehicle, there is a small gap between the propellant tanks that must be covered. The structure used to do this is called the intertank fairing.

- **Mass of the avionics.**

- **Mass of the wiring.**

Once all the structure and system masses have been estimated, they are summed together and compared to the original estimate for m_{se} derived through mass ratio and mass fraction manipulation in the Launch Vehicle Mass Estimation Module. It is desirable that the original estimate be higher than the estimate obtained through the MERs to provide a margin for mass growth. References [100], [69], and [118] recommend a mass margins between 5 and 30% as historically launch vehicles and spacecraft have seen a significant mass increase over the original estimate during their development [119], [120]. Figure 3.13 indicates how launch vehicles and spacecraft weights tend to fluctuate during the design process.

It is important to note that MERs are a function of the technology available at the time they were developed, and as such need updating to reflect the improvement in technology available. This also means that MERs must be used with caution when attempting to recreate historical vehicles as the available technology and legal requirements for the inclusion of certain features from that time period may results in design points being excluded or included when they should not have been. It is up to the designer to analyze whether or not the results from the Systems Mass Check Module makes sense.

3.11 Trajectory Module

Once a vehicle has been successfully sized, it runs through a trajectory simulation. The ascent-to-orbit trajectory simulation was created based on the information provided by Curtis and Walter on reaching orbit and the sounding rocket example problem from Curtis [84], [85]. The primary goal of a trajectory simulation during the conceptual design phase is to verify the ΔV_{losses}

assumption used in the Sizing Module. If the assumption is too low, then the launch vehicle is incapable of reaching the desired orbit; if it is too high, then the vehicle is larger than what is required for the payload mass.

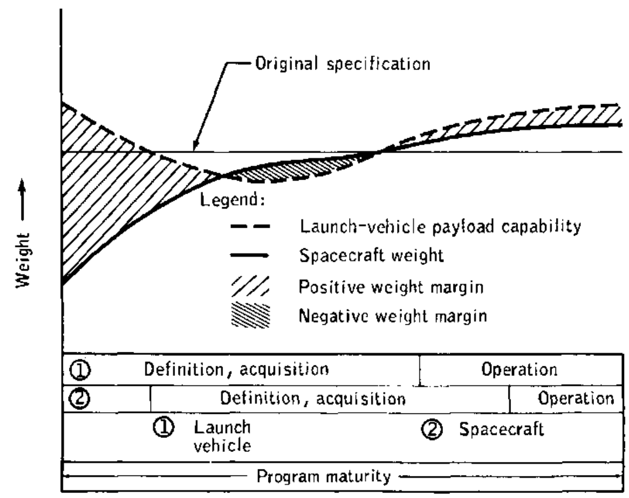

Figure 3.13 Spacecraft Weight and Launch Vehicle Performance vs. Program Maturity [71]

A simplified view of a launch vehicle during its ascent-to-orbit is show in Figure 3.14. Six differential equations are required to simulate its motion and calculate ΔV_{losses}. These equations are derived in many texts on orbital mechanics, such as references [69], [84], and [85]. The reader is directed to one of those references for information on the equations' derivations. The equations for acceleration, velocity in the vertical and downrange directions, and losses for drag and gravity are:

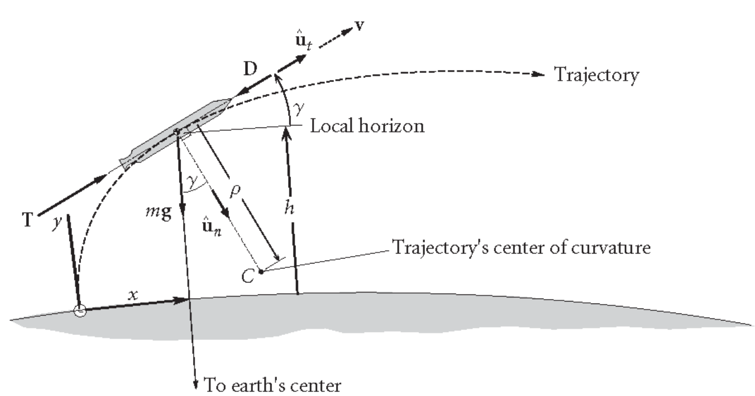

Figure 3.14. Launch Vehicle Ascent Trajectory [84]

$$\dot{v} = \frac{T - D}{m} - g \tag{3.28}$$

$$\dot{h} = v * \sin(\gamma) \tag{3.29}$$

$$\dot{x} = \left(\frac{R_E}{R_e + h}\right) * v * \cos(\gamma) \tag{3.30}$$

$$\Delta V_{drag} = -\frac{D}{m} \tag{3.31}$$

$$\Delta V_{gravity} = -g * \sin(\gamma) \tag{3.32}$$

where

$\dot{v}$ is the acceleration;

T is the current thrust;

D is the current drag;

m is the instantaneous mass;

g is the acceleration due to gravity;

$\dot{h}$ is the vertical velocity;

γ is the flight path angle, the angle between the horizon and the vehicle's velocity vector;

$\dot{x}$ is the downrange velocity;

R_E is the radius of the Earth;

h is the current altitude; and

v is the current velocity.

The sixth equation, the equation for the change in γ with respect to time ($\dot{\gamma}$), is more complicated. As first introduced in Section 2.2.2, the launch vehicle needs to not only reach the desired altitude and velocity maintain orbit, but also adjust its velocity vector such that it is in the proper direction to maintain the orbit's shape. Rather than expending large amounts of propellant to accomplish this, gravity is used to adjust the launch vehicle's trajectory. If a pure gravity turn trajectory is used (i.e., no powered steering), then

$$\dot{\gamma} = -\left(\frac{g}{v} - \frac{v}{R_E + h}\right) * \cos(\gamma) \tag{3.33}$$

A gravity turn is not the most optimal trajectory for launch in the atmosphere due to drag losses and a maximum dynamic pressure limitation on the vehicle's structure. Instead, the launch vehicle uses a combination of gravity and powered steering. For an actual launch, the flight path is pre-determined and the vehicle makes pitch adjustments through thrust gimbaling in order to compliment gravity and shape the trajectory as desired. The Trajectory Module works in a similar fashion by taking a set of inputs for $\dot{\gamma}$ and the times at which it is supposed to change. Once γ reaches zero, the system locks γ and $\dot{\gamma}$ at zero.

The programming logic for the Trajectory Module is shown in the NS diagram in Figure 3.15. The module takes the vehicle data, various input flight conditions such as the initial position and velocity of the vehicle, the set $\dot{\gamma}$ and its corresponding times, and maximum allowable dynamic pressure and g-load. It runs this information and the simplified equations of motion, described below, through a fourth order Runge-Kutta integrator. The simulation runs until the current stage's propellant is depleted. Then, the flight conditions are stored, the vehicle stages, and the simulation continues using the stored flight conditions as the new initial conditions.

The Trajectory Module contains two checks which can terminate the simulation early. The first, illustrated in Figure 3.16, is a check against the maximum dynamic pressure. While aerodynamic heating is not typically a concern due to the short period of time traveling through the dense atmosphere at high speed, the experienced dynamic pressure puts a stress on the launch vehicle's structure. Exceeding this limit can result in a structural failure and a loss of vehicle and payload, and as such if the limit is exceeded then the simulation is terminated. There are two methods to reduce the dynamic pressure experienced during flight and avoid this limit:

1. Change the flight path. If the launch vehicle's flight path is altered such that the early phase of the flight maintains a high γ, it will enter the thinner portions of the atmosphere more quickly and experience an overall lower max dynamic pressure. However, this increases gravity losses because the vehicle's thrust vector remains nearly opposite in direction to the force of gravity.

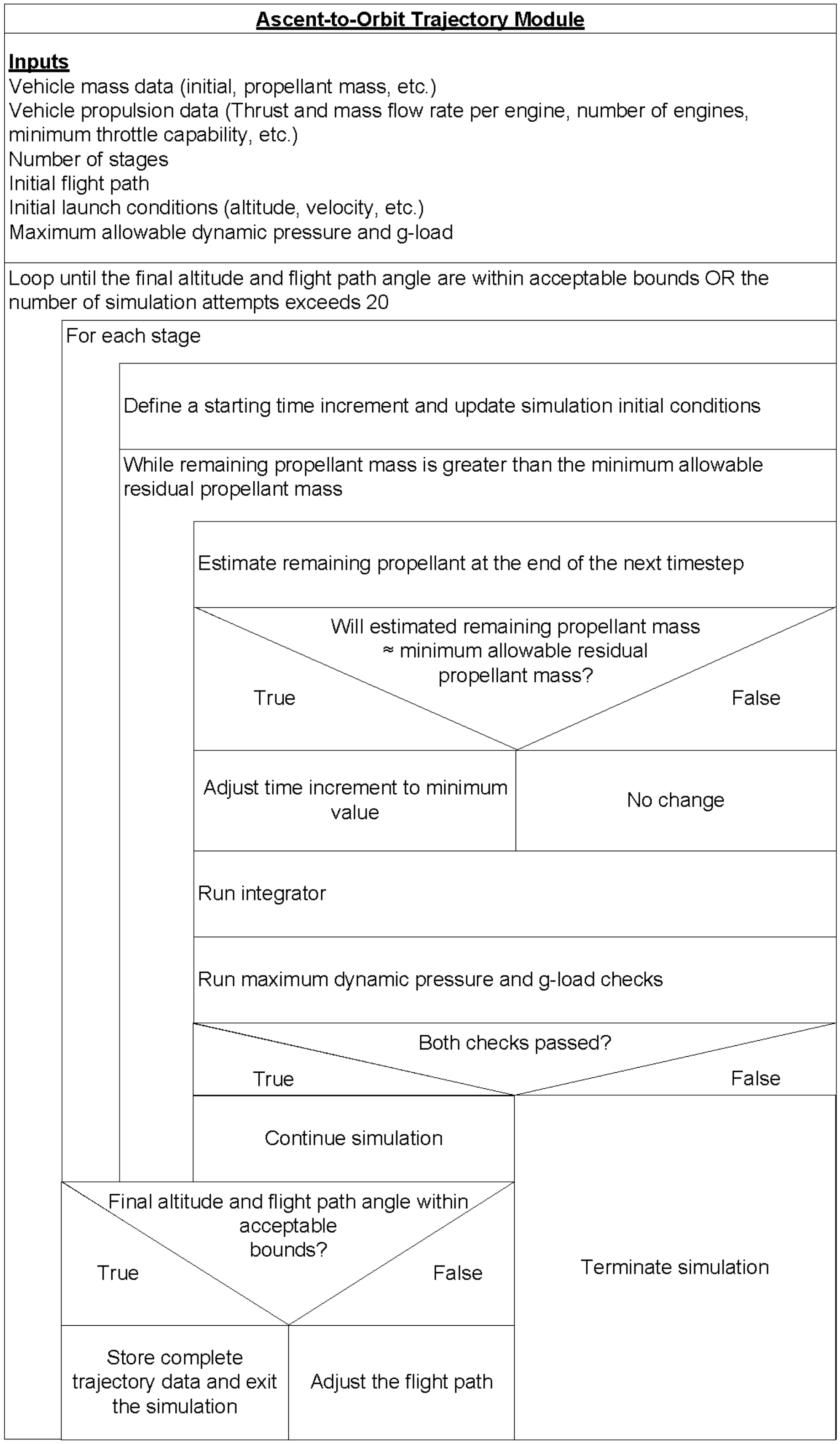

Figure 3.15. NS Diagram for the Trajectory Module

The second check is a check against the maximum allowable g-load. Exceeding this acceleration limit results in damage to the payload and/or the launch vehicle. Unlikely maximum dynamic pressure, the maximum g-load will be experienced when a stage is approaching burnout. Launch vehicles avoid exceeding this limit by adjusting their thrust level through throttling or engine shutdown. An NS diagram of the Trajectory Module's g-load check is presented in Figure 3.17.

Once a simulation has been completed, the Trajectory Module checks the final altitude against the parking orbit altitude and γ_{final}. Either of these is outside of the acceptable bounds (defaulted at ±5% of the parking orbit and 5^o), minor adjustments are made to the provided trajectory path based on the error in altitude and/or γ.

When a successful simulation has been completed, the trajectory data for each individual stage and the total vehicle are compiled and stored.

The Trajectory Module was validated using the first case study (see Chapter 4).

3.11.1 Atmosphere and Coefficient of Drag Submodules

In order to perform all of the calculations necessary to solve Equations **(3.28)** to **(3.32)**, the current drag experienced by the vehicle must be calculated. The drag on an object may be found with

$$D = \frac{1}{2} * \rho * v^2 * C_D \tag{3.34}$$

where

ρ is the current atmospheric density; and

C_D is the coefficient of drag.

In order to determine ρ, the Atmosphere Submodule uses the equations for the 1976 Standard Atmosphere [121] to calculate the temperature and pressure at the current altitude, and use those to calculate the current density. This submodule was validated by comparing answered generated to provided values in lookup tables.

C_D is a function of both geometry and the current Mach number. While there are tools such as DATCOM which can calculate all aerodynamic coefficients, it is the goal of this new software to size a large number of launch vehicles in a short period of time. Integrating in high fidelity tools such as DATCOM would increase run time significantly and introduces potential

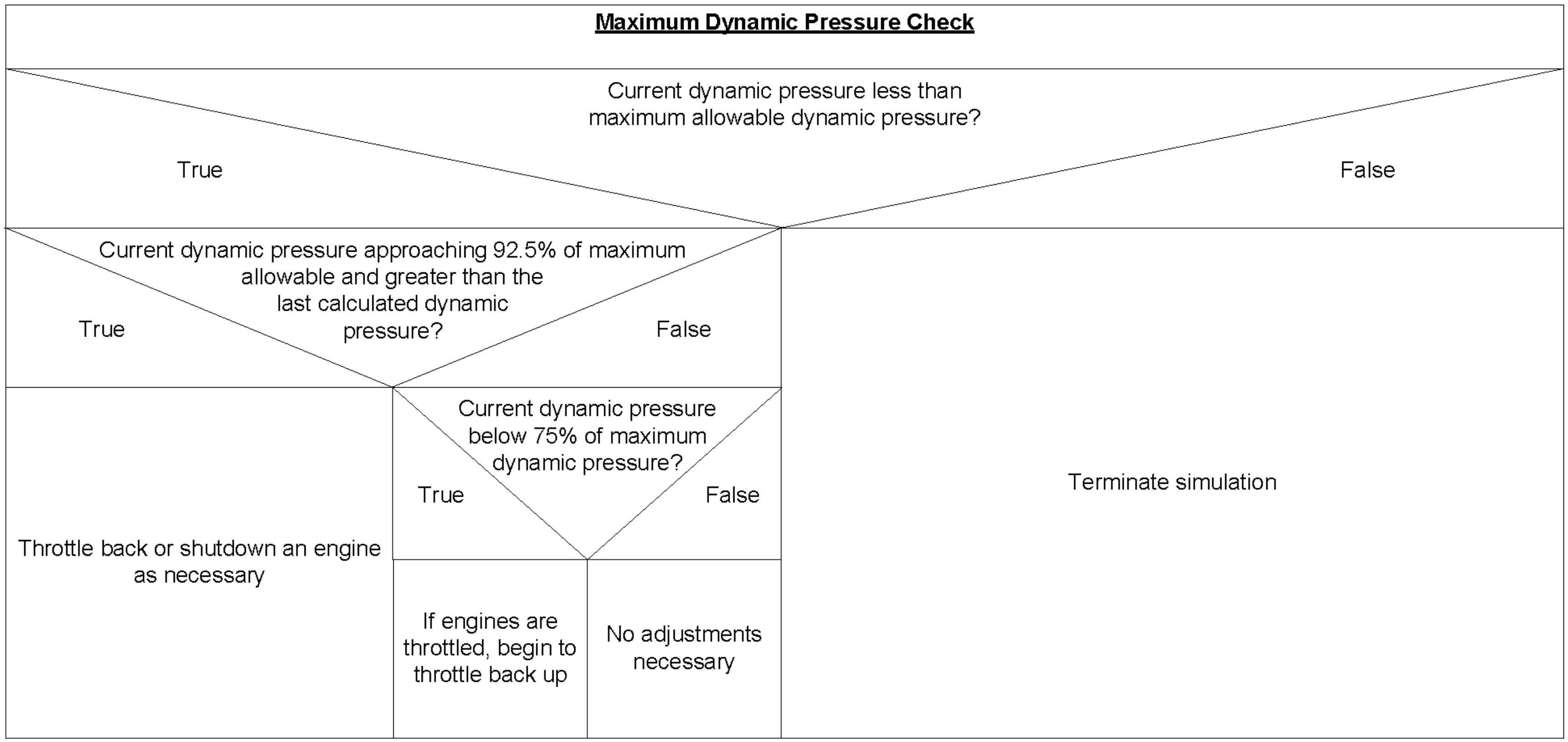

Figure 3.16. Maximum Dynamic Pressure Check

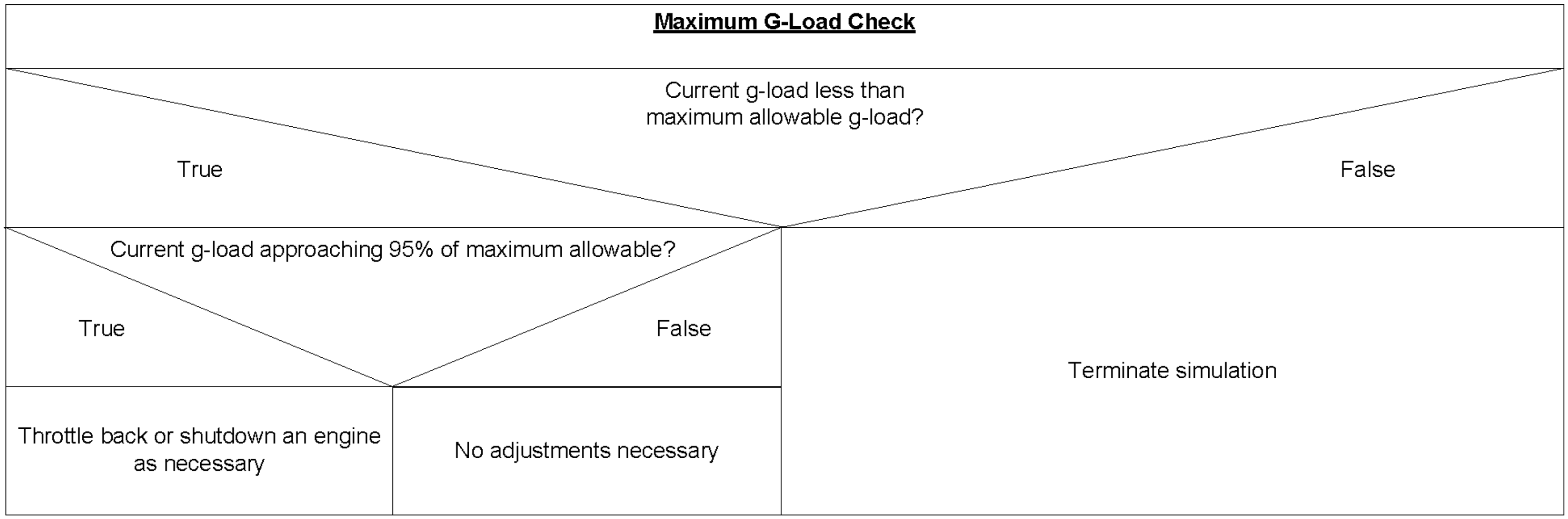

Figure 3.17. Maximum G-Load Check

compatibility problems as this software is improved upon in the future. Thus, a simple method for calculating the C_D of a variety of vehicles at different altitudes and Mach numbers is needed.

Missile Design and System Engineering by Eugene L. Fleeman [122] is a text on the conceptual design of missiles. Included in its section on aerodynamics are equations which approximate the drag coefficient on a missile. As launch vehicles and missiles are very similar in geometry, these equations can also be used to approximate C_{D0} for a launch vehicle. The Coefficient of Drag Submodule takes the current density, Mach number, and vehicle geometry, uses the proper equations from Fleeman, outputs the vehicle's C_{D0}. This submodule was validated using example problems from the text.

3.11.2 Stability and Control Submodule

During a launch vehicle's ascent in the lower atmosphere it may be subjected to significant crosswinds. In order to keep the vehicle stable and maintain its flight path, launch vehicle's use thrust gimballing to make necessary adjustment in their thrust vector. It is desired to calculate the degree to which the engines must be gimbaled in order to maintain the required stability criteria. A launch vehicle's stability is determined by calculating the ratio of control torque (the ability of the vehicle to control its pitch) to aerodynamic torque (the pitch generated by aerodynamic forces). The traditional stability criteria requirement at NASA Marshall Space Flight Center is for the torque ratio to be a minimum of 1.5 at all times. [92]

In "Design of Launch Vehicle Flight Control Augmentors and Resulting Flight Stability and Control" [92], Barret develops the equation necessary to calculate the gimballing angle δ_E in order to maintain the stability criteria, which is the ratio of control torque to aerodynamic torque. This equation is:

$$CR = \frac{T * \sin(\delta_E)\,(E) * l_E}{C_{N,\alpha} * \alpha * Q * S * l_{cp}} \tag{3.35}$$

where

CR is the stability criteria;

T is the current thrust level;

δ_E is the gimbal angle;

E is the number of engines;

l_E is distance between the gimbal plane and the vehicle's current center of gravity;

$C_{N,\alpha}$ is the derivative of the coefficient of normal force with respect to α;

α is the vehicle's current angle of attack;

Q is the current dynamic pressure;

S is the launch vehicle's cross-sectional area; and

l_{CP} is the distance between the current center of pressure and the center of gravity.

Figure 3.18 depicts the distances and angles used in Equation (3.43).

The submodule takes flight data from the most aerodynamically stressful point in the flight trajectory, when maximum dynamic pressure occurs, and calculates the angle required to maintain stability. It was verified using the example provided by Barret [92].

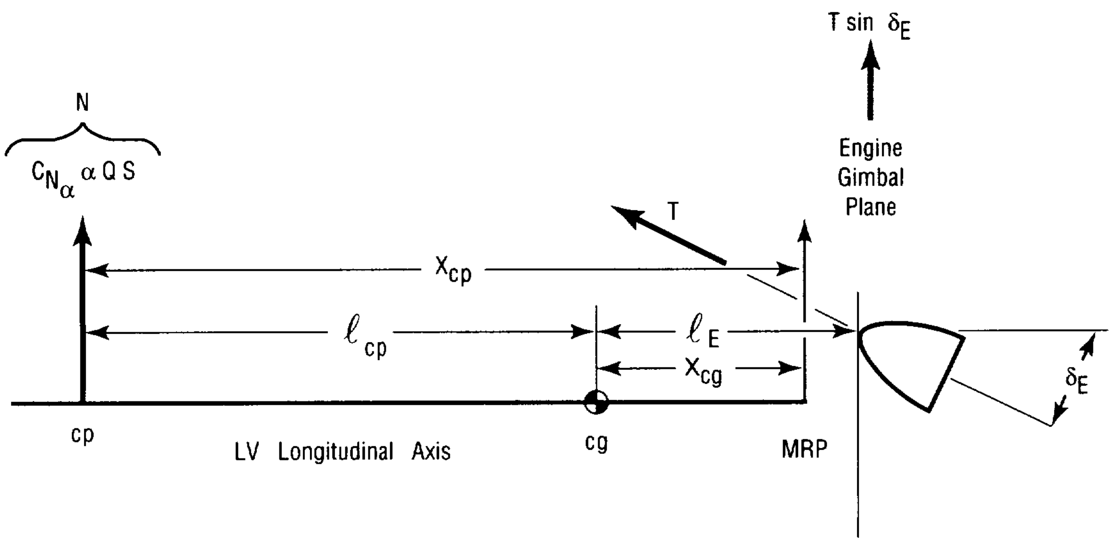

Figure 3.18. Lengths and Angles for Equation (3.43) [92]

3.12 Cost

Early in launch vehicle design history, vehicles have been designed for maximum performance through minimized gross lift-off weight [123]. However, minimum GLOW doesn't necessarily minimize cost, and in today's competitive market cost engineering practices must be considered from the early stages of design [123], [69].

Three components make up launch vehicle design costs:

1. **Development costs**
   Development costs (also referred to as Design, Development, Test, and

Evaluation Costs) are the cost required to design and test the launch vehicle. It makes up the largest portion of costs in terms of raw dollars.

2. **Production costs**
   Production costs are all of the costs related to the manufacturing and production of a launch vehicle, including all testing, quality, and engineering support costs. The vehicle's engines take up a plurality of the production cost (see infographic in Figure 3.19).

3. **Operations costs**
   Operations costs are defined by what is required to take a launch vehicle and send it into orbit. This includes transporting and assembling the launch vehicle stages; the cost of propellants and other consumables for the flight; all costs associated with the launch site and the crew who handles launch operation activities; and any additional fees such as insurance for potential loss of payload or damage to property in the event of a catastrophic failure.

The cost to the launch vehicle service provider is made up of the production and operations costs with some portion of the development costs added in to spread its cost out over the total number the launch vehicles that will be produced and flown. The higher than amount of launches, the smaller the amortized portion is and the lower the cost per flight becomes. A margin for profit is added to the cost per flight which results in the price per launch that is charged to the user.

The Cost Module was developed using the *Handbook of Cost Engineering*, revision 4a with TransCost 8.2, by D. E. Koelle [124]. It dedicates a chapter to each type of cost as well as chapters on how to add the types of cost to determine the cost and price per flight, the history of cost engineering, and examples. The *Handbook* estimates costs through the use of Cost Estimating Relationships (CERs). CERs are similar in nature to MERs: actual cost data for launch vehicles is plotted against relevant vehicle data and normalized using certain cost factors, and the equation for the resulting trend line becomes the CER. The *Handbook* develops the CER as such:

First, all actual cost data is converted to from their currency value (USD, Euros, Yen, etc.) into "Work-Years" (WYr). The *Handbook* defines WYr as

"*The Work-Year (WYr) cost are by definition the total company annual budget (excluding subcontracts) divided by the number of productive full-time people. This means that all secondary costs like office cost, travel, material, etc. as well as taxes and profit are* *included, plus a certain share of administration, management and support staff costs.*" [125]

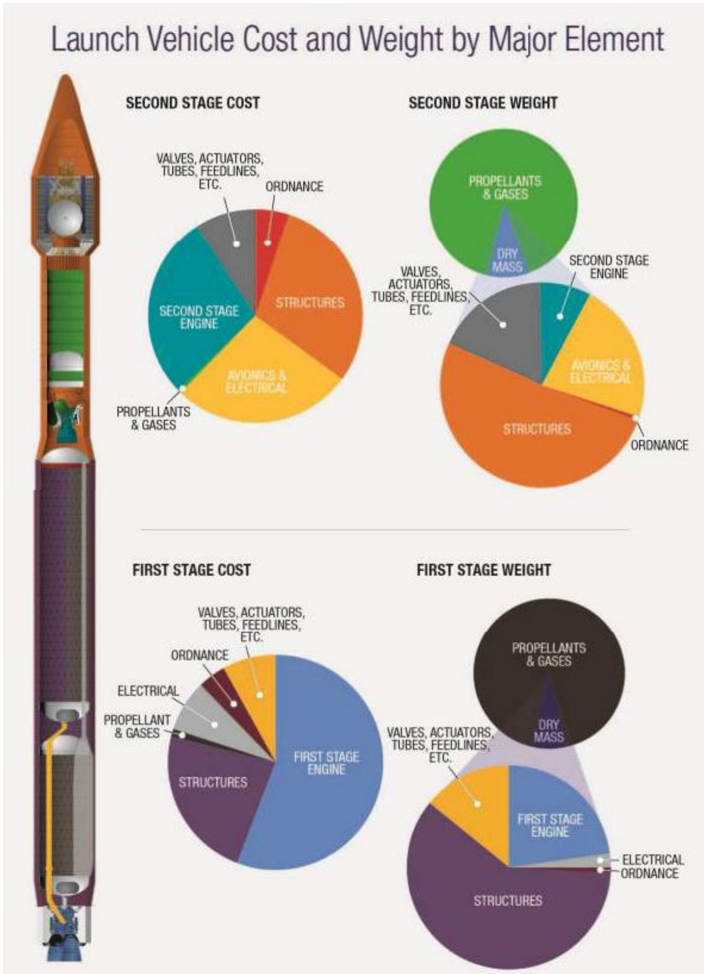

Figure 3.19 Atlas V Cost and Weight By Element [48]

A conversion table that has been reproduced in Appendix E. is used to convert from WYr to either USD, Euros, or Yen.

Second, the generic form of the equation is

$$C = a * M^x \tag{3.36}$$

Where

C is the cost in WYr;

a is a constant;

M is a primary characteristic of the item being costed; and

x is an exponential value.

Third, certain cost factors are applied to the generic costing equation to develop a normalized cost equation for the item being sized. There are a total of fourteen different cost factors, although not all fourteen are applied to every equation. What cost factors to use as well as the value to use for it depends on the specific CER and factors surrounding the item that is being costed. The factors are:

1. **f_0, the system engineering/integrator factor.**
   This factor is applied to CER used to find the total development or production cost and is not applied to individual elements of the vehicle. The value to use for this factor depends on whether it is being applied to development or production costs.

2. **f_1, the technical development status factor.**
   This factor is applied to development costs and indicates how new the project's concept and technology are. Values for this factor vary from 0.3 to 1.4.

3. **f_2, the technical quality factor.**
   Unlike other cost factors, the value for f_2 depends on what specifically is being sized. For example, for the development cost of an expendable ballistic or transfer stage the value of f_2 is found by dividing a ratio of actual masses and a ratio of reference masses in plots provided by the *Handbook* while the value for the development of a liquid-propellant engine with turbopumps is calculated based on the number of engine development and qualification firings.

4. **f_3, the team experience factor.**
   The team experience factor is used to modify the development cost based on the level of experience the team has with designing that type of item. It ranges from a low of 0.5 for teams with "superior experience" to a high of 1.4 for a team that has little to no prior experience with what is being designed.

5. **f_4, the learning cost reduction factor.**
   This factor is used on production and operation CERs to apply the effects of cost reduction for increased familiarization with the processes involved and making

those processes more efficient. This factor varies between 0.70 and 0.85, but only if there is a launch rate of at least five of the same type of vehicle per year. Otherwise this factor is set to 1.00.

6. **f_5, the refurbishment cost factor.**
   The refurbishment cost factor applies only to reusable launch vehicle components or systems and is based on a fraction of the theoretical cost of production of a new unit.

7. **f_6, the optimum development schedule factor.**
   f_6 applies the cost effect of putting in extra work to finish a project early or of keeping staff on longer than expected to complete work due to delays and/or rework. The value for f_6 is determined from a figure provided in the handbook based on the percentage ahead or behind schedule the project is where 100% is on time and results in a factor equal to 1.00.

8. **f_7, the program organization factor.**
   Costs begin to increase if a program is split up amongst several major contractors. This factor adjusts the development cost in these instances, and is calculated by taking the number of parallel major contractors and raising it to the 0.2 power.

9. **f_8, the country productivity factor.**
   The amount of time, and thus money, spent on a project can vary significantly from country to country for a variety of reasons. The cost factor f_8 adjusts the cost based on the location the work is being done in, and the value for the factor depends on the specific country the work is begin done in.

10. **f_9, the subcontractor cost factor.**
    Similar to f_7, costs begin to increase as subcontractors are used to handle different parts of a project. This factor is determined through interpolation of two different figures provided by the *Handbook* based on the number of subcontractors, their estimated profit margin, and their percentage share of the total work being performed.

11. **f_{10}, the cost engineering factor.**
    All CERs in the *Handbook* were developed using data from "business-as-usual" government contracts. These tend to be slow, expensive projects that frequently have cost overruns and focus on maximizing performance while paying little to no attention on how to minimize cost. As cost has become an increasingly important part of design a cost factor has been introduced to take cost engineering principles

into consideration. The value of this factor varies depending on how extensively cost engineering principles are pursued and whether the CER is a for development or production costs. If they are not considered, the factor is set to 1.00.

12. **f_{11}, the commercial venture factor.**
Private companies such as SpaceX and Blue Origin are not under the same governmental and customer restrictions and interference that the "business-as-usual" contractors have been, and as such the factor f_{11} was introduced to reduce the cost estimate generated by the CER to be more in line with what can be expected by a commercial venture. The value for this factor varies depending on whether the CER is for development, production, or operations costs.

13. **f_v, the launch vehicle type factor.**
This factor applies only to certain operations CERs and is used to factor in the change in handling for different vehicle types. The value to use is specific to the type of vehicle system. For example, liquid-propellant vehicles with cryogenic propellants require specialized equipment and procedures for handling propellants and thus has an f_v of 1.0, while liquid-propellant vehicles with storable propellants do not require these and have a factor of 0.80.

14. **f_c, the assembly and integration factor.**
There are several different ways to prepare a launch vehicle system for launch. Three methods are noted by the *Handbook*: vertical assembly and checkout on a launch pad; vertical assembly and checkout in an assembly building, then transportation to a launch pad; or horizontal assembly and checkout, then transportation to a launch pad. Values of 1.00, 0.85, and 0.70 are respectively assigned to these three methods.

Some CERs have other additional factors applied to them, such as the launch rate per year, but these are few and far between. See the *Handbook* [125] for additional details.

The CER equation for development of an expendable launch vehicle stage and its corresponding plot with normalized and original data may been seen in Equation (3.37) and Figure 3.20. The yellow diamonds are the original data while the black squares are the results after the cost factors have been applied to the CER.

$$H_{VE} = 98.5 * M^{0.555} * f_1 * f_2 * f_3 * f_8 * f_{10} * f_{11} \quad \textbf{(3.37)}$$

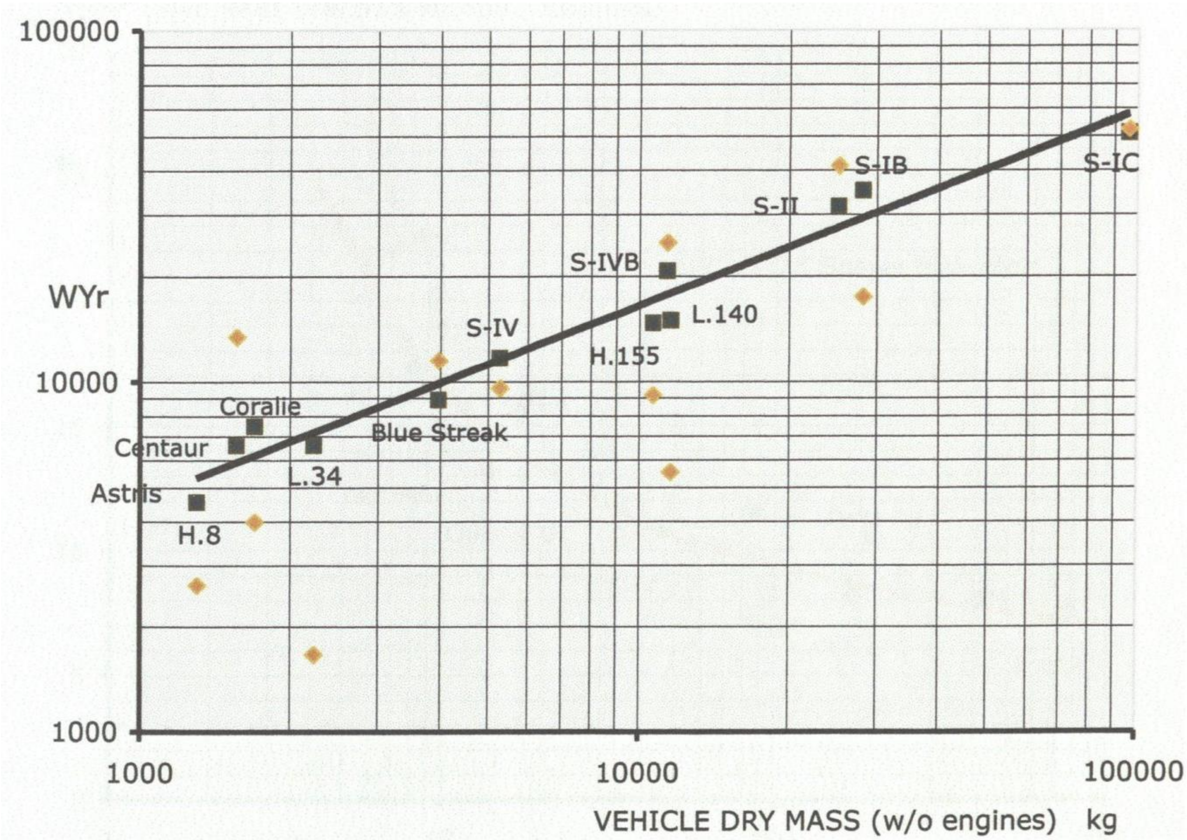

Figure 3.20 CER Plot for Sizing an Expendable Launch Vehicle Stage [125]

It is important to note three things about these CERs: first, while the *Handbook* provides specific values to use for some cost factors it provides a range of values for other factors, and it is up to the user to determine what to use. Two users many have slightly differing opinions on the value for the factor and thus may end up with slightly different cost values. Second, because TransCost uses actual cost data, development cost estimates already include overruns and are thus "15 to 20 % higher than the 'ideal cost' or typically proposal costs" [125]. In order to compare values generated from TransCost CERs to estimates from other tools either the TransCost CER results must be revised down or the other set of values revised up. Third, cost values are given in units of "Work-Year" (WYr), which is a custom unit developed D. E. Koelle to more accurately compare cost data between countries and different years. The handbook provides a table to convert from WYr to USD, Euros, or Yen. A copy of this table may be found in Appendix E.

The cost module was validated by coding up the CERs found in the *Handbook of Cost Engineering* [125] and testing them with the provided examples and values pulled from the plots used to generate the CER. As TRANSCOST is widely accepted in industry no additional verification was done.

3.13 Output Module

Once the vehicle is sized and its cost determined, the information is stored in a Python dictionary. If multiple vehicles are being sized in a single run, all vehicles deemed as feasible are stored in the dictionary and all vehicles are passed to the Output Module at once. The module extracts the vehicle's sizing and trajectory data then writes it to an Excel file for additional data reduction and plotting.

3.14 Modifications to the System

To demonstrate the flexibility of the sizing and broaden the types of launch vehicles the system could size, two modifications were made: a modification for the last stage of the launch vehicle to perform a partial burn to reach orbit and use the remainder of its propellant for orbital maneuvers; and a modification for a stage to be sized to perform a boostback recovery.

3.14.1 Partial Burn Modifications

It is sometimes desirable for the final stage of a launch vehicle to contain excess propellant to perform additional orbital maneuvers after reaching the parking orbit. These maneuvers could be used to put the payload into a higher orbit or, in the case of the third stage of the Saturn V for the Apollo missions, be used for escaping Earth's gravity well. This is referred to as a "partial burn" or "transfer" stage.

Sizing a vehicle where the final stage can be used for additional burns requires minor modifications to the sizing program: at the start of the Sizing Module, the $\Delta V_{partial\ \mathrm{burn}}$ requirement is added to the ΔV for the final stage of the vehicle. After that, the sizing process precedes the same as for a standard expendable vehicle. The only other changes are changes to the Input Module to accept the $\Delta V_{partial\ \mathrm{burn}}$ value and to the Output Module to include $\Delta V_{partial\ \mathrm{burn}}$ in the Excel file. Figure 3.21 contains the NS diagram for sizing a launch vehicle with a partial burn.

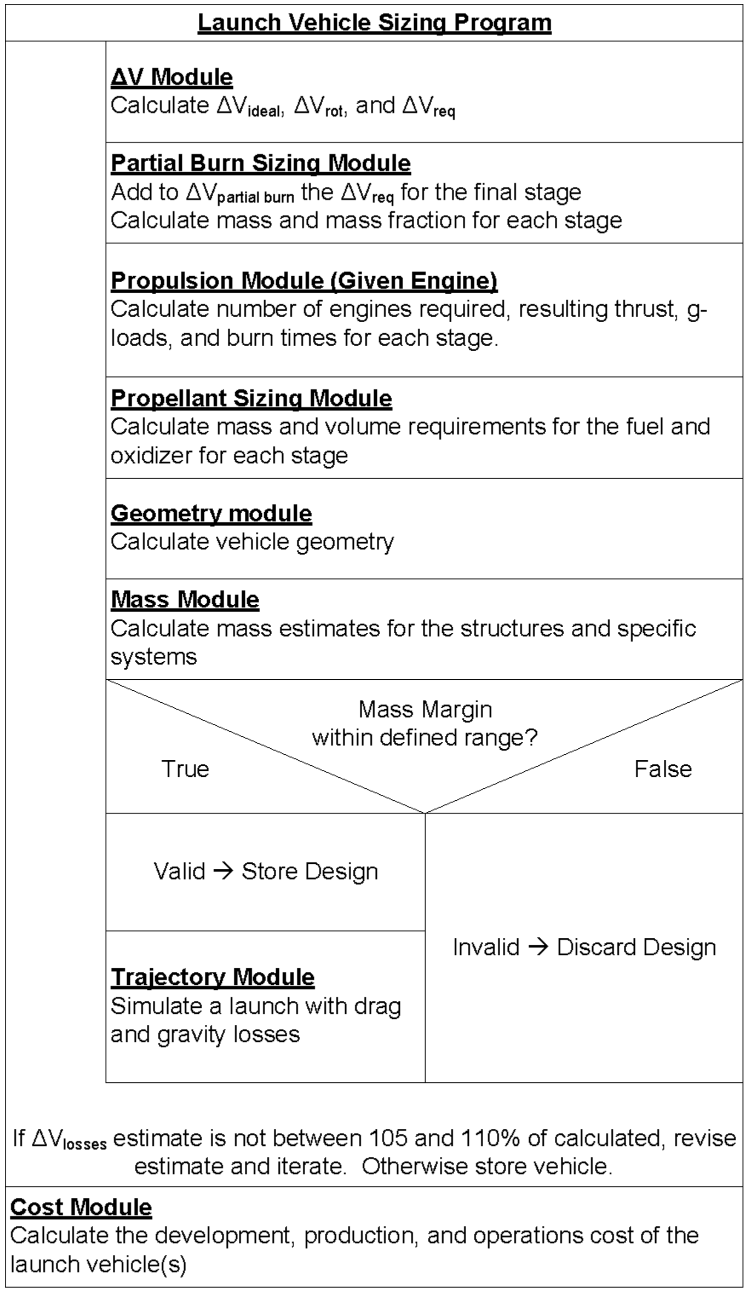

Figure 3.21 Partial Burn Sizing System NS Diagram

3.14.2 Boostback Reusability Modifications

The modification for sizing a vehicle with boostback reusability is slightly more complicated. In fact, the overwhelming majority of papers discussing the sizing of a boostback vehicle do not actually discuss how their sizing was done. Only Reference [126] explains their methodology: first, an expendable vehicle was sized. Then the additional hardware necessary for boostback returns was added to the structures and equipment mass of the vehicle. Next, a fraction of the propellant mass was reserved for the necessary boostback maneuvers. Finally, the payload capacity was reduced by the amount necessary for the launch vehicle to reach its required parking orbit with the propellant remaining for its ascent flight.

While this method used by reference [126] does work, it is desirable to design the vehicle with a specific payload capacity in mind rather than proceeding through a guess-and-check method of sizing an expendable vehicle and converting it to have boostback capability. Therefore the author undertook the task developing a methodology to size a boostback vehicle by starting with the desired payload capacity and working forward towards a vehicle configuration. This methodology is verified in the third case study described in Section 4.4.

It is not possible to analytically size a launch vehicle with partial or full boostback reusability with a method as described in Sections 3.6 or 3.14.1. However, by manipulating the fundamentals equations of launch vehicle design and numerically iterating them it is possible to size such a vehicle. To perform a boostback recovery, a stage must have additional ΔV to perform various burns to slow down and land. Consider the rocket equation and definitions for $m_{initial}$ and m_{final} for an expendable launch vehicle that were introduced in Equations (3.16), (3.6), and (3.11), respectively, and are reproduced below:

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

$$m_{initial} = m_{payload} + m_{propellant} + m_{se}$$

$$m_{final} = m_{payload} + m_{se}$$

From the rocket equation it can be seen that

$$\Delta V_{ascent} = g_o * I_{sp} * \ln\left(\frac{m_{initial,ascent}}{m_{final,ascent}}\right) \quad \textbf{(3.38)}$$

and

$$\Delta V_{boostback} = g_o * I_{sp} * \ln\left(\frac{m_{initial,boostback}}{m_{final,boostback}}\right) \quad \textbf{(3.39)}$$

For a boostback vehicle the definition for $m_{initial}$ and m_{final} are slightly different for the ascent and boostback phases of flight:

$$m_{initial,ascent} = m_{payload} + m_{propellant} + m_{se}$$

$$m_{initial,ascent} = m_{payload} + m_{propellant,ascent} + m_{propellant,boostback} + m_{se} \quad \textbf{(3.40)}$$

$$m_{final,ascent} = m_{payload} + m_{propellant,boostback} + m_{se} \quad \textbf{(3.41)}$$

$$m_{initial,boostback} = m_{propellant,boostback} + m_{se} \quad \textbf{(3.42)}$$

$$m_{final,boostback} = m_{se} \quad \textbf{(3.43)}$$

The value for $\Delta V_{boostback}$ cannot be directly added to $\Delta V_{stage,\ ascent}$ because of differing definitions of $m_{final,\ ascent}$ and $m_{initial,\ boostback}$ combined with the nonlinearity of the rocket equation: too many unknowns remain in the equations to solve analytically if one attempts to do so.

Thus, instead of attempting to directly add together the two ΔV values to obtain a ΔV_{stage}, a boostback stage can be sized via the process displayed in Figure 3.22:

1. A value for $\Delta V_{boostback}$ is assumed.

2. A fraction of $\Delta V_{boostback}$ is added to the $\Delta V_{ascent,\ required}$, the ΔV that the stage must produce during ascent to reach for the payload to reach its parking orbit.

3. The stage is sized as normal.

4. By rearranging the rocket equation and using the definitions of equations (3.42) and (3.43), the propellant mass required to perform the boostback burns, $m_{prop,\ boostback}$ is calculated.

5. $\Delta V_{ascent,\ actual}$ is calculated through the rocket equation and definitions for $m_{final,\ ascent}$ and $m_{initial,\ ascent}$.

6. A check is now performed to compare $\Delta V_{ascent,\ actual}$ to $\Delta V_{ascent,\ required}$. If $\Delta V_{ascent,\ actual}$ is greater than $\Delta V_{ascent,\ required}$ but not by more than some maximum percentage, then

the stage is sized and the sizing software proceeds to the next stage or module as required. The software produced for this thesis checks for a value not greater than 1%. It is important to note that while the amount that $\Delta V_{ascent,\ actual}$ exceeds $\Delta V_{ascent,\ required}$ is subjective, it is **required** that $\Delta V_{ascent,\ actual}$ is greater than $\Delta V_{ascent,\ required}$. If it were not the stage could not provide the necessary ΔV required to reach the parking orbit.

7. The remainder of the sizing process remains the effectively the same as sizing an expendable vehicle, although the Mass Module must now include MERs to include landing hardware such as landing gears, fins, thermal protection as needed, etc.

Assume $\Delta V_{boostback}$ estimate

> Add a fraction of $\Delta V_{boostback}$ to ΔV_{stage}
>
> $\Delta V_{ascent,\ required} = \Delta V_{stage} + f * \Delta V_{boostback}$
>
> Size the stage as done normally
>
> Calculate $m_{prop,\ boostback}$ via rocket equation
>
> $m_{final,\ boostback} = m_{se} + m_{prop,\ boostback}$
> ...
> $m_{prop,\ boostback} = m_{se} * e^{((\Delta V boostback\ /\ (goeIsp)\ -1)}$
>
> Calculate $\Delta V_{ascent,\ actual}$ via rocket equation
>
> $m_{final,\ ascent} = m_{se} + m_{payload} + m_{prop,\ boostback}$
> ...
> $\Delta V_{ascent,\ actual} = g_{oe} I_{sp} * \ln(m_{initial} / (m_{final,\ ascent}))$

If the ratio of $\Delta V_{ascent,\ actual}$ to $\Delta V_{ascent,\ required}$ is not between 100 to 101%, revise $\Delta V_{ascent,\ required}$ and iterate

Continue sizing the rest of the sizing process as normal

Figure 3.22 Boostback Sizing Process

This sizing process hinges on the value for $\Delta V_{boostback}$ as being a valid estimate. In order to confirm this value, the sizing process must be taken a step further with a trajectory simulation. The process works as indicated in Figure 3.23: an ascent trajectory simulation is first performed with the stage, and then any boostback stages are run through a boostback trajectory simulation using the position and motion data at the point of staging as initial conditions. If the landing is unsuccessful then $\Delta V_{boostback}$ is too low; alternatively, if the landing was successful but the stage had an excess amount of propellant remaining then $\Delta V_{boostback}$ was too high. In either case, $\Delta V_{boostback}$ is adjusted appropriately and the sizing process returns back to the Sizing Module to resize the vehicle.

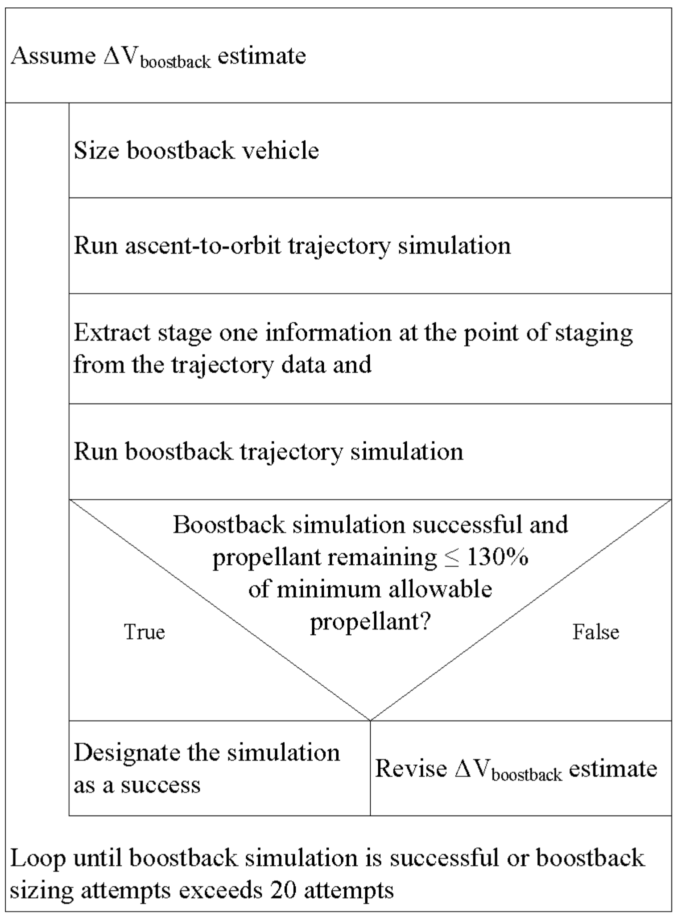

Figure 3.23 Boostback Sizing Trajectory Verification

This ultimately leads to the NS diagram presented in Figure 3.24.

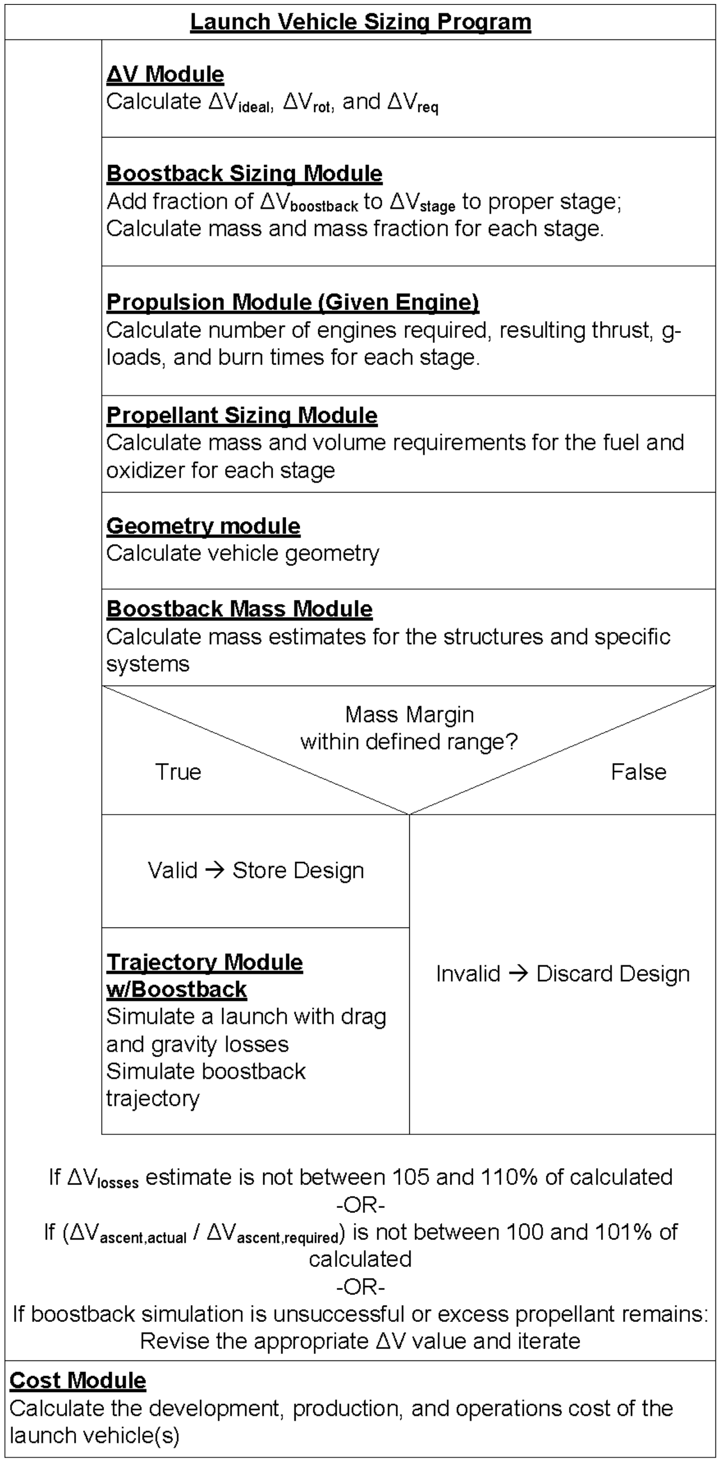

Figure 3.24 Boostback Sizing System NS Diagram

Boostback Trajectory Module

As discussed above, a boostback trajectory simulation is required in order to verify the assumed value for $\Delta V_{boostback}$ is sufficiently large enough that the launch vehicle can return to the launch site. A return-to-launch-site (RTLS) simulation is significantly more complex than an ascent-to-orbit trajectory, and thus it important to first examine the path a boostback vehicle follows and the various actions performed by the vehicle prior to explaining how the simulation runs.

Prior to SpaceX and Blue Origin's efforts in boostback returns, the most extensively studied RTLS trajectories were a Space Shuttle abort scenario and the trajectory studies on the McDonnell Douglas Tossback Booster [41], [127]. The Shuttle's never-used abort scenario, presented in Figure 3.25, would have been used in the event of a non-catastrophic engine failure, loss of cabin pressure, or other problem that would have resulted in the need to abort the mission but where control of the Shuttle could be maintained. In order to abort and return to the launch site, the Shuttle would begin a powered pitch-maneuver in order to re-orient itself to face towards the launch site, reverse the direction of its downrange velocity such that the Shuttle was traveling towards the launch site instead of away, and position the external propellant tank between the orbiter and the ground. The external tank would then be discarded, and the Shuttle would perform an unpowered glide to the runway near the launch site [127].

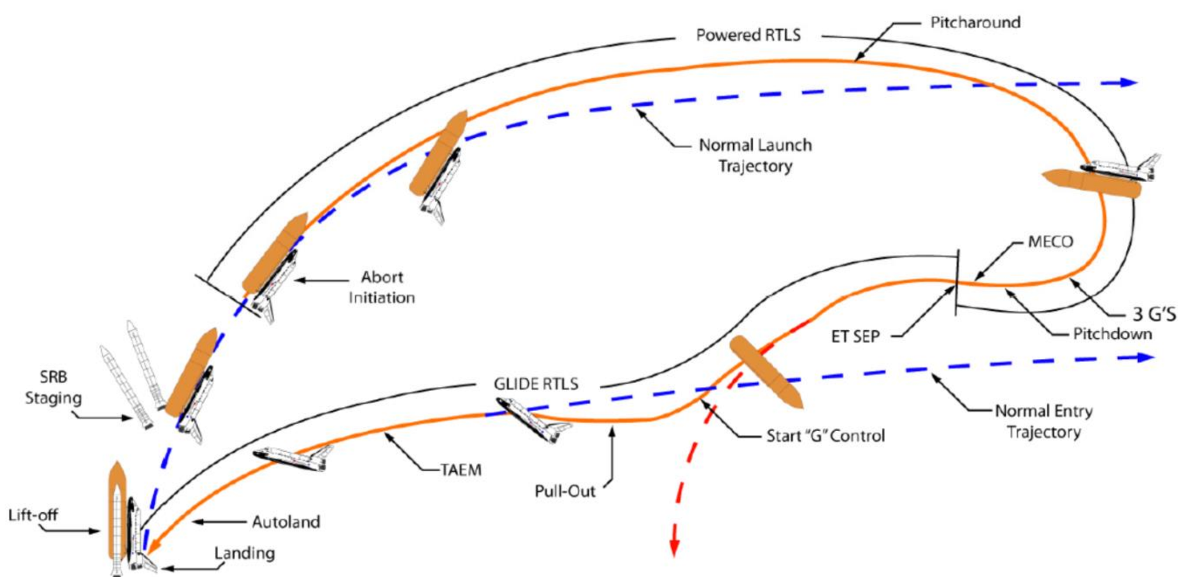

Figure 3.25 Space Shuttle RTLS Abort Scenario [127]

The McDonnell Douglas Tossback Booster followed a similar-style flight path. As indicated in Figure 3.26, the first stage would pitch and re-ignite its engines after staging. This burn would continue until the resulting trajectory shape for an unpowered, parabolic flight would end at the launch location. The booster then followed this trajectory until it approached the landing site. Once the vehicle was at a pre-determined altitude, a landing burn would slow the velocity such that its velocity was near zero when it reached the ground [41], [127].

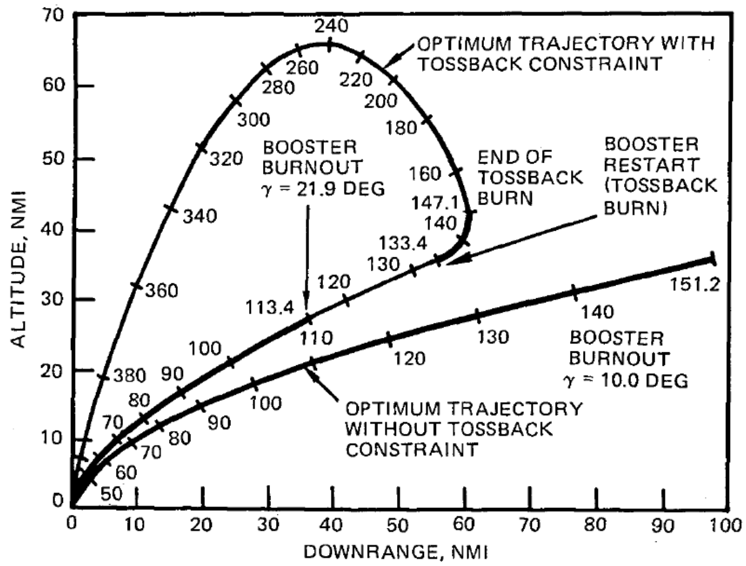

Figure 3.26 McDonnell Douglas Tossback Booster Trajectory [41]

SpaceX does not officially publish the flight paths used in its missions, but the data required to reconstruct the Falcon 9's trajectory is available in their webcasts. There is some noise in the broadcast data, but it is sufficient enough to reconstruct a general flight path for the eight missions shown in Figure 3.27. Seven of these included landings: four that followed a RTLS trajectory, and three of which landed on a drone ship downrange [128]. The four RTLS trajectories follow paths similar to the McDonnell Douglas Tossback Booster, although they staged at a significantly higher γ.

By analyzing these trajectories, it is determined that a RTLS trajectory simulation must contain the following features:

1. **Perform a boostback burn.**
   The simulation needs to pitch the first stage towards the launch location after staging occurs and burn until its velocity vector is such that its momentum will carry it back to land.

2. **Determine when to begin a re-entry burn and when to end it.**
   The launch vehicle needs to reduce its velocity prior to entering the dense atmosphere to avoid potential structural failure and aerothermal concerns as well as increase the time it takes to pass through the lower atmosphere. While the vehicles sized by this software do not possess the wide base of the boostback designs presented in Section 2.1.1, the vehicles will still experience significant drag during its descent in the lower atmosphere that can be used to further reduce velocity without expending additional propellant. By reducing the velocity of the vehicle prior to reaching dense

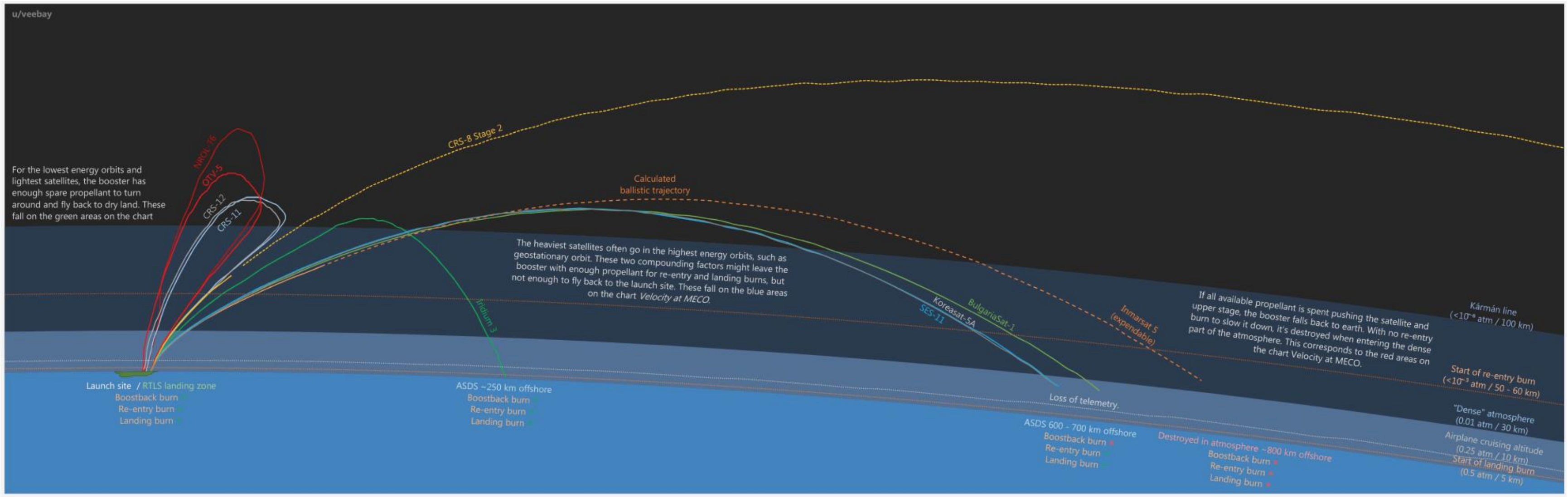

Figure 3.27 Reconstructed Falcon 9 Stage 1 Trajectories [128]

Table 3.1 Starting and Ending Data for Five Falcon 9 Re-entry Burns

Mission	Start velocity (m/s)	End velocity (m/s)	Start alt (km)	End Alt (km)	Time Start (s)	Time end (s)
NROL-76 [129]	1391	736	71.3	39.2	428	456
OTV-5 [130]	1290	747	53.7	33.9	394	414
CRS-12 [131]	1211	911	52.4	37.6	368	382
CRS-11 [132]	1246	1033	50.2	38.1	370	381
Iridium 3 [133]	1223	873	48.2	36.3	345	359

atmosphere, the vehicle will remaining in it for a longer period of time and the resulting increase in total drag experience will reduce the propellant mass required to land.

3. **Calculate when to begin the landing burn and at what thrust level.**
   If the landing burn begins to early or at too high a thrust level, the launch vehicle's velocity vector will flip directions and it will begin to ascend prior to reaching the ground. Should it begin burning too late or without enough thrust, it will crash into the ground.

To perform all of these functions precisely requires multiple nested, iterating simulations for each phase of flight. The demands of such an approached exceed the computational limits and time constraints of this thesis. Therefore, the Boostback Trajectory Module uses the following assumptions:

1. **The pitch angle during the boostback burn is fixed at 190° to the horizon.**
   Through testing of the Boostback Trajectory Module, it is found that a pitch angle of 190° reshaped the trajectory such the vehicle followed a path similar to the Falcon 9 RTLS trajectories. Other pitches and a dynamically changing pitch angle during the boostback burn may provide a more optimum trajectory, but a study of optimized RTLS trajectories is beyond the scope of this thesis.

2. **The re-entry burn begins at a time and altitude with sufficient thrust such that the vehicle's altitude and velocity at the end of the re-entry burn are approximately 38 km and 800 m/s.**
   Table 3.1 presents the data at the start and end of the re-entry burn for five of the Falcon 9's stage one recoveries, including the four RTLS trajectories. In each of these missions, the starting altitude of the re-entry burn varies by as much as 23 km, but the deviation between final altitudes is less than 5 km. The ending velocities vary between a low of 736 m/s and a high of 1,033 m/s. An ending velocity of 800 m/s provides an approximate result close to the average.

3. **The landing burn begins at an altitude of 12 km.**
   Determining the exact altitude at which to begin a landing burn requires a series of simulations in which a landing burn is attempt at every possible altitude and velocity during the post re-entry burn descent. Beginning the landing burn at an altitude of 12 km is comparable to the Falcon 9 landing burns while being high enough that the estimated propellant required to perform the landing is larger than what should actually be required, providing a safe margin-of-error in the design.

4. **The throttle for the landing burn may be any percentage between the minimum and maximum throttle level, but it remains constant the entire landing burn.** While the throttle level may vary during an actual landing burn, the vehicle's landing capability is validated if it can land at a constant thrust level when coupled with the next assumption.

5. **A successful landing simulation permits the vehicle's final altitude and velocity to be as high as 0.1 km and 90 m/s.** While the 90 m/s appears extraordinarily high, it is important to remember the above assumption that the throttle level is fixed the entire duration of the landing burn. Starting the landing burn at a high throttle and dialing back to the minimum thrust level would result in a significantly lower final velocity. However, such complex dynamic modeling is beyond the scope of this thesis.

An NS diagram depicting the programming logic for Boostback Trajectory Module is presented in Figure 3.28. It is important to note that allowing the system to determine the points to begin the re-entry and landing burns, terminate the boostback, re-entry, and landing burns, and calculate the average throttle level to use for landing creates a computationally heavy process. While an ascent-to-orbit simulation may take one to three seconds to complete, a single RTLS simulation takes between fifteen and thirty seconds to run. If the landing process is unsuccessful, the system will revise the $\Delta V_{boostback}$ estimate and iterate. To prevent excessive run times this iteration is capped at a maximum of ten attempts. If the ascent-to-orbit simulation is successful each time but the boostback simulation iterates for the maximum number of attempts, this runs in a minimum run time of 3.67 minutes. If the ascent-to-orbit simulation fails due to missing the target orbit and the entire sizing process must iterate to correct the ΔV_{losses} estimate, a total run time of between fifteen and thirty minutes for a single configuration is likely.

C_D Submodule Updates

In addition to the creation of the Boostback Trajectory Module, a modification to the C_D Submodule is required to simulate the RTLS trajectory of a Falcon 9-like first stage. The Falcon 9 uses four grid fins which are used to steer the first stage during its descent in the dense atmosphere in addition to providing increased drag. Among the characteristics of grid fins listed by Fleeman is an advantage of "high control effectiveness at low subsonic and high supersonic Mach numbers" while having a low hinge moment and short chord length, and a disadvantage of high drag at both transonic and low supersonic Mach numbers [122]. However, in the case of a boostback trajectory, all of these are advantages: the high control effectiveness combined with short size means the grid fins provide a small penalty during ascent while exceeding at control during the return flight, and the high drag reduces the velocity without the use of propellants.

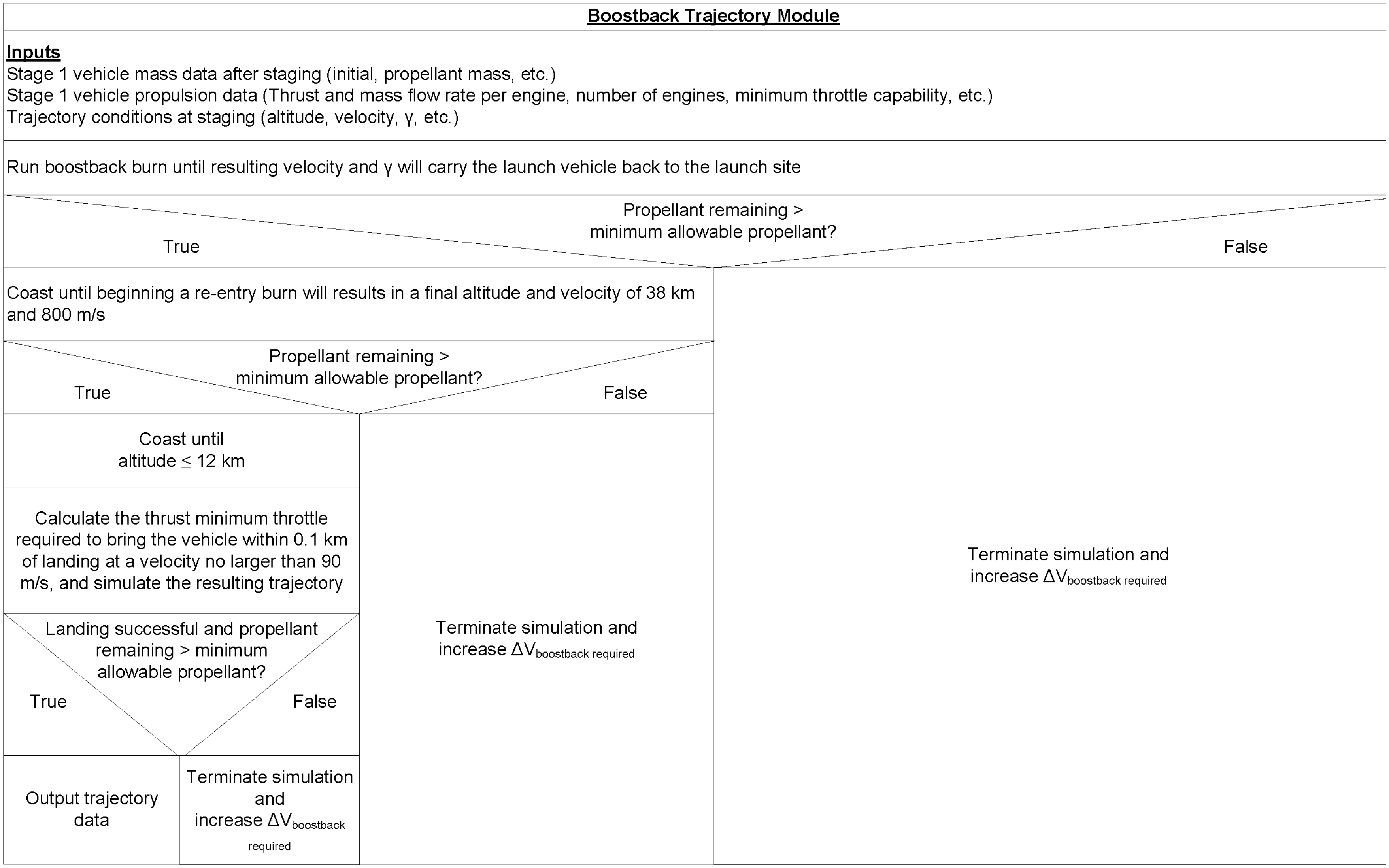

Figure 3.28 NS Diagram of the Boostback Trajectory Module

In order to model the drag by the grid fins during descent, the C_D Submodule is updated using a method developed by one of the MAE 4350 and 4351 senior design teams. Each grid fin is broken up into two halves as depicted by Figure 3.29. Each half is treated as a series of individual planar fins whose drag can be approximated using methods provided by Fleeman [122]. The C_D for each section is then added together via superposition to provide the total C_D for a single grid fin. That is multiplied by the number of grid fins and added to the body's C_{D0}, producing the total vehicle C_{D0} .

It is important to note that there is a small amount of error generated in the C_{D0} for the grid fins as this method ignores interference effects between the individual planar fins and breaks down as the Mach number approaches one. However, it is sufficient for producing the first order approximation for C_D0 that is required for the boostback simulation.

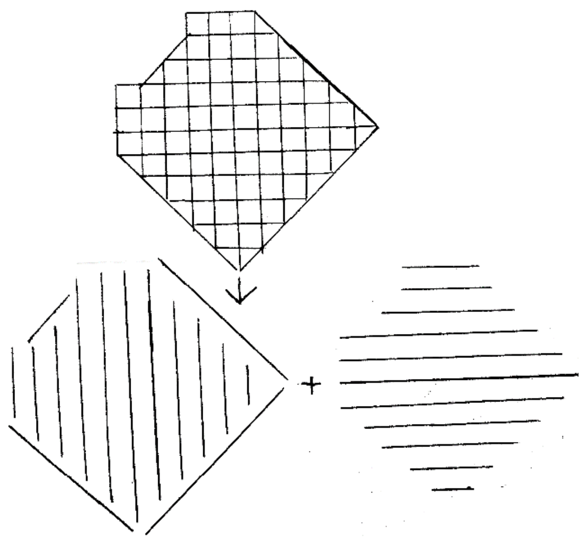

Figure 3.29 Sectioned Grid Fins [134]

3.15 Conclusions

With the modules coded, tested using information from the reference used to develop it, and assembled together to function as a single sizing program, it was time to begin running case studies to verify the program as a whole.

Chapter 4. Case Studies

While each module of the program was tested as the system developed and confirmed to be working properly, it must be used as a system in order to verify its overall functionality to correctly size launch vehicles. To this end, three verification studies have been executed to test its capabilities:

1. Fully expendable stages, used entirely for ascent to parking orbit: **Gemini Launch Vehicle**;

2. Fully expendable system, with the final stage containing additional propellant for orbital transfer burns: **Saturn V**; and

3. Partially boostback reusable, with stages used entirely for ascent to parking orbit and a first stage boostback recovery: **Falcon 9 Full Thrust, reusable**.
    - Due to the proprietary nature of the Falcon 9, some data is unknown and must either be assumed or calculated based on known information. To that end, the software is used to generate a vehicle similar to the expendable version of the Falcon 9 Full Thrust first. If the software can size the expendable version with moderate accuracy, it is also capable of sizing the reusable version.

Each verification study begins with a point check. Each point check takes input variables specific to verification study's launch vehicle. The resulting output vehicle is compared to the actual vehicle to determine the accuracy of the system. Once the software's output for each capability has been verified, an iterative sweep is run over a range of values for variables. The goal of this sweep is to test the system's ability to size and output a large number of different vehicles at once. Results from selected inputs are plotted in order to identify trend lines which can be used to identify optimal designs as well as determine the sensitivity of change to the input parameters. The variables used and ranges for each are discussed in each section.

4.1 Gemini Launch Vehicle

Like *Project Mercury* (1958 – 1963) before it, the *Gemini Program* (1961 – 1966) was a test platform used to develop the knowledge and technology necessary to reach the Moon. Among the many completed goals of the program was to test the effect of extended zero-gravity stays on the human body, learn how to perform spacewalk, change and control of the orbit of a spacecraft, and determine how to dock a spacecraft to another object in orbit [135].

The Gemini Launch Vehicle (GLV), shown in Figure 4.1, was a Titan II ICBM that was modified to meet the needs of the Gemini Program. The vehicle was used on all twelve of the

Gemini missions and was successful each time [136]. The modifications made from the baseline Titan II were primary to meet NASA's "man rating" requirements. Changes include the introduction of several backup, a new guidance system, and removal of unnecessary equipment. Additionally, a new structure was added to the top of the second stage in order to mate the Gemini capsule to the launch vehicle [137].

Figure 4.1. Gemini Launch Vehicle at Liftoff [138]

The GLV is selected as the first case study due to the plethora of information available on the vehicle and on its missions. Through the vehicle and trajectory data derived from the Gemini Program Mission Reports, the created software's ability to size fully expendable launch vehicles is verified.

4.1.1 Point Check

Input Data

Table 4.1 through Table 4.3 contain the required input information based on the data available for the GLV:

Table 4.1 GLV Sizing Input Data

Variable name	**Value**		**Units**
	Stage 1	**Stage 2**	
Launch Latitude [87]	28.474		deg.
Launch Altitude [87]	200		m
Parking Orbit Altitude [136], [137]	185		km
Payload mass [87]	3,200		kg
Payload diameter [136], [137]	3.05		m
Payload length [136], [137]	5.74		m
ΔV losses*	1,676.40		m/s
Number of stages [136], [137]	2		-
$\Delta V_{\text{fraction}}$ [136], [137]	39.58%	60.42%	-
π_{se} [136], [137]	2.92%	6.97%	-
$I_{sp,\,mean}$ [139], [140]	258	316	s
T_{eng} [87]	956,320	444,800	N
Minimum throttle percentage [139], [140]	100%	100%	-
Mass per engine [139], [140]	713	565	kg
Engine length [139], [140], [141]	3.13	2.794	m
Engine nozzle exit area [139], [140]	1.450	2.378	m^2
Gap between engines	-	-	m
Fuel-to-oxidizer ratio [139], [140]	0.526	0.559	-
$\rho_{oxidizer}$	1,450	1,450	kg/m^3
ρ_{fuel}	903	903	kg/m^3
T/W_{min}**	1.1	0.5	-
T/W_{max}**	2	2	-
Gravity turn start altitude [136], [137]	950		m
Max permitted g-load [136], [137]	8		-
Max permitted dynamic pressure [136], [137]	36,006		Pa

* Based on the standard assumptions for ΔV_{losses}
**Input range selected based on actual T/W and standard recommended values

Table 4.2. GLV Initial Flight Plan*

Start time (s)	**$\dot{\gamma}$ (deg./s)**
0.0000	0.0000
23.7640**	-0.1488
27.3528**	-0.4113
29.8875**	-0.5770
32.4231	-0.7150
43.8374	-0.9144
54.8340	-0.8181
64.5595	-0.6910
75.7617	-0.5936
86.3275	-0.5429
97.1032	-0.4716
106.3983	-0.3850
115.0580	-0.3699
123.5061	-0.2455
131.9518	-0.2586
139.9754	-0.2308
147.1539	-0.1521
153.9090	-0.1800
161.9311	-0.1538
169.9528	-0.1914
179.6641	-0.2214
192.7541	-0.2053
205.8436	-0.1360
216.3980	-0.1697
226.1088	-0.1708
234.5529	-0.1160
245.1068	-0.1114
256.0827	-0.1210
267.9032	-0.0855
279.7227	-0.1097
289.0099	-0.0652
301.2510	-0.0762
311.8038	-0.0677
320.6681	-0.0380
341.3484	-0.0367

* Modified trajectory based off of the Gemini IV mission trajectory [87]

Table 4.3. GLV Cost Factors

Variable	Value	Justification
Average number of launches per year	4	12 flights over 3 years [87]
Total number of vehicles to produce	12	12 total flights [87]
Annual vehicle production	4	Set to match average number of launches per year
Number of flights per vehicle	1	Vehicle is fully expendable
$f_{0,\ development}$	1.0816	Fixed input value based on number of stages
$f_{0,production}$	1.03	GLV was based on existing technology but the program was new and complicated [87]
f_1	0.6	GLV is a derivative of the Titan II [87], [136], [137]
f_3	0.7	Team had prior experience with Project Mercury [87]
$f_{4,operations}$	1	Number of flights per year is less than 5
Development schedule	100%	GLV was on-time [87]
f_7	1.246	Three major contractors [142]
f_8	1	Country is the US
f_9	1	-
f_{10}	1	Cost engineering was not a factor
f_{11}	1	Government project
f_v	0.8	Expendable, liquid-propellant vehicle with storable propellant [87], [136], [137]
f_c	0.85	Stages assembled vertically and transported to launch pad [87]

Results

As described in the User's Guide in Appendix F. all of the required input variables are loaded into an input file, see Figure 4.2. This input file is passed to the software which then calls and runs to sizing modules in the appropriate order. Once the sizing is complete, a trajectory simulation verifies the vehicle's ability to complete the desired mission. If the vehicle successfully completes its mission all of the data is exported to an Excel file. If the simulation fails, the software checks the source of this error and attempts to make corrections to the necessary variables. Possible sources of error include an input trajectory path that causes the calculated vehicle to miss the target altitude, a final γ which will not maintain the correct orbit shape and cannot be corrected with minor adjustments, and a high or low assumption for ΔV_{losses}. In order to avoid any infinite loops, the simulation attempts to correct these errors a maximum of twenty times. The total time to run the GLV point check and output all of the data was 3.13 seconds.

```
def inputFileGLV():

    #Define constant
    g_oe = 9.81


    h_Launch = 200
    latitude = 28.4740
    alt_orbit = 185 * 1000
    m_payload = 3200
    d_payload = 3.05
    l_payload = 5.74
    delta_V_losses = 1.6764 * 1000
    n_stages = 2
    delta_V_fraction = np.array([0, 0.3919187, 0.6080813
    pi_se = np.array([0, 0.029222467, 0.06968694], dtype
    Isp_mean = np.array([0, 258, 316], dtype=np.float)
    T_eng = np.array([0, 956320, 444800], dtype=np.float
    Throttle = np.array([0, 1.0, 1.0], dtype=np.float)
    m_eng = np.array([0, 713.0, 565.0], dtype=np.float)
    l_eng = np.array([0, 3.13, 2.794], dtype=np.float)
    A_e = np.array([0, 1.4503, 2.3778], dtype=np.float)
```

Figure 4.2 Snippet of the GLV Input File

A summary of results for the initial point check of the GLV is presented in Table 4.4 and Table 4.5, and a simplified model of the sized vehicle is shown in Figure 4.3. The software predicted slightly below the actual values for the required ΔV and mass components while achieving an orbit slightly higher than the desired parking orbit. While it is not expected for a conceptual design tool to predict final design values exactly, it is unusual that all of the predicted values are below the actual values.

Table 4.4 GLV Point Check Sizing Results Summary

Variable	Calculated Value	Actual Value	Percent Error
$\Delta V_{stage\ 1}$ (m/s)	3,552.31	3,633.00	-2.22%
$\Delta V_{stage\ 2}$ (m/s)	5,511.59	5,566.00	-0.98%
$m_{initial,\ stage\ 1}$ (kg)**	148,849.71	155,866.00	-4.50%
$m_{initial,\ stage\ 2}$ (kg)**	32,226.71	32,960.00	-2.22%
$m_{se,\ stage\ 1}$ (kg)	4,349.76	4,554.00	-4.48%
$m_{se,\ stage\ 2}$ (kg)	2,245.78	2,295.00	-2.14%
$m_{propellant,\ stage\ 1}$ (kg)	112,273.24	118,352.00	-5.14%
$m_{propellant,\ stage\ 2}$ (kg)	26,780.93	27,465.00	-2.49%
$m_{fuel,\ stage\ 1}$ (kg)	38,714.15	40,703.85	-4.89%
$m_{fuel,\ stage\ 2}$ (kg)	9,598.90	9,996.83	-3.98%
$m_{oxidizer,\ stage\ 1}$ (kg)	73,559.09	78,047.17	-5.75%
$m_{oxidizer,\ stage\ 2}$ (kg)	17,182.03	17,469.03	-1.64%
Stage 1 engine count	2	2	0.00%
Stage 2 engine count	1	1	0.00%
Stage 1 diameter (m)	3.05	3.05	0.00%
Stage 2 diameter (m)	3.05	3.05	0.00%
Total vehicle length (m)	29.94	33.22	-9.89%
Stage 1 length (m)	17.13	21.60	-20.68%
Stage 2 length (m)	7.06	8.80	-19.75%
Gimbal angle required to maintain stability (deg.)	2.57	-	-
Max gimbal angle available (deg.)	-	± 5.00	-

*All GLV data comes from [87], [136], and [137]
**Based on actual propellant and structure/equipment mass values with a 3,200 kg payload

Table 4.5 GLV Point Check Trajectory Results Summary

Variable	Calculated Value
Final altitude (m)	187,945.8
Altitude target (m)	185,000.0
Altitude surplus/deficit (m)	2,945.8
Percent error in altitude	1.59%
γ_{final} (deg.)	3.60
Final velocity (m/s)	7,262.1
Velocity gain from launch location (m/s)	407.9
Total vehicle velocity (m/s)	7,670.0
Velocity required to maintain orbit achieved (m/s)	7793.6
Velocity surplus/deficit for orbit achieved (m/s)	-123.6
Percent error in velocity for orbit achieved	-1.59%
Estimated ΔV_{losses} (m/s)	1,676
Actual ΔV_{losses} (m/s)	1,543
Percent error in ΔV_{losses}	8.68%

Further investigation into the GLV reveals that the payload mass used in this initial point check is low: the GLV carried payloads with a higher mass than the 3,200 kg value that is commonly cited as the payload capacity. Adjusting the input value to 3,400 kg and rerunning the software provides the results shown in Table 4.6 and the simplified model in Figure 4.4. The percent error for the vehicle's mass is now positive with GLOW within 1.5% of the actual value. The vehicle's geometry remains short of the actual values. Potential sources of this error include insufficiently accounting for the space between the end of the first stage and start of the second stage's engines and not accounting for additional volume required to package the vehicle's electronics. Additionally, the fault may lay with a definition error. Where the first stage ends and second stage begins is not always consistently defined with launch vehicles. In some cases the length of the first stage refers to the length from the end of the nozzles of first stage engines to the end of the second stage engine nozzles, as is done in this thesis, while in other cases the first stage ends at the top of the interstage fairing and the second stage's length begins instead of at the end of the second stage engine nozzle(s). The interstage fairing is a structure the houses the next stage's engine nozzles and falls away as part of the depleted stage during the staging process. The sizing software developed during this thesis defines stage length with the first definition (engine nozzle of one stage to the engine nozzle of the next stage). While it would not explain all of the error with the GLV's geometry, switching definitions would lengthen the sized first stage and greatly reduce the error for that stage.

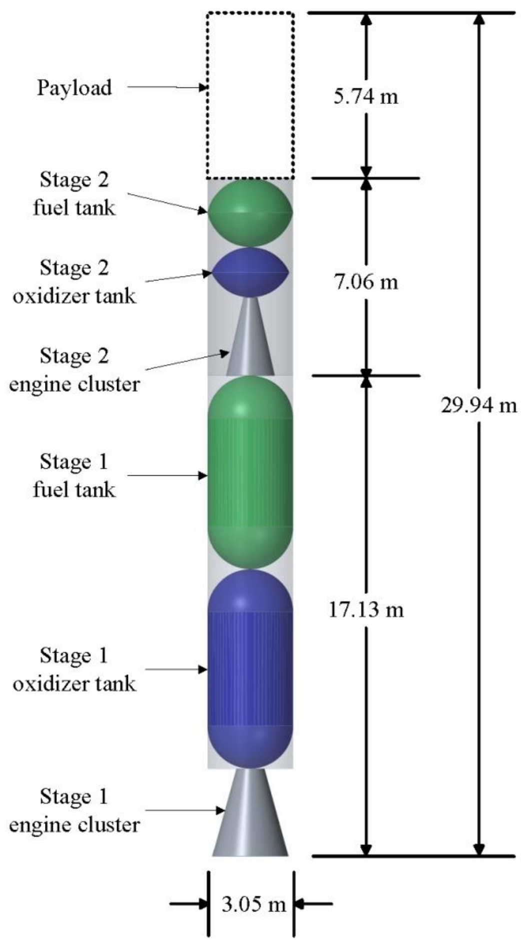

Figure 4.3 Simplified Model of the Sized GLV With a Payload Mass of 3,200 kg

It is worth noting that the vehicle's design fails if the mass margin check from the *Systems Mass Check Module* is enabled. Normally, this would indicate the input values for π_{se} are lower than technologically feasible for the vehicle's configuration. The results of this check, presented in Table 4.7, provide a positive mass margin of above 30% for the second stage, a mass margin of less than 1% for the first stage, and a negative mass margin for the complete vehicle. When the result from the Systems Mass Check Module is compared to the actual mass, it comes out

673 kg higher than the real value. This emphasizes the warning provided during the introduction of this module that the designer needs to be cautious with the secondary calculation of structure and equipment mass generated by the MERs and use it as a guideline, not a rule.

Table 4.6 GLV Point Sizing Results Summary for a Payload Mass of 3,400 kg

Variable	Calculated Value	Actual Value	Percent Error
$\Delta V_{stage\ 1}$ (m/s)	3,552.31	3,633.00	-2.22%
$\Delta V_{stage\ 2}$ (m/s)	5,511.59	5,566.00	-0.98%
$m_{initial,\ stage\ 1}$ (kg)	158,152.81	156,066.00	1.34%
$m_{initial,\ stage\ 2}$ (kg)	34,240.88	33,160.00	3.26%
$m_{se,\ stage\ 1}$ (kg)	4,621.62	4,554.00	1.48%
$m_{se,\ stage\ 2}$ (kg)	2,386.14	2,295.00	3.97%
$m_{propellant,\ stage\ 1}$ (kg)	119,290.32	118,352.00	0.79%
$m_{propellant,\ stage\ 2}$ (kg)	28,454.74	27,465.00	3.60%
$m_{fuel,\ stage\ 1}$ (kg)	41,133.78	40,703.85	1.06%
$m_{fuel,\ stage\ 2}$ (kg)	10,198.83	9,996.83	2.02%
$m_{oxidizer,\ stage\ 1}$ (kg)	78,156.54	78,047.17	0.14%
$m_{oxidizer,\ stage\ 2}$ (kg)	18,255.91	17,469.03	4.50%
Stage 1 engine count	2	2	0.00%
Stage 2 engine count	1	1	0.00%
Stage 1 diameter (m)	3.05	3.05	0.00%
Stage 2 diameter (m)	3.05	3.05	0.00%
Total vehicle length (m)	30.93	33.22	-6.90%
Stage 1 length (m)	17.93	21.60	-16.97%
Stage 2 length (m)	7.25	8.80	-17.56%
Gimbal angle required to maintain stability (deg.)	2.48	-	-
Max gimbal angle available (deg.)	-	± 5.00	-

When analyzing the results of the trajectory simulation, it is found that the vehicle's velocity falls short by 123.6 m/s and γ remains high at 3.6^{o}. However, minor adjustments to the vehicle's pitch rate earlier in the path would correct this.

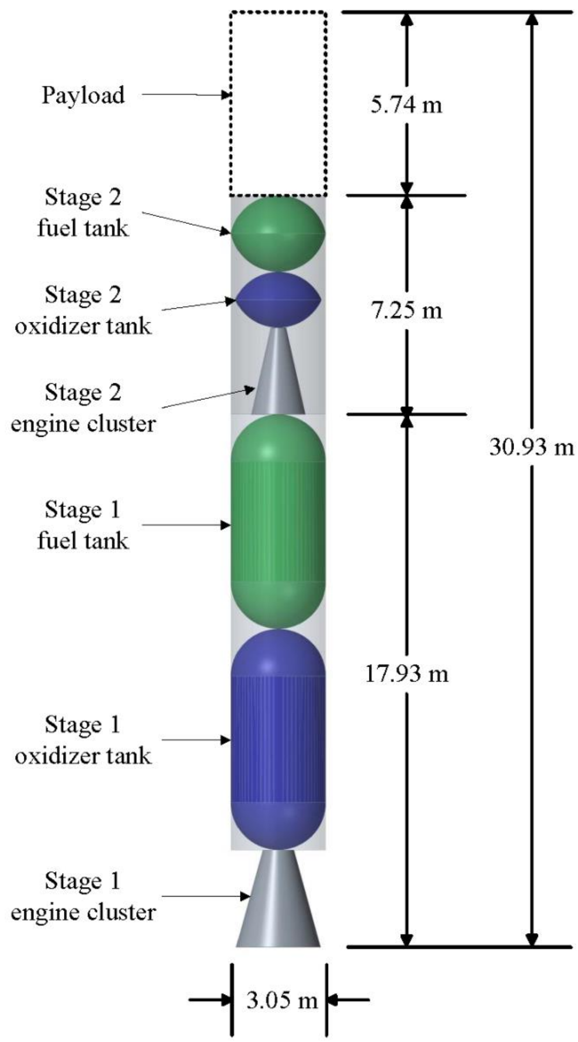

Figure 4.4 Simplified Model of the Sized GLV With a Payload Mass of 3,400 kg

There are several potential problems with analyzing the cost for the GLV. First, TRANSCOST assumes that a launch vehicle is being developed as a clean-sheet design which was not the case with the GLV. Second, based on the context of references discussing the cost of the GLV, the procurement cost of the Titan II ICBM is often lumped into the development cost associated with converting it to be man-rated. Third, costs of various aspects of the Gemini Program are often grouped together in the reported value which difficult to discern what portion of costs went to what. To attempt to alieve these problems, the calculated development cost is compared to the development cost of the Titan II ICBM, the calculated production cost is compared to the production cost of a Titan II ICBM, and the calculated combination of development and production costs is compared to the total cost for the GLV on the Gemini Program as reported in 1962. The results of this compare are shown in Table 4.8. The calculated total development and total production costs are significantly different that the actual values for the Titan II development and production. However, the sum of the Cost Module's total development and total production costs are comparable to the total cost of the GLV to the Gemini Program.

Table 4.7 Comparison of calculated structure and equipment masses

Actual structures and equipment mass estimate (kg)	6,849.00
Initial structures and equipment mass estimate (kg)	6,595.54
Mass check structures and equipment mass estimate (kg)	7,522.02
Calculated mass margin (kg)	-926.48
Percent mass margin	-14.05%

Table 4.8 GLV Cost Comparison Based on the 3,200 kg Payload Sizing Results

Variable	Calculated Value (WYr)	Actual Value (WYr)	Calculated Value (in $M, 1975 dollars)	Actual Value (in $M, 1975 dollars)	Percent Error
Total Development Cost* [143]	20,579.614	44,319.328	$1,224.49	$2,637.00	-53.57%
Total Production Cost** [144]	298.435	53.076	$17.76	$3.16	462.28%
Total Development Plus Production Cost***	20,878.050	18,152.863	$1,242.24	$1,080.10	15.01%

*Based on the development cost of the Titan II ICBM provided in reference [143]
**Based on the production cost of the Titan II ICBM provided in reference [144]
***Based on the reported total cost of the GLV to the Gemini Program in 1962 and converted to 1975 dollars [145]

With the successful sizing of the GLV completed, the sizing software is verified for sizing fully expendable launch vehicles.

4.1.2 Iterative Sweep

With the software's ability to size a launch vehicle similar to the GLV verified, the system was then set to size a large number of vehicles over a range of input values in order test the software's ability to size a variety of based in a single run. These values, shown in Table 4.9, include a varied π_{se}, $\Delta V_{fraction}$, and payload mass. As discussed in Section 4.3.1, the GLV would not be returned as a valid design with the mass margin check enabled as the secondary calculation of the structures and equipment mass for the input π_{se}, it is bypassed for this iterative sweep.

These inputs result in a total of 14,976 potential configurations. Out of these, 5,131 are returned as potentially viable vehicles; the remaining configurations were discarded due to their inability to successfully simulate the mission or the combination of input variables resulting in a vehicle with negative mass for one or more stages. Utilizing a desktop computer with an Intel Core i7 5820k and 32 GB of RAM in conjunction with using multithread processing tools in Python, it took the software 57 minutes to run and output the data for this range of outputs.

Table 4.9. GLV Sweep Inputs

Variable	Minimum Value	Maximum Value	Interval
$\pi_{se,\ stage\ 1}$	0.92%	6.92%	0.5%
$\pi_{se,\ stage\ 2}$	3.97%	9.97%	0.5%
$\Delta V_{fraction,\ stage\ 1}$*	25%	75%	2%
Payload mass (kg)	2,400	5,400	1000

*$\Delta V_{fraction,\ stage\ 2}$ = 100% - $\Delta V_{fraction,\ stage\ 1}$

This data is used to generate the carpet plot show in Figure 4.5. The plot is for a constant $\pi_{se,\ stage\ 2}$ = 6.97% and payload mass of 3,400 kg, and the results are excluding configurations whose total vehicle mass margin was less than -1,500 kg. The plot consists of constant contours of $\pi_{se,\ stage\ 1}$ and $\Delta V_{fraction,\ stage\ 1}$ plotted against GLOW. These form a grid where each point represents a vehicle sized by the system. Any combination of $\pi_{se,\ stage\ 1}$ and $\Delta V_{fraction,\ stage\ 1}$ inside of this grid results in a valid vehicle configuration when run through the sizing process.

As is immediately evident, the plots for a constant $\pi_{se,\ stage\ 1}$ of 1.9%, 3.9%, and 4.9% display unusual behavior as $\Delta V_{fraction,\ stage\ 1}$ decreases. Further analysis of the results reveals that the sizing program adjusted the trajectory in order to reach the parking orbit, and while making these adjustments the actual ΔV_{losses} exceeded the original estimate causing the software to adjust the initial estimate and resize the vehicle. The end result is a valid vehicle design and trajectory, but not necessarily one that follows the most optimal trajectory. Additionally, despite the provided range of inputs there is a hard stop at $\Delta V_{fraction}$ equals 43%. Continued analysis shows that configurations beyond the $\Delta V_{fraction}$ upper bound resulted in vehicles that exceeded that maximum g-load due to the GLV's inability to throttle its engines. While that's not a concern today, exceeding the g-load limit was a significant design problem at the time. Design combinations at the lower limit of $\pi_{se,\ stage\ 1}$ dropped sufficiently low enough in the mass margin that they are excluded from consideration.

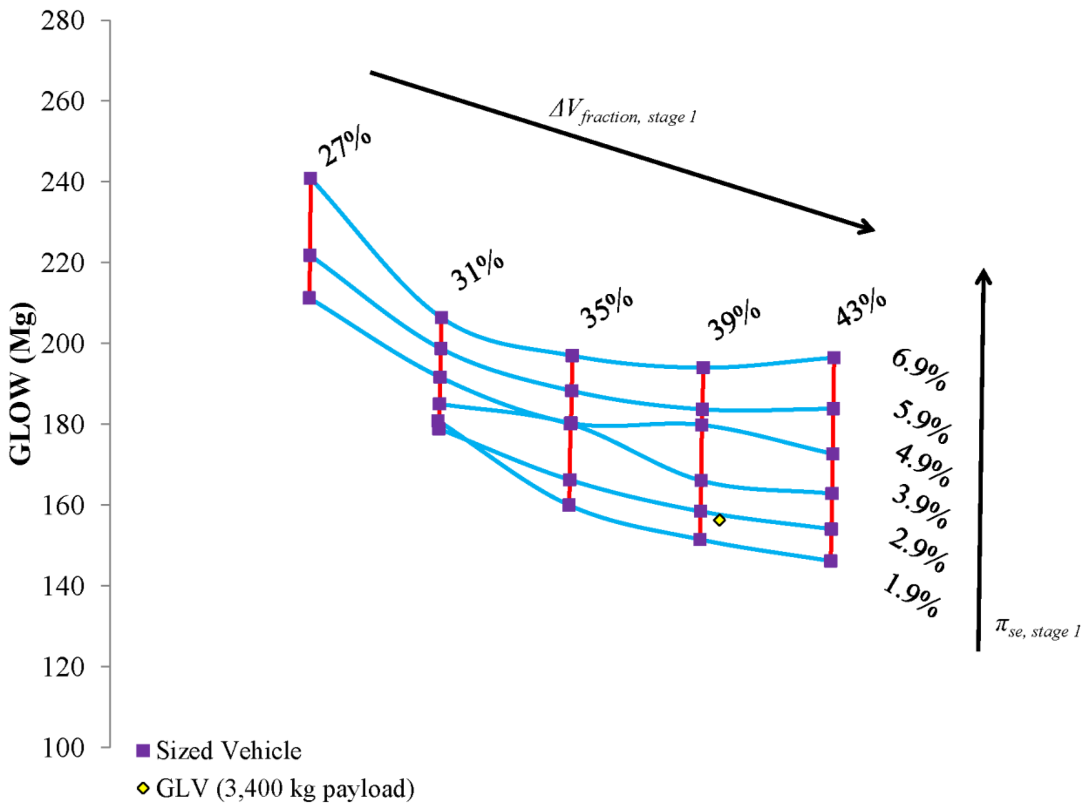

Figure 4.5. $\pi_{se,\ stage\ 1}$ and $\Delta V_{fraction,\ stage\ 1}$ vs. GLOW

In order to better visualize the trend lines, configurations which resulted in an increased value for ΔV_{losses} are rerun through the system to determine the GLOW that would occur with a trajectory path better shaped for the specific configuration. The data points from this secondary run replace the originals in the carpet plot, resulting in the smoothed carpet plot provided in Figure 4.6. The GLV falls near the lower bound of feasible design with the minimum GLOW.

This reveals that the GLV pushed technological limits of the time and the optimal design available for the mission.

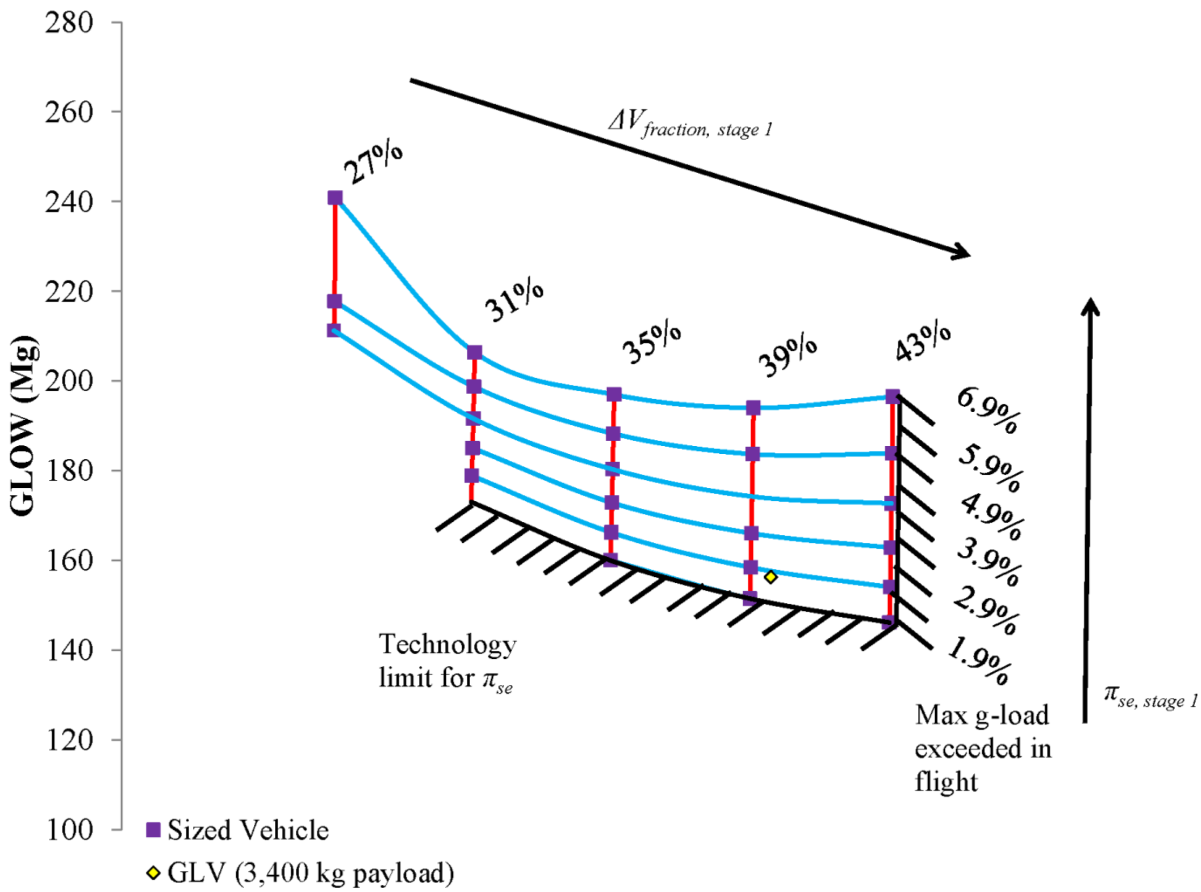

Figure 4.6 Smoothed $\pi_{se,\ stage\ 1}$ and $\Delta V_{fraction,\ stage\ 1}$ vs. GLOW with limitations

4.2 Saturn V

The Saturn V was designed with the intent of transporting a manned lunar spacecraft into Earth's orbit. Unlike most other launch vehicles of the time, the Saturn V was not a collection of converted ICBMs/IRBMs but was instead a clean-sheet design meant expressly for this purpose. In addition, the Saturn V is unique compared to other launch vehicles of the time because the third stage was designed to complete the ascent to orbit trajectory and then place the Apollo spacecraft into the Lunar Transfer Orbit.

The Saturn V was selected at the second case study because of this history as a clean-sheet design, its third stage's use as a combination ascent-to-orbit and transfer vehicle, and the availability of both design and cost data in the Apollo Mission Reports and sources such as the *Handbook of Cost Engineering* [125], [146].

It is important to note here that the software does not perform the sizing of fins such at the ones on the base of Saturn V. Aerodynamic control surfaces on launch vehicles were phased out early on in the history of launch vehicles as thrust gimballing technology developed. As noted by Wernher von Braun himself, the fins on the Saturn V were not added for stability and control purposes. Instead, the fins were added to delay loss of vehicle in one specific failure scenario in which one of the engines would get locked into its maximum gimbal angle. This would result in the Saturn V entering an uncontrollable pitch and break apart shortly afterwards due to the lateral aerodynamic forces. Adding the fins at the base of the Saturn V would delay vehicle failure long enough for the escape tower to pull the astronauts to safety [147], [148].

Figure 4.7 Saturn V at Launch [149]

4.2.1 Point Check

Input Data

Table 4.10 through Table 4.12 contain the required input information based on the data available for the Saturn V:

Table 4.10 Saturn V Sizing Input Data

Variable name	Value Stage 1	Stage 2	Stage 3	Units
Launch Latitude	28.474			deg.
Launch Altitude	200			m
Parking Orbit Altitude	185.2			km
Payload mass**	49,735			kg
Payload diameter	6.614			m
Payload length	25.66			m
ΔV losses***	1,800			m/s
Number of stages	3			-
$\Delta V_{fraction}$(4)	37.18%	52.32%	10.50%	-
$\Delta V_{transfer\ burn}$	-	-	3,160	m/s
π_{se}(4)	4.50%	5.60%	7.00%	-
$I_{sp,\ mean}$	260	424	427	s
T_{eng}	6,805,779	1,023,091	920,782	N
Minimum throttle percentage	100%	100%	100%	-
Mass per engine	8,391	1,579	1,579	kg
Engine length	5.79	3.35	3.35	m
Engine nozzle exit area	9.79	3.00	3.00	m^2
Gap between engines	-	-	-	m
Fuel-to-oxidizer ratio	0.441	0.182	0.182	-
$\rho_{oxidizer}$	1,138.4	1,138.4	1,138.4	kg/m^3
ρ_{fuel}	802.7	70.5	70.5	kg/m^3
T/W_{min}(5)	1.05	0.7	0.5	-
T/W_{max}(5)	1.3	1	0.7	-
Gravity turn start altitude	1300			m
Max permitted g-load	4.25			-
Max permitted dynamic pressure	36,000			Pa

*All non-assumption data comes directly or was calculated from references [146], [150], [151], [152], [153]
**Despite some sources doing so, the payload mass used in this case study does not include the Saturn IVB's dry mass and propellant remaining after reaching LEO.
*** Based on the standard assumptions for ΔV_{losses} and revised during initial testing for the case study
(4)Calculated based on masses provided in "Saturn V Launch Vehicle Flight Evaluation Report-AS-506 Apollo 11 Mission" [146]
(5)Input range selected based on actual T/W and standard recommended values

Table 4.11. Saturn V Initial Flight Plan*

Start time (s)	$\dot{\gamma}$ (deg./s)
0	0
0	-0.2780
80	-0.4461
135	-0.2735
165	-0.505
185	-0.0075
320	-0.0665
460	-0.1145
480	-0.0736
550	-0.1835
570	-0.0882
640	-0.0251

*Modified trajectory based off of the trajectory from "Saturn V Launch Vehicle Flight Evaluation Report-AS-506 Apollo 11 Mission" and a Apollo 11 trajectory simulation by Robert Braeunig [146], [154]

Results

A summary of results for the initial point check of the Saturn V is presented in Table 4.13 and Table 4.14, and a simplified model of the vehicle is shown in Figure 4.8. The total time to run the software for the point check was 1.64 seconds. The software created a vehicle nearly equivalent in launch mass for each stage. The structures and equipment mass for all stages is slightly deficient, indicating that the value used for π_{se} was too low. The calculated fuel mass is also notably different than the actual mass for the third stage. Since the J-2 engines are used on both the second and third stages, and the second stage fuel masses are within a much closer margin, it is implied that the fuel-to-oxidizer ratio utilized by the third stage is different than what is used by the second. As with the GLV, the calculated stage lengths are also under the actual length of the vehicle.

The trajectory simulation of the sized vehicle results in the third stage being placed in an orbit of 186.4 km at 8,603 m/s, higher than the 7,794.5 m/s required to maintain this orbit. The ΔV_{losses} estimate was 9.31% higher than the calculated losses, within the upper bound what was is desired.

It is important to clarify definitions from the cost module before analyzing the results. The *Handbook of Cost Engineering* calculates the total production and development cost by summing the results for a stage and applying a systems engineering factor to determine the entire project cost. Thus, the sum of the engine and stage development or production costs will **not** be the same as the total cost.

Table 4.12. Saturn V Cost Factors

Variable	Value	Justification
Average number of launches per year	1.857	13 flights over 7 years
Total number of vehicles to produce	13	13 total flights
Annual vehicle production	1.857	-
Number of flights per vehicle	1	Vehicle is fully expendable
$f_{0,\ development}$	1.124864	Fixed value based on number of stages
$f_{0,production}$	1.03	Saturn V was based on existing technology but the program was new and complicated
f_1	1.3	Saturn V was a complex system with many new components. Not set to the max value of 1.3 due to experience from Mercury and Gemini
f_3	0.9	Team had prior experience with Project Mercury and Gemini Program, but increased over the value used for the GLV due to the vehicle’s increased complexity
$f_{4,operations}$	1	Number of flights per year is less than 5
Development schedule	110%	Saturn V was behind schedule
f_7	1.3195	Three major contractors [142]
f_8	1	Country is the US
f_9	1	-
f_{10}	1	Cost engineering was not a factor
f_{11}	1	Government project
f_v	1.0	Expendable, liquid-propellant vehicle with cryogenic propellant for two of the stages
f_c	0.85	Stages assembled vertically and transported to launch pad

Table 4.13 Saturn V Point Check Sizing Results Summary*

Variable	Calculated Value	Actual Value	Percent Error
$\Delta V_{stage\ 1}$ (m/s)	3,415.60	3,342.25	2.19%
$\Delta V_{stage\ 2}$ (m/s)	4,807.60	4,704.38	2.19%
$\Delta V_{stage\ 3}$ (m/s)**	4,124.70	4,110.35	0.35%
$m_{initial,\ stage\ 1}$ (kg)	2,916,335.56	2,938,315.00	-0.75%
$m_{initial,\ stage\ 2}$ (kg)	633,068.15	654,420.00	-3.26%
$m_{initial,\ stage\ 3}$ (kg)	163,837.62	170,793.00	-4.07%
$m_{se,\ stage\ 1}$ (kg)	131,235.10	132,890.00	-1.25%
$m_{se,\ stage\ 2}$ (kg)	35,451.80	36,729.00	-3.48%
$m_{se,\ stage\ 3}$ (kg)	11,468.60	12,024.00	-4.62%
$m_{propellant,\ stage\ 1}$ (kg)	2,152,032.00	2,145,798.00	0.29%
$m_{propellant,\ stage\ 2}$ (kg)	433,778.70	443,235.00	-2.13%
$m_{propellant,\ stage\ 3}$ (kg)	102,633.98	107,095.00	-4.17%
$m_{fuel,\ stage\ 1}$ (kg)	658,113.86	646,319.00	1.82%
$m_{fuel,\ stage\ 2}$ (kg)	66,735.19	71,720.00	-6.95%
$m_{fuel,\ stage\ 3}$ (kg)	15,789.84	19,780.00	-20.17%
$m_{oxidizer,\ stage\ 1}$ (kg)	1,493,918.45	1,499,479.00	-0.37%
$m_{oxidizer,\ stage\ 2}$ (kg)	367,043.50	371,515.00	-1.20%
$m_{oxidizer,\ stage\ 3}$ (kg)	86,844.14	87,315.00	-0.54%
Stage 1 engine count	5	5	0.00%
Stage 2 engine count	5	5	0.00%
Stage 3 engine count	1	1	0.00%
Stage 1 diameter (m)	10.05	10.06	-0.08%
Stage 2 diameter (m)	10.05	10.06	-0.08%
Stage 3 diameter (m)	6.61	6.61	0.00%
Total vehicle length (m)	102.13	110.64	-7.69%
Stage 1 length (m)	37.36	42.06	-11.17%
Stage 2 length (m)	24.04	24.84	-3.22%
Stage 3 length (m)	15.06	18.07	-16.66%
Gimbal angle required to maintain stability (deg.)	1.83	-	-
Max gimbal angle available (deg.)	-	± 6.00	-

*All non-assumption data comes directly or was calculated from references [146], [150], [151], [152]
**Includes ΔV for Lunar Transfer Orbit

Table 4.14 Saturn V Point Check Trajectory Results Summary

Variable	Calculated Value
Final altitude (m)	186,457.4
Altitude target (m)	185,200.0
Altitude surplus/deficit (m)	1,257.4
Percent error in altitude	0.68%
γ_{final} (deg.)	0
Final velocity (m/s)	8,195.0
Velocity gain from launch location (m/s)	407.9
Total vehicle velocity (m/s)	8,603.9
Velocity required to maintain orbit achieved (m/s)	7,794.5
Velocity surplus/deficit for orbit achieved (m/s)	808
Percent error in velocity for orbit achieved	10.37%
Estimated ΔV_{losses} (m/s)	1,800
Actual ΔV_{losses} (m/s)	1,647
Percent error in ΔV_{losses} (m/s)	9.31%

Initial review of the cost results provide mixed results. The total production costs of the engines and of the stages are relatively close to the actuals with an error less than 1.75% and 4.50%, respectively. With the exception of the third stage, all of the calculated development costs run higher than their actual counterparts. Overall, the calculated total development cost and total production cost with 17.79% and 12.78%, respectively, above the actual value, well within the ±25% margin recommended by the *Handbook of Cost Engineering* [125].

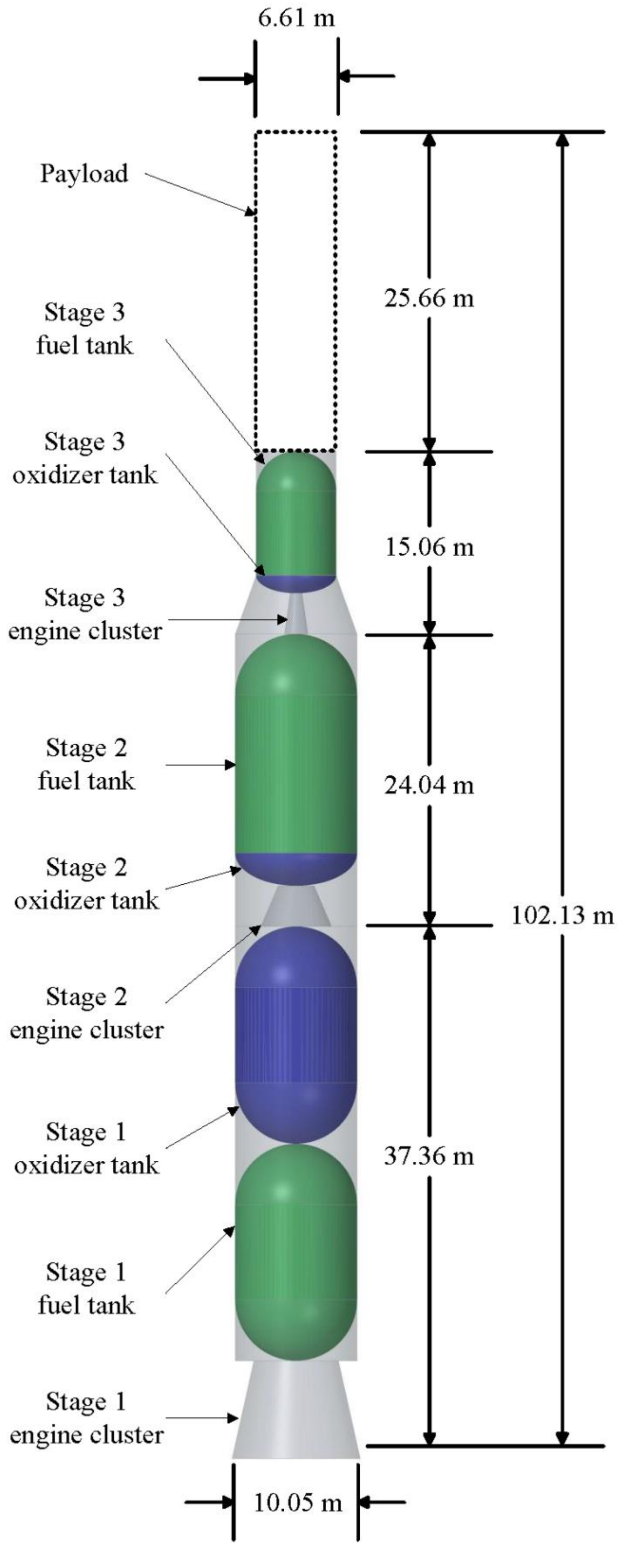

Figure 4.8 Simplified Model of the Sized Saturn V

Table 4.15 Saturn V Cost Results

Variable		Calculate Value (WYr)	Actual Value (WYr)	Calculate Value (in $M, 2015 dollars)	Actual Value (in $M, 2015 dollars)	Percent Error
Engine Development Cost	Stage 1	24,882.67	19,240.00	$8,387.95	$6,485.80	29.33%
	Stage 2	11,160.66	10,099.00	$3,762.26	$3,404.37	10.51%
	Stage 3	11,160.66	10,099.00	$3,762.26	$3,404.37	10.51%
	Sum	47,204.00	39,438.00	$15,912.47	$13,294.55	19.69%
Stage Development Cost	Stage 1	62,395.31	48,565.00	$21,033.46	$16,371.26	28.48%
	Stage 2	40,584.27	30,553.00	$13,680.96	$10,299.42	32.83%
	Stage 3	17,937.22	20,639.00	$6,046.64	$6,957.41	-13.09%
	Sum	120,916.80	99,757.00	$40,761.05	$33,628.08	21.21%
Total Development Cost		258,868.04	219,764.79	$87,264.42	$74,082.71	17.79%
Engine Production Cost	Stage 1	985.27	980.50	$332.13	$330.53	0.49%
	Stage 2	612.38	624.00	$206.43	$210.35	-1.86%
	Stage 3	161.81	124.80	$54.55	$42.07	29.66%
	Sum	1,459.46	1,729.30	$593.12	$582.95	1.74%
Stage Production Cost	Stage 1	1,003.74	722.20	$338.36	$243.45	38.98%
	Stage 2	729.69	887.90	$245.98	$299.31	-17.82%
	Stage 3	398.62	431.00	$134.37	$145.29	-7.51%
	Sum	2,132.04	2,041.10	$718.71	$688.05	4.46%
Total Production Cost		4,252.35	2,175.44	$1,433.47	$733.34	12.78%

Data from references [125] and [155]. Data from [125] are normalized values used to develop the CER used by the *Handbook of Cost Engineering*.

Table 4.15 Saturn V Cost Results (cont.)

Variable	Calculate Value (WYr)	Actual Value (WYr)	Calculate Value (in $M, 2015 dollars)	Actual Value (in $M, 2015 dollars)	Percent Error
Prelaunch Costs	1,761.87	1,400.00	$593.93	$471.94	25.85%
Flight Operations Cost	16.05	-	$5.41	UNKNOWN	-
Direct Operations Cost	1,796.21	-	$605.50	UNKNOWN	-
Cost per Flight	6,099.16	4,280.00	$2,056.03	$1,442.79	42.50%

Data from references [125] and [155]. Data from [125] are normalized values used to develop the CER used by the *Handbook of Cost Engineering*.

4.2.2 Iterative Sweep

Since the Saturn V is a three stage variable with a final stage that is used for orbital maneuvers, a range of values for π_{se} for first two stages, another set of π_{se} ranges for the third stage, and a range of $\Delta V_{fractions}$ for the first two stages are varied for the iterative sweep as shown in Table 4.16. The range of $\Delta V_{fractions}$ is set such that the third stage would contribute between 4% and 44% of the energy required to reach the parking orbit. As before, as with the GLV iterative sweep, the mass margin check was bypassed in order to view and filter out results manually.

These inputs result in a total of 48,600 potential configurations. Out of these, 42,900 are returned as potentially viable vehicles. Utilizing a desktop computer with an Intel Core i7 5820k and 32 GB of RAM in conjunction with using multithread processing tools in Python, it took the software 1194 minutes (19.9 hours) to run and output the data for this range of outputs.

Two sets of analysis are performed on this data set. The first compares the $\Delta V_{fraction}$ for stages one and two vs. GLOW. For this analysis the payload mass is fixed at 49,735 kg, $\pi_{se,\ stage\ 1}$ at 4.5%, $\pi_{se,\ stage\ 2}$ at 5.6%, and $\pi_{se,\ stage\ 3}$ at 7.0% so the data may be compared to the Saturn V. Additionally, vehicles with that returned a negative value for the mass margin check are removed from consideration. The interval of plotted data for $\Delta V_{fraction,\ stage\ 1}$ and $\Delta V_{fraction,\ stage\ 2}$ has also been reduced in order to better visualize the plot. The second plot holds the payload mass at 49,735 kg as well as the $\Delta V_{fraction}$ for each stage at 38%, 52%, and 10% while plotting $\pi_{se,\ stage\ 1}$ and $\pi_{se,\ stage\ 2}$ against GLOW to visually how a Saturn V like vehicle's size would change with improvements in technology. In order to visually the results in a 2D solution space, two plots are generated in which the value for $\pi_{se,\ stage\ 3}$ is held at 5% in the first and 7% in the second. Vehicles that returned a negative value for the mass margin check are not filtered here, but are instead indicated in the solution space.

Table 4.16. Saturn V Sweep Inputs

Variable	Minimum Value	Maximum Value	Interval
$\pi_{se,\ stage\ 1}$	2.5%	6.5%	1%
$\pi_{se,\ stage\ 2}$	1.6%	9.6%	1%
$\pi_{se,\ stage\ 3}$	5.0%	7.0%	2%
$\Delta V_{fraction,\ stage\ 1}$*	22%	40%	2%
$\Delta V_{fraction,\ stage\ 2}$*	34%	56%	2%
$m_{payload}$	29,735	69,735	20,000

*$\Delta V_{fraction,\ stage\ 3}$ = 100% - ($\Delta V_{fraction,\ stage\ 1}$ + $\Delta V_{fraction,\ stage\ 1}$)

The carpet plot for the first analysis is shown in Figure 4.9. As with the GLV, several of the sized vehicles' GLOW display erratic shifts the further away from the actual values they are. Detailed review of the data reveals this occurred for the same reason as it did in the GLV iterative sweep: the input value for ΔV_{losses} was revised for that particular configuration in order to successfully complete its mission with the provided trajectory path. Smoothing the plot provides the carpet plot in Figure 4.10.

The range of values for $\Delta V_{fraction}$ run was not large enough to reveal hard limitations, but a distinct reduction in GLOW leveling in GLOW is identified as the $\Delta V_{fraction,\ stage\ 2}$ approach 54%. In this region, continued shifts of ΔV to from the second stage to the third with a constant $\Delta V_{fraction,\ stage\ 1}$ provide a margin change in GLOW. If the ΔV for stage 2 is held constant and remaining ΔV is traded between the first and third stages a significant reduction in GLOW is obtained. However, this reduction in GLOW is not as straightforward as it appears. Continued analysis of the data reveals that as the change in $\Delta V_{fraction}$ reduces overall GLOW and either increases or decreases individual stage mass, the number of engines required to meet the minimum T/W also changes. As the number of engines changes, it is possible that the constant π_{se} assumption will fail due to the increase in mass the stages' propulsion system. This is does necessarily mean the designs are not feasible, but simply that more refined analysis techniques are required. This limitation is applied to the carpet plot in Figure 4.11. Inside the region where the assumption holds, it can be seen that the Saturn V is close to the region of minimum GLOW.

It is important to note here that results such as these emphasize the important of this type of conceptual design sizing and analysis. A quick calculation for the $\Delta V_{fraction}$ which resulted in the minimum GLOW would have produced a result which may not hold to the original assumptions. By generating solutions spaces such as the carpet plots in this research project and analyzing them to understand why the trend lines exist as well as how underlying assumptions may change provide the designer with the ability to not only correctly choose the best combination of feasibility and optimum performance but understand why that choice is the best.

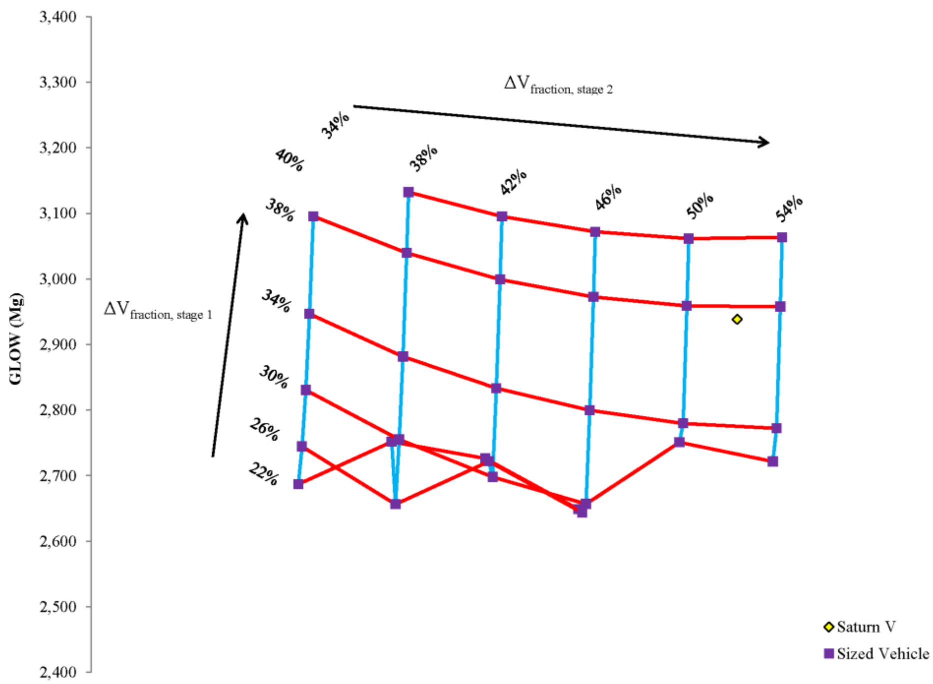

Figure 4.9 Rough $\Delta V_{fraction}$ for stage 1 and 2 vs. GLOW

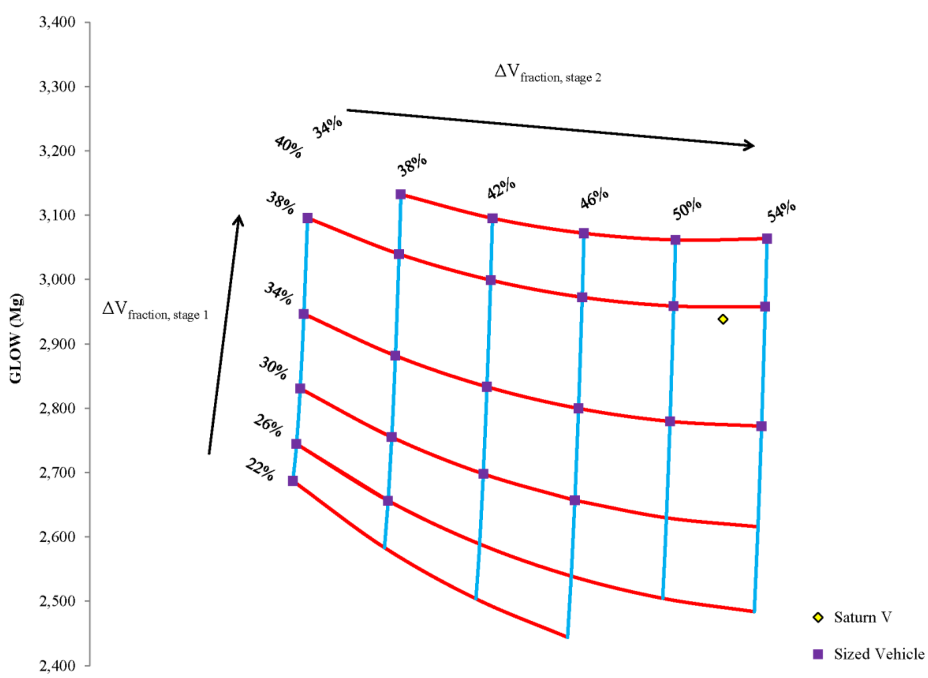

Figure 4.10 Smoothed $\Delta V_{fraction}$ for stage 1 and 2 vs. GLOW

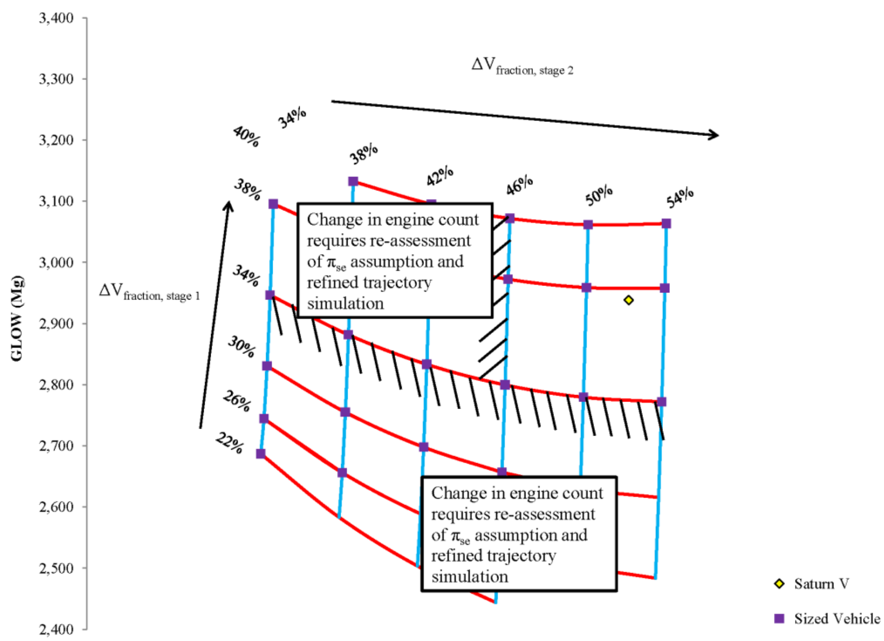

Figure 4.11 Smoothed $\Delta V_{fraction}$ for stage 1 and 2 vs. GLOW with Engine Count Change

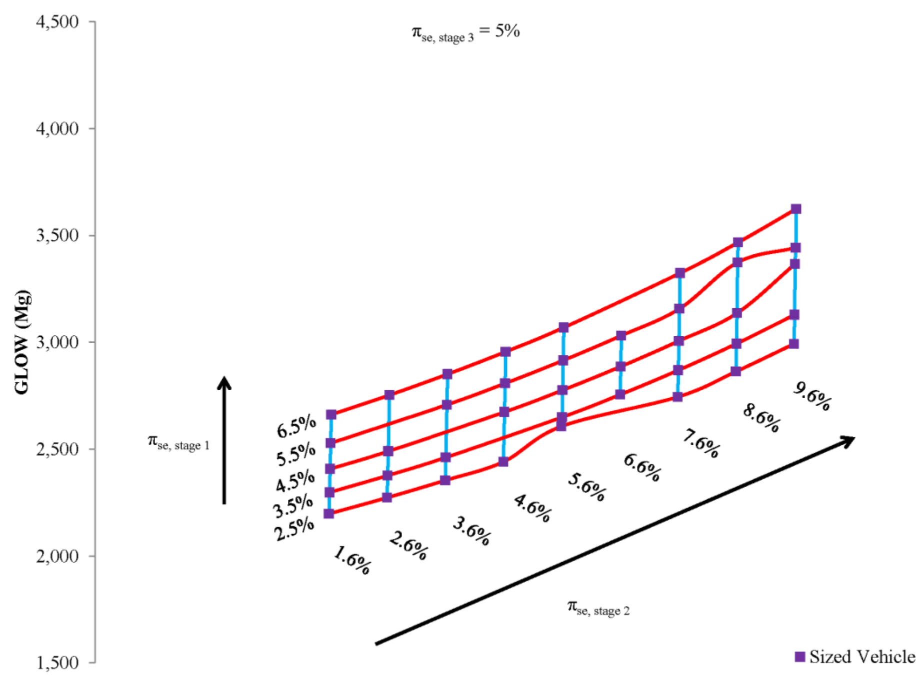

Figure 4.12 GLOW vs. $\pi_{se,\ stage\ 1}$ and $\pi_{se,\ stage\ 2}$ With Constant $\pi_{se,\ stage\ 3}$ of 5%

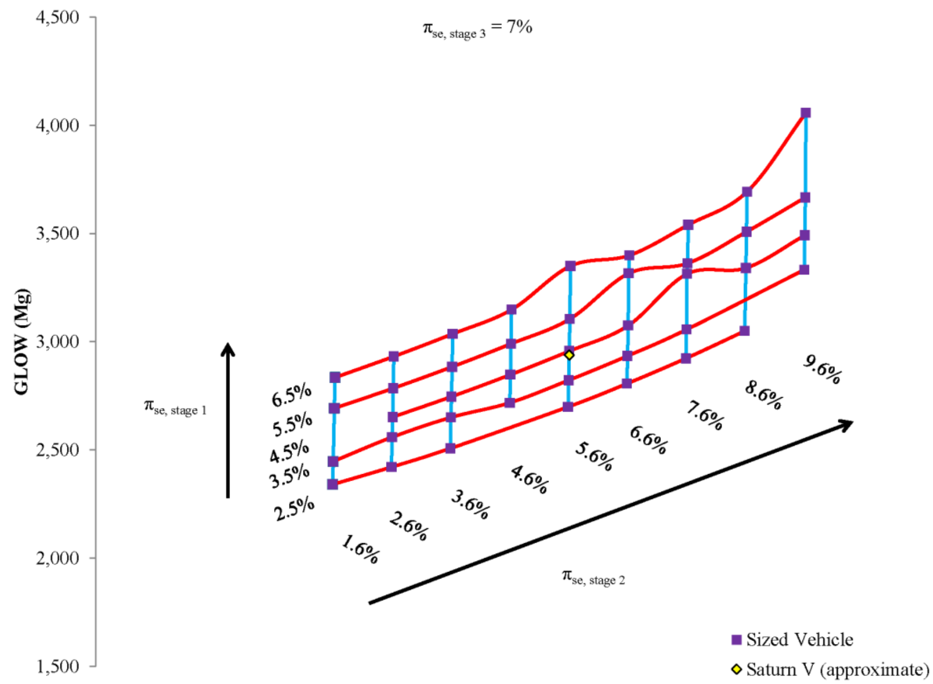

Figure 4.13 GLOW vs. $\pi_{se,\ stage\ 1}$ and $\pi_{se,\ stage\ 2}$ With Constant $\pi_{se,\ stage\ 3}$ of 7%

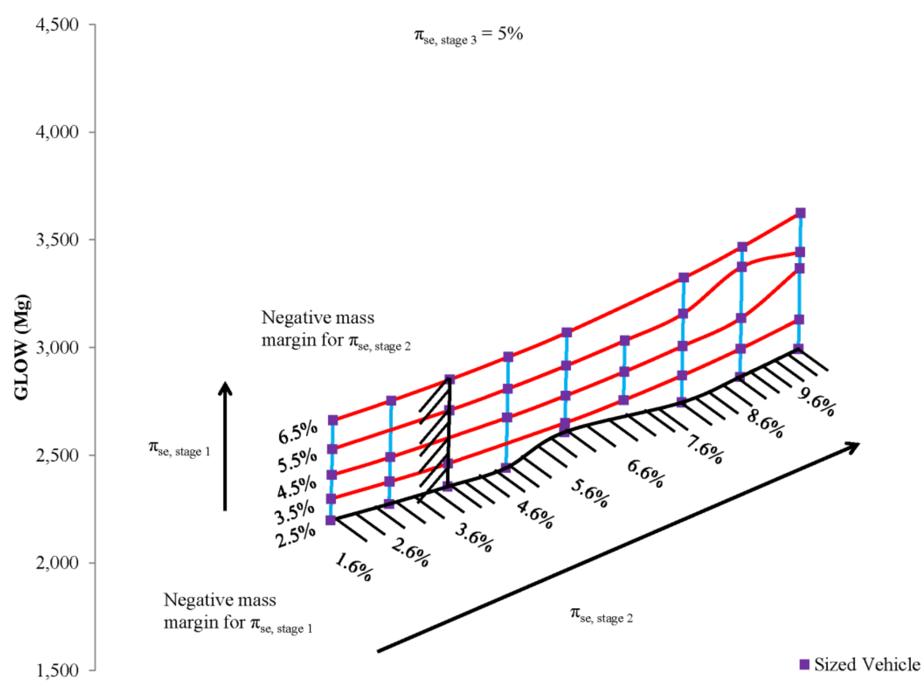

Figure 4.14 GLOW vs. $\pi_{se,\ stage\ 1}$ and $\pi_{se,\ stage\ 2}$ With Constant $\pi_{se,\ stage\ 3}$ of 5% With Limits for Negative Mass Margins

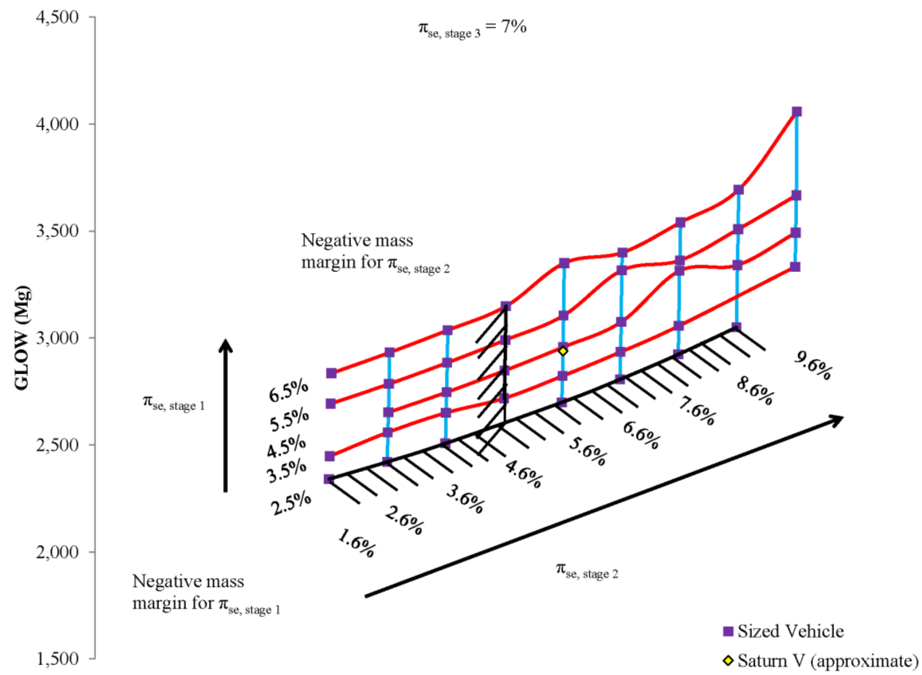

Figure 4.15 GLOW vs. $\pi_{se,\ stage\ 1}$ and $\pi_{se,\ stage\ 2}$ With Constant $\pi_{se,\ stage\ 3}$ of 7% With Limits for Negative Mass Margins

Figure 4.12 and Figure 4.13 present the plots for GLOW vs. $\pi_{se,\ stage\ 1}$ and $\pi_{se,\ stage\ 2}$ with a fixed payload mass of 48,735 kg; fractional ΔV values of 38%, 52%, and 10% for the first, second, and third stages, respectively; and $\pi_{se,\ stage\ 3}$ of 5% for the first plot and 7% for the second plot. Both plots display the same noise as seen before due to the changing value of $\Delta V_{losses,}$ but as it does not interfere with the interpretation of the trend lines these plots are not smoothed. The plot for a constant and $\pi_{se,\ stage\ 3}$ of 7% contains the approximate location for the Saturn V configuration. Note that this is an approximate placement for the Saturn V because the values used for the ΔV vary slightly from the actuals.

These figures include several configurations whose resulting mass margin from the Systems Mass Check Module are below zero. As was seen in the GLV case study, the results of this module should be used as a guideline and not a rule, and as such limits of negative 5% of π_{se} for each stage is selected as the limit to apply. When applied, both values for $\pi_{se,\ stage\ 3}$ return as potentially feasible, and $\pi_{se,\ stage\ 1}$ runs a negative mass margin at 2.5% and below, and $\pi_{se,\ stage\ 2}$ has a negative mass margin at 3.6% if $\pi_{se,\ stage\ 3}$ is 5% and at 4.6% if $\pi_{se,\ stage\ 3}$ is 7%. If a modern version of the Saturn V were to be designed today with the same ΔV fractions, the data indicates the lowest technologically feasible values for π_{se} are 2.5% for the first stage, 3.6% for the second stage, and 5% for the third stage. This would results in a GLOW of 2,353,913 kg, a 9.8% reduction from the Saturn V.

4.3 Falcon 9 Expendable

SpaceX broke into the space-access industry when the Falcon 1 successfully delivered a payload to orbit on July 15, 2009. The Falcon 1 and any potential variants were shelved after its first flight in favor of focusing on the next part of the Falcon family, the Falcon 5. However, that too was shelved when a customer whose payload exceeded the Falcon 5's planned payload mass and geometry capacity inquired with SpaceX about purchasing a launch. In order to meet this customer's requirements, the Falcon 9 was born [156], [157], [158].

The Falcon 9 is a two-stage, liquid-propellant launch vehicle. In contrast to how launch vehicles have historically been designed, the Falcon 9 has been designed using cost engineering principles. Many features of the vehicle, such as equal stages diameters, the octaweb engine structure, and common engines between stages, have been designed with the goal of keeping development and production costs low [117], [125], [159].

There are currently five variants for the Falcon 9: v1.0; v1.1; v1.2, more commonly referred to as "Full Thrust"; "Block 4"; and "Block 5", which has been stated to be the "final version" [160]. A comparison summary of the different variants may be found in Table 4.17.

4.3.1 Point Check

While the Falcon 9 is a proprietary system, the vast majority of the data required for conceptual design sizing of a similar vehicle is available publicly or can be calculated based on the available data. However, several inconsistencies and errors were noticed during the compiling of this data. Consider the vehicle data presented in Table 4.18. Adding together the second stage propellant mass, structure and equipment mass, and the expendable payload capacity produces a resulting GLOW of 134,300 kg. However, this is larger than the mass of the first stage at burnout (138,154 kg). Further investigation into the Falcon 9 Full Thrust revealed that attempts to reverse engineer the Falcon 9 have run into similar issues, and that this has been a consistent problem with backing out the Falcon 9’s mass and performance since the Falcon 9 v1.1. The proposed reasoning behind this is that the vehicle GLOW and payload capacity displayed in Table 4.17 are based on planned final version of the Falcon 9 and are for a baseline vehicle and not necessarily specific to the Falcon 9 Full Thrust [161], [162]. Without an official answer from SpaceX on this matter, the author opted to treat the first and second stage initial masses as the sum of the propellant masses, the structure and equipment masses, and the expendable payload mass.

Table 4.17 Falcon Family Comparison

		v1.0 [163], [164]	v1.1 [165], [166]	Full Thrust [163], [167], [168]	Block 4 [169], [170]	Block 5 [169], [170]
Engine Count & Name	Stage 1	9x Merlin-1C	9x Merlin-1D	9x Merlin-1D	Unspecified performance increases	Enhancements for increased reusability and unspecified performance increases
	Stage 2	1x Merlin-1C (vac.)	1x Merlin-1D (vac.)	1x Merlin-1D (vac.)		
Propellant	Stage 1	RP-1 / LOX	RP-1 / LOX	RP-1 / LOX		
	Stage 2	RP-1 / LOX	RP-1 / LOX	RP-1 / LOX		
Total Length (m)	-	55	68.4	70		
Diameter (m)	-	3.6	3.7	3.7		
Payload Capacity to LEO (kg)	Expendable	10,450	13,150	22,800		
	Reusable*	N/A	9,205	13,680		
GLOW (kg)	-	333,400	505,846	549,054		
First Successful Launch Date	-	6/4/2010	9/7/2014	12/22/2015	8/14/2017	early 2018 (est.)

* Based on a 40% payload capacity reduction [171]

Table 4.18 Falcon 9 Expendable Stage Mass Data

Variable	Stage 1	Stage 2
Advertised Expendable Payload Mass (kg) [167]	22,800	
GLOW (kg) [167]	549,054**	134,300*
Propellant mass (kg) [172]	410,900	107,500
Structure and equipment mass (kg) [172]	22,200	4,000

*Calculated based on the summation of propellant mass, structure and equipment mass, and advertised payload capacity
**Published GLOW but does not match what is expected when the values in the table are summed.

Input Data

Table 4.19 through Table 4.21 define the vehicle information that is used as system inputs. The software's sizing and trajectory results are summarized in Table 4.22 and Table 4.23.

Results

The software took 4.71 seconds to produce results, with the increase in time over the GLV and Saturn V point checks due to the iterations made by the software to converge on a ratio of estimated to actual ΔV_{losses} of 1.05 to 1.10. These results and the simplified model of the sized vehicle are shown in Table 4.22, Table 4.23, and Figure 4.16.

When compared to the overall vehicle results, the software sized a vehicle slightly smaller than the Falcon 9 with most mass data short by 4 to 6%. The overall vehicle length is short by 5.33 meters with a length deficit in the first stage and a surplus in the second stage. The trajectory simulation placed the payload in an orbit 1.2% below what is required with a minor mass deficit. The software increased the ΔV_{losses} to 1,693.8 m/s in order to make up for a deficit from its initial ascent-to-orbit attempt, but the final actual losses came out to be 1,584.2 m/s which places the used estimate between the 105 and 110% of the actual losses as is desired by the software. Further analysis of the calculations performed by the system reveals that the trajectory simulation iterated multiple times in order to adjust the flight path and ΔV_{losses} such that the payload reached the desired parking orbit and the ΔV_{losses} was 105% to 110% of the actual losses. This emphasizes the importance of a properly selected trajectory and the requirement of a trajectory simulation in the conceptual design of launch vehicles: based purely on the original inputs, the first vehicle that was sized was incapable of completing the mission. Only after revisions to the ΔV_{losses} and the flight path was a valid design found.

Table 4.19 Falcon 9 Expendable Input Data

Variable name	Value Stage 1	Value Stage 2	Units
Launch latitude [88]	28.474		-
Parking Orbit Altitude [88]	200,000		m
Payload mass [167]	22,800		kg
Payload diameter [167]	3.7		m
Max permitted g-load [88]	6		-
Payload length [167]	13.1		m
ΔV_{losses}	1,676		m/s
Number of stages [88]	2		-
$\Delta V_{fraction}$*	40.71%	59.29%	-
π_{se}*	3.91%	2.98%	-
$I_{sp,\,mean}$ [173], [174]	296.5	348	s
T_{eng} [167]	845,200	914,100	N
Minimum throttle percentage [175]	40%	40%	-
Mass per engine** [176]	467.2	467.2	kg
Engine length	1.0668	6.8072	m
Engine nozzle exit area	0.7762	8.004	m^2
Gap between engines	-	-	m
Fuel-to-oxidizer ratio	0.3906	0.4238	-
$\rho_{oxidizer}$ [172]	1253.9	1253.9	kg/m^3
ρ_{fuel} [172]	826.5	826.5	kg/m^3
T/W_{min}***	1.3	0.5	-
T/W_{max}***	2	2	-
Gravity turn start altitude(4)	500		m
Time between MECO and staging	5	N/A	s
Delay between staging and next stage startup	N/A	5	s
q_{max}(5)	29893	29893	Pa

*Calculated based on other known input or output data
**The mass of the second stage engine is likely higher due to the modifications made from Merlin 1D to Merlin 1D vacuum. However, no information on the increase in mass was found during the literature search, and therefore Merlin 1D mass was used for both stages.
***Input range selected based on actual T/W and standard recommended values
(4)Based on launch webcasts such as reference [177]. Actual value varies.

Table 4.20. Falcon 9 Expendable Initial Flight Plan*

Start time (s)	**$\dot{\gamma}$ (deg./s)**
0.0000	0.0000
27.3528	-0.3613
29.8875	-0.5770
32.4231	-0.6550
43.8374	-0.7000
54.8340	-0.8181
64.5595	-0.6910
75.7617	-0.5936
86.3275	-0.5429
97.1032	-0.4716
106.3983	-0.3850
115.0580	-0.3699
123.5061	-0.2455
131.9518	-0.2586
139.9754	-0.2308
147.1539	-0.1521
153.9090	-0.1800
161.9311	-0.1538
169.9528	-0.1914
179.6641	-0.2214
192.7541	-0.2053
205.8436	-0.1360
216.3980	-0.1697
226.1088	-0.1708
234.5529	-0.1160
245.1068	-0.1114
256.0827	-0.1210
267.9032	-0.0855
279.7227	-0.1097
289.0099	-0.0652
301.2510	-0.0762
311.8038	-0.0677
320.6681	-0.0380
341.3484	-0.0367

* Modified trajectory based off of the Gemini IV mission trajectory.

Table 4.21 Falcon 9 Expendable Cost Factors

Item	Value	Justification
Average number of launches per year	20	Based on SpaceX's actual launch rate for 2017
Total number of vehicles to produce	200	Assumption that 20 flights remains constant over 10 years
Annual vehicle production	20	-
Number of flights per vehicle	1	Vehicle is fully expendable
$f_{0,\ development}$	1.0816	Fixed value based on number of stages
$f_{0,\ production}$	1.02	The Falcon 9 is a low complexity vehicle and program
f_1	0.9	The Falcon 9 is a standard project with some new design goals in mind (reusability)
f_3	1	The Falcon 9 team has a combination of new and experienced engineers
$f_{4operations}$	0.7	Falcon 9 stages are simple
Development schedule	100%	Assumption that the vehicle is on time
f_7	1	No major contractors are used for design or production
f_8	1	Country is the US
f_9	1	SpaceX is the lead designer and manufacturer of their components
f_{10}	0.7	SpaceX has a strong emphasis on cost engineering
f_{11}	0.5	SpaceX is a commercial venture
f_v	0.8	Expendable, liquid-propellant vehicle with storable propellant
f_c	0.70	Stages assembled horizontally and transported to launch pad
Insurance Premium Cost On the Cost-Per-Flight*	5%	-

*SpaceX is required to purchase insurance for damage to the facility but doesn't purchase insurance for the rocket or cover insurance for the payload. SpaceX would either refund the customer's launch fee or provide a new launch without cost if the launch vehicle fails [178], [179]. Therefore, based on the assumptions that TRANSCOST uses, the insurance cost is set of 5% of the cost-per-flight for SpaceX

Despite the calculated second stage mass being lower than the actual values, the output geometry is nearly 25% larger than the real vehicle. This is likely due to the potential definition error discussed in Section 4.1.1. If the published value for the first stage length includes the interstage fairing and the second stage length excludes the second stage engine nozzle, this would shift approximately 6.8 meters from the second stage to the first brings the first stage's length to 43.44 meters (for an error of 1.97%) and the second stage's length to 10.13 meters (an error of -19.60%). While the error for the second stage remains high and the overall length error is unchanged, this emphasizes the importance of using the same definition for length of the vehicle if the software is being used to size and compare a launch vehicle similar to an existing one.

Cost data for the Falcon 9 is much more difficult to source. Although they can be used for ball-park estimates, values from studies one performed by NASA in 2011 [180], are for the Falcon 9 v1.0 and are not directly applicable to the total development cost of the Falcon 9 Full Thrust. Other sources make assumptions based on based on other assumptions, such as those in references [181] and [182], and as such their level of accuracy is unknown. The information that is known for certain is:

- From 2002 to 2010 SpaceX spent somewhere under $800M. This cost included development costs for the Falcon 1, Falcon 9, and Dragon capsule; building launch site facilities in three locations and their manufacturing facilities; and flights for five Falcon 1, two Falcon 9, and one Dragon capsule. Additionally, the development costs for just the Falcon 9 were "just over $300M" [183]. Although not directly stated, as the $800M value was total expenditures for the company it can be safely assumed that this also included the development costs for the Merlin rocket engine and not just the stages alone.

- The first stage of the Falcon 9 represents approximately 70% of the total production cost [184], and in general engines represent the largest share of cost for a launch vehicle stage [48].

- SpaceX charges $62M for a standard launch [167]. If the launch is for the government, such as a resupply mission to the International Space Station, the cost significantly rises due to the addition requirements that must be met and costs related to the Dragon capsule [183], [185].

- The propellant cost of the ranges somewhere between $200,000 and $300,000 [186].

- SpaceX has previously stated it would cost $1B to develop a new rocket engine [187].

In a 2017 Skype interview with Tom Mueller, CTO of Propulsion Development at SpaceX, Mueller recalls a discussion between himself and Elon Musk regarding the production cost of a Merlin engine vs. the production cost of a Tesla. During the conversation, Musk asks Mueller why it costs "some fraction of a million dollars" per rocket engine [188]. Mueller's reflection on the conversation implies a production cost as little as $100,000 per engine, but this isn't directly stated so all that can be said for certain is the cost to produce a single Merlin engine is less than $1M.

Table 4.22 Falcon 9 Expendable Sizing Results Summary

Variable	Calculated Value	Actual Value*	Percent Error
$\Delta V_{stage\ 1}$ (m/s)	3,693.71	3,746.00	-1.40%
$\Delta V_{stage\ 2}$ (m/s)	5,378.69	5,502.00	-2.24%
$m_{initial,\ stage\ 1}$ (kg)	532,525.51	567,400.00	-6.15%
$m_{initial,\ stage\ 2}$ (kg)	128,743.34	134,300.00	-4.14%
$m_{se,\ stage\ 1}$ (kg)	20,821.75	22,200.00	-6.21%
$m_{se,\ stage\ 2}$ (kg)	3,836.55	4,000.00	-4.09%
$m_{propellant,\ stage\ 1}$ (kg)	382,960.42	410,900.00	-6.80%
$m_{propellant,\ stage\ 2}$ (kg)	102,106.79	107,500.00	-5.02%
$m_{fuel,\ stage\ 1}$ (kg)	107,573.15	123,500.00	-12.90%
$m_{fuel,\ stage\ 2}$ (kg)	30,388.93	32,300.00	-5.92%
$m_{oxidizer,\ stage\ 1}$ (kg)	275,387.27	287,400.00	-4.18%
$m_{oxidizer,\ stage\ 2}$ (kg)	71,717.86	75,200.00	-4.63%
Stage 1 engine count	9	9	0.00%
Stage 2 engine count	1	1	0.00%
Stage 1 diameter (m)	3.70	3.70	0.00%
Stage 2 diameter (m)	3.70	3.70	0.00%
Total vehicle length (m)	64.67	70.00	-5.04%
Stage 1 length (m)	34.64	42.60	-11.53%
Stage 2 length (m)	16.93	12.60	24.44%
Gimbal angle required to maintain stability (deg.)	0.94	-	-
Max gimbal angle available (deg.)	-	UNKNOWN	-

*Actual values are either calculated based on known values or are from references [167] and [172]

Table 4.23 Falcon 9 Expendable Trajectory Result Summary

Variable	Calculated Value
Final altitude (m)	197,596.0
Altitude target (m)	200,000.0
Altitude surplus/deficit (m)	-2,404.0
Percent error in altitude	-1.20%
γ_{final} (deg.)	0.0
Final velocity (m/s)	7,238.0
Velocity gain from launch location (m/s)	407.9
Total vehicle velocity (m/s)	7,645.9
Velocity required to maintain orbit achieved (m/s)	7,787.9
Velocity surplus/deficit for orbit achieved (m/s)	-142.0
Percent error in velocity for orbit achieved	-1.82%
Estimated ΔV_{losses} (m/s)	1,693.8
Calculated ΔV_{losses} (m/s)	1,584.2
Percent error in ΔV_{losses}	9.31%

Based on this information, it can be expected that the total development cost from the Cost Module that a clean sheet design of the Falcon 9 Full Thrust should be less than $2B, the production cost of the engines should be well under $1M, and the total production cost of the first stage should be approximately 70% of the total production costs.

Results from the cost module are presented in Table 4.24. Taking into account the above, the total development costs are significantly higher than expected. The author consulted the *Handbook of Cost Engineering* [125] in an attempt to find methods which would be used to bring the calculated development cost down, but the Cost Module is already at the lower bounds of what the *Handbook* recommends. The engine production cost is in the range of what is expect for the first stage at $5M for all nine engines ($555,555 per engine). The stage alone production cost for the first stage is also abnormally high compared to what is expected when compared to the production cost of all of its engines, with the stage costing nearly three times as much as all nine engines. However, the resulting calculated total production cost of the first stage is 71.9% of total production cost, close to the 70% that was expected at.

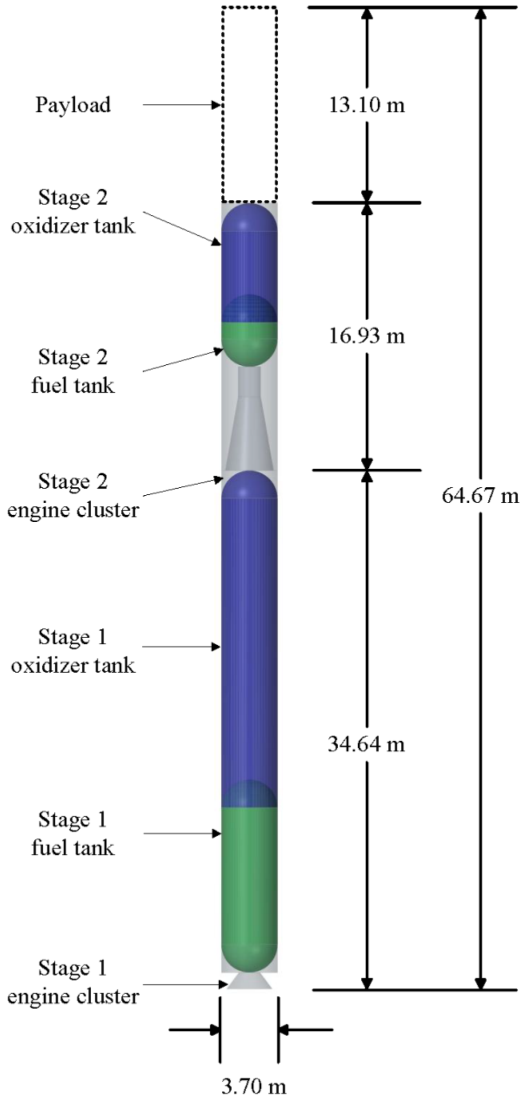

Figure 4.16 Simplified Model of the Sized Falcon 9 (Expendable)

The calculated cost to purchase a Falcon 9 launch is $22.7M higher than the actual price. In order to analyze this error, it is important to understand the definitions of Cost per Flight, Price per Flight, and Complete User Cost. The *Handbook of Cost Engineering* uses the following definitions [125]:

$$\begin{aligned} &Cost\ per\ Flight \\ &\qquad = Total\ Vehicle\ Production\ Cost + Direct\ Operating\ Costs \\ &\qquad + Indirect\ Operating\ Costs \end{aligned} \tag{4.1}$$

Table 4.24 Falcon 9 Expendable Cost Results

Variable		Calculate Value (WYr)	Actual Value (WYr)	Calculate Value (in $M, 2015 dollars)	Actual Value (in $M, 2015 dollars)	Percent Error
Engine Development Cost	Stage 1	4,785.028	-	$1,613.033	UNKNOWN	-
	Stage 2	4,785.003	-	$1,613.024		-
	Sum	9,570.056	-	$3,226.066		-
Stage Development Cost	Stage 1	6,893.167	-	$2,323.686	UNKNOWN	-
	Stage 2	4,538.049	-	$1,529.776		-
	Sum	11,431.215	-	$3,853.463		-
Total Development Cost	Stage 1	12,631.135	-	$4,257.956	Expected under $2B	-
	Stage 2	10,083.840	-	$3,399.262		-
	Sum	22,714.979	-	$7,657.220		-
Engine Production Cost	Stage 1	14.855	-	$5.008	Less than $1M per engine	-
	Stage 2	5.814	-	$1.960		-
	Sum	20.669	-	$6.968		-
Stage Production Cost	Stage 1	41.794	-	$14.089	UNKNOWN	-
	Stage 2	16.302	-	$5.495		-
	Sum	58.096	-	$19.584		-

Table 4.24 Falcon 9 Expendable Cost Results (cont.)

Variable		Calculate Value (WYr)	Actual Value (WYr)	Calculate Value (in $M, 2015 dollars)	Actual Value (in $M, 2015 dollars)	Percent Error
Total Production Cost	Stage 1	58.979	81.578	$19.882	$27.500*	-27.70%
	Stage 2	23.009	-	$7.756	UNKNOWN	-
	Total	81.947	-	$27.624	UNKNOWN	-
Prelaunch Costs		11.529	-	$3.886	UNKNOWN	
Flight Operations Cost		1.598	-	$0.539	UNKNOWN	-
Direct Operations Cost		21.664	-	$7.303	UNKNOWN	-
Cost Per Flight		116.178	-	$39.164	UNKNOWN	-
Price Per Flight		239.629	-	$80.779	-	-
Complete User Cost		209.208	183.92	$82.737	$62.000	33.45%

$$Price\ per\ Flight = Cost\ per\ Flight + Development\ Amortization + Nominal\ Profit\ Amount \tag{4.2}$$

$$Complete\ User\ Cost = Price\ per\ Flight + Insurance \tag{4.3}$$

The nominal profit amount used in the Price per Flight calculation is a percentage of the Cost per Flight and was fixed at 8.5% for all calculations in this thesis. For the expendable Falcon 9, this means the nominal profit is \$3.33M, only a small fraction of the increase from the Cost per Flight to Price per Flight; \$38.29M comes from the amortization of the \$7,657M development costs. Based on the cost information listed above, the actual development costs should be significantly lower than this calculated value. As this point check uses the lower bound for all cost factors in its analysis of the Falcon 9, it is not possible to more closely match the inferred development costs incurred by SpaceX without making improvements to the CERs, and such changes to the *Handbook of Cost Engineering* is beyond the scope of this thesis. However, if the calculated development costs could be reduced by a factor of 2, the new Complete User Cost would be approximately \$63.6M, only \$1.6M higher than the actual cost.

4.3.2 Iterative Sweep

The values presented in Table 4.9 the range of values for which the Falcon 9 data sweep is run. As was done with the previous two case studies, the mass margin check is disabled.

These inputs result in a total of 47,430 potential configurations. Out of these, 17,819 are returned as potentially viable vehicles. Utilizing a desktop computer with an Intel Core i7 5820k and 32 GB of RAM in conjunction with using multithread processing tools in Python, it took the software 262 minutes to run and output the data for this range of outputs.

Figure 4.17 presents a plot of GLOW vs. $\pi_{se,\ stage\ 1}$ and $\Delta V_{fraction,\ stage\ 1}$ for vehicles with a constant payload mass of 22,800 kg and $\pi_{se,\ stage\ 2}$ of 2.98%. The plot displays fluctuations in GLOW inconsistent with the general trend in several locations, such as a $\pi_{se,\ stage\ 1}$ of 2.91% and $\Delta V_{fraction,\ stage\ 1}$ of 35%. As with the prior two case studies, these variations exist due to the software's adjustment to the ΔV_{losses} estimate to successfully deliver the payload into orbit generating slightly higher or lower GLOW than what is calculated using the original estimate.

When verifying the feasibility with the vehicles with the results from the mass margin check, it is revealed that vehicles with a $\pi_{se,\ stage\ 1}$ have a total vehicle margin of negative 4,000 kg or less indicating a hard limitation here due to the currently available technology. There also appears to be limits on fraction of ΔV as the sizing results do not contain any vehicles with a $\Delta V_{fraction,\ stage\ 1}$

less than 31% or greater than 47%. However, this is due to the limits of the trajectory simulation: the vehicle designs are feasible and could be design, built, and flown, but the software could not determine a viable trajectory using the path provided at the start.

Table 4.25. Falcon 9 Expendable Sweep Inputs

Variable	Minimum Value	Maximum Value	Interval
$\pi_{se,\ stage\ 1}$	1.91%	6.91%	1%
$\pi_{se,\ stage\ 2}$	0.98%	4.98%	1%
$\Delta V_{fraction,\ stage\ 1}$*	25%	75%	1%
Payload mass (kg)	12,800	42,800	1,000

*$\Delta V_{fraction,\ stage\ 2}$ = 100% - $\Delta V_{fraction,\ stage\ 1}$

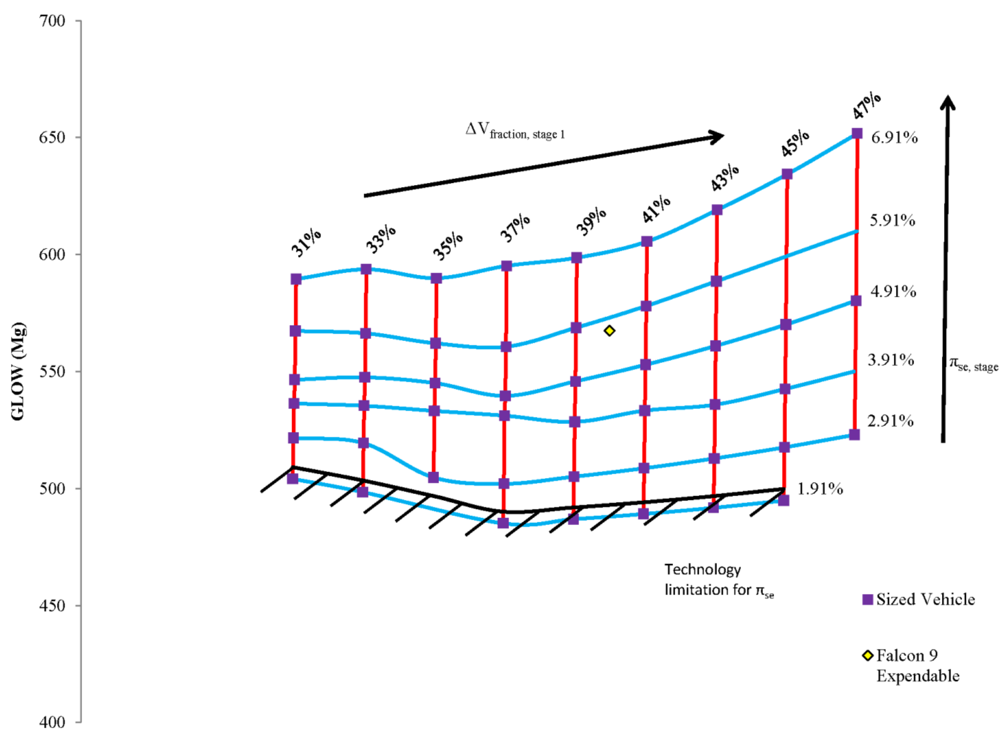

Figure 4.17. GLOW vs. varying $\pi_{se,\ stage\ 1}$ and $\Delta V_{fraction,\ stage\ 1}$ for the Falcon 9 Expendable

4.4 Falcon 9 Reusable

On December 21, 2015, SpaceX successfully launched a payload into orbit on the Falcon 9 and then recovered the first stage with a boostback landing recovery by flying the stage to the launch site. The stage of the vehicle was equipped with four landing legs at the base of the stage for use when landing, nitrogen gas thrusters at the top of the stage for steering in low-density parts of the atmosphere, and four grid fins at the top of the stage for aerodynamic steering. By mid-September 2017 SpaceX has landed sixteen first stages on both land and on drone ships out floating downrange of the launch site. Two of these were stages that had flown and been recovered previously. [1], [5], [6], [7], [8], [189].

The reusable version of the Falcon 9 Full Thrust is nearly identical to the expendable version. The differences come from the inclusion of landing hardware (landing legs, grid fins, etc.), reduced payload capacity due to propellants reserved for the required boostback burns, and the programmed trajectory information.

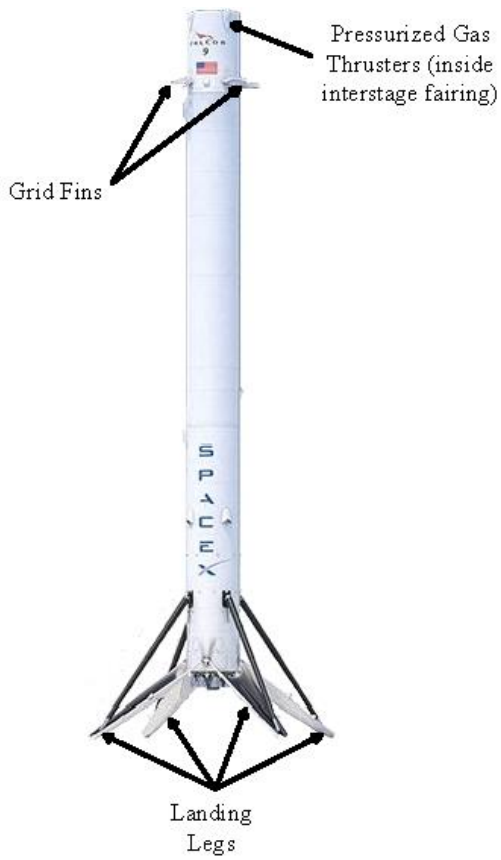

Figure 4.18 Falcon 9 First Stage Landing Hardware [190]

4.4.1 Point Check

Input Data

All of the sizing input data for the reusable version of the Falcon 9 Full Thrust are the same as the Falcon 9 expendable except for the sizing values in Table 4.26. As noted in Section 4.3, the available data for the Falcon 9 does not add up to published GLOW of 549,054 kg, and this number is instead what is expected for the final version of the Falcon 9. This inconsistency holds true for the partially reusable version as well. Therefore, a similar calculation to the one made before to estimate the current GLOW for the vehicle at full payload capacity based on a payload penalty of 35% [171]. Calculations for the values of $\Delta V_{fractions}$ and π_{se} are based on the resulting GLOW.

The trajectory path and ΔV_{losses} estimate are also revised for the reusable Falcon 9. As discussed in Section 3.14.2, the trajectory path followed by a boostback vehicle is significantly different than the path a fully expendable vehicle utilizes. While this trajectory reduces the energy requirements to return the first stage to the launch site, it increases the gravity losses significantly due to the shallow change in γ early in the vehicle's flight. Therefore, the ΔV_{losses} estimate is revised up to 1,925 based on values from initially testing the boostback sizing code, and the default trajectory provided to the sizing program is based on a reconstruction of SpaceX's OTV-5 mission [130].

With reusability now taken into a change to the number of flights per vehicle must be made to the cost inputs. SpaceX has estimated the first stage for the Full Thrust variant to only see two to three reuses but as many as a dozen reuses for the Block 5 version [191]. For the purposes of this thesis the first stage is set at ten reuses.

These values are shown in Table 4.27.

Table 4.26 Revised Input Data for the Falcon 9 Reusable

Variable name	Value		Units
	Stage 1	Stage 2	
Payload mass	14,820		kg
ΔV_{losses}	1,925		m/s
$\Delta V_{fraction}$*	31.26%	68.74%	-
π_{se}	3.97%	3.20%	-
Number of stage 1 reuses	10		-

*Used for ascent

Results

The sizing and trajectory results of the point check are presented in Table 4.28 and Table 4.29, and took the software 17.02 seconds to produce. The simplified model of the sized vehicle is provided in Figure 4.19. This significant increase in computation time over the expendable vehicles is due to the calculations required by the RTLS simulation to determine when to begin and terminate the boostback, re-entry, and landing burns. Unlike the expendable results, the calculated size of the reusable Falcon 9 is larger in the first stage and slightly small for the second stage. The propellant reserved for ascent is nearly 15% higher than the expected result, but in spite of this the propellant reserved for the boostback trajectory was 18% less than what was estimated from the OTV-5 mission. The trajectory simulation ended with an altitude below that of the parking orbit and final velocity lower than required to maintain its orbit, but within the bounds set for the low-order simulation. The calculated ΔV_{losses} was within 1% of the original estimate.

Table 4.27. Falcon 9 Reusable Initial Flight Plan*

Start time (s)	$\dot{\gamma}$ (deg./s)
0	0.0000
28	-0.1613
38	-0.2770
47	-0.3550
69	-0.4500
70	-0.5500
71	-0.6500
72	-0.7500
93	-0.5429
94	-0.4716
116	-0.3850
118	-0.3699
121	-0.2455
122	-0.2586
131	-0.2308
133	-0.1521
136	-0.1800
137	-0.1538
139	-0.1914
140	-0.2214
141	-0.2053
148	-0.1360
150	-0.3000

* Based on SpaceX's broadcasted data for the OTV-5 mission [130].

Table 4.28 Falcon 9 Reusable Sizing Results Summary

Variable	Calculated Value	Actual Value*	Percent Error
$\Delta V_{stage\ 1}$ (m/s)**	4,063.30	3,874.00	4.89%
$\Delta V_{stage\ 1,\ ascent}$ (m/s)***	2,913.30	2,554.21	14.06%
$\Delta V_{stage\ 1,\ boostback}$ (m/s)***	4,000.00	3,891.90	2.78%
$\Delta V_{stage\ 2}$ (m/s)	6,406.29	6,682.00	-4.13%
$m_{initial,\ stage\ 1}$ (kg)	589,193.80	558,280.00	5.54%
$m_{initial,\ stage\ 2}$ (kg)	122,308.00	125,180.00	-2.29%
$m_{se,\ stage\ 1}$ (kg)	23,426.35	22,200.00	5.52%
$m_{se,\ stage\ 2}$ (kg)	3,907.76	4,000.00	-2.31%
$m_{propellant,\ stage\ 1}$ (kg)	443,458.79	410,900.00	7.92%
$m_{propellant,\ stage\ 1\ ascent}$ (kg)***	374,213.48	326,283.42	14.69%
$m_{propellant,\ stage\ 1\ boostback}$ (kg)***	69,245.31	84,616.58	-18.17%
$m_{propellant,\ stage\ 2}$ (kg)	103,580.91	107,500.00	-3.65%
$m_{fuel,\ stage\ 1}$ (kg)	124,567.08	123,500.00	0.86%
$m_{fuel,\ stage\ 2}$ (kg)	30,827.65	32,300.00	-4.56%
$m_{oxidizer,\ stage\ 1}$ (kg)	318,891.71	287,400.00	10.96%
$m_{oxidizer,\ stage\ 2}$ (kg)	72,753.26	75,200.00	-3.25%
Stage 1 engine count	9	9	0.00%
Stage 2 engine count	1	1	0.00%
Stage 1 diameter (m)	3.70	3.70	0.00%
Stage 2 diameter (m)	3.70	3.70	0.00%

*Actual values are either calculated based on known values or are from references [167] and [172]
**If all propellant is used for ascent.
*** Based on burn times from SpaceX's broadcasted data for the OTV-5 mission [130].

Table 4.28 Falcon 9 Reusable Sizing Results Summary (cont.)

Variable	Calculated Value	Actual Value*	Percent Error
Total vehicle length (m)	69.86	70.00	-0.20%
Stage 1 length (m)	39.71	42.60	-6.79%
Stage 2 length (m)	17.05	12.60	35.35%
Gimbal angle required to maintain stability (deg.)	1.03	-	-
Max gimbal angle available (deg.)	-	UNKNOWN	-

*Actual values are either calculated based on known values or are from references [167] and [172]
**If all propellant is used for ascent.
*** Based on burn times from SpaceX's broadcasted data for the OTV-5 mission [130].

Table 4.29 Falcon 9 Reusable Trajectory Result Summary

Variable	Calculate Value
Final altitude (m)	191,007.8
Altitude target (m)	200,000.0
Altitude surplus/deficit (m)	-8,992.2
Percent error in altitude	-4.50%
γ_{final} (deg.)	0.0
Final velocity (m/s)	6,843.5
Velocity gain from launch location (m/s)	407.9
Total vehicle velocity (m/s)	7,251.4
Velocity required to maintain orbit achieved (m/s)	7,791.8
Velocity surplus/deficit for orbit achieved (m/s)	-540.4
Percent error in velocity for orbit achieved	-6.94%
Estimated ΔV_{losses} (m/s)	1,925.0
Actual ΔV_{losses} (m/s)	1,908.8
Percent error in ΔV_{losses}	0.85%

A detailed review of the trajectory reveals that the sized vehicle staged earlier and at a lower altitude than what occurred in the OTV-5 mission. The noise in the broadcast data and error generated from reconstructing a trajectory from that data likely resulting in the vehicle ascending at a steeper rate than the actual flight, resulting in higher gravity losses than experienced in the actual mission.

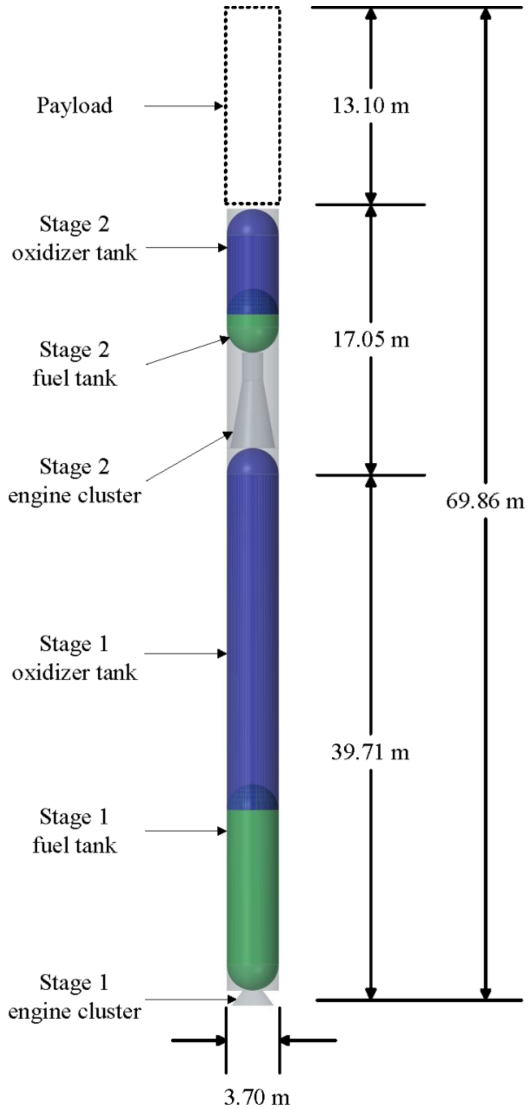

Figure 4.19 Simplified Model of the Sized Falcon 9 (Reusable)

The cost results in Table 4.30 include two additional rows of information. The first is the effective total production cost per flight. This value is the complete production cost of a single stage divided by the number of times that stage can be reused. The second is the effective refurbishment cost per flight which is the total stage and engine refurbishment cost divided by the time of times the stage can be reused. This number is then added to the direct operations cost.

Statements from SpaceX claim a cost reduction of as much as 30% which would bring the cost of a Falcon 9 launch down from $62M to $43.4M [186]. With the cost of the first stage spread over a lifetime of ten flights, the Cost Module calculates a final cost to purchase a launch at $61.911M which is a reduction of 25.17% over the cost of expendable Falcon 9 sized by the software. It is also worth noting that the production cost of a single first stage increased significantly over the cost for calculated for the expendable version of the Falcon 9. This is a direct result of the change in quantity produced per year. The Cost Module has a learning factor built into the production costs. This learning factor takes into account the decrease in cost that occurs when a large number of items are produced due to the familiarity the manufacturer gains with the process and item being created. With the reusable version of the Falcon 9 and an assumption of twenty flights per year, the production rate of first stages dropped from twenty produced per year to two produced per year. This lead to a shift in the learning factor, and production costs increased over the expendable case study as a result.

4.4.2 Iterative Sweep

As discussed at the end of the trajectory simulation modifications in Section 3.14.2, the total runtime for a sizing a single boostback vehicle configuration is expected to be sixteen seconds at a minimum. If significant iterations are required to correct the vehicle's ascent-to-orbit flight path, the ΔV_{losses} estimate, or the $\Delta V_{boostback}$ estimate, the runtime may increase to between fifteen and thirty minutes per configuration. Due to this significant run time and the limited time frame of this research project, only a limited number of cases can be run, and thus only the payload mass, first stage $\Delta V_{fraction}$ are performed, and number of annual launches are varied. This range of inputs is presented in Table 4.31 and results in a total of 35 vehicle configurations and 245 cases for the software to analyze. Out of these, 34 vehicle configurations are returned as potentially viable vehicles. Utilizing a desktop computer with an Intel Core i7 5820k and 32 GB of RAM in conjunction with using multithread processing tools in Python, it took the software 87 minutes to run and output the data for this range of outputs. However, when the input values for $\Delta V_{fraction}$ and payload mass were varied more significantly from the baseline Falcon 9 Reusable, the sizing time increased drastically and ran for over six hours before the author terminated the run due to the time constraints on requiring output data.

As before, the results provided by the sizing software produce the results that varied ΔV_{losses} in order to complete the trajectory simulation. The carpet plot generated by these results is presented in Figure 4.20. All of the sized vehicles identified on the plot are valid, but it is desirable to smooth the plot in order to better identify trends. The smoothed plot is shown in Figure 4.21. The dataset is too small to identify as significant limitations that would indicate design are not feasible, but it does agree with the study performed in reference [127] that decreasing the ascent-to-orbit energy requirements of the first stage of a reusable vehicle can result in an overall decrease to GLOW due to the reduced propellant necessary to return to the launch site and land.

Table 4.30 Falcon 9 Reusable Cost Results

Variable		Calculate Value (WYr)	Actual Value (WYr)	Calculate Value (in $M, 2015 dollars)	Actual Value (in $M, 2015 dollars)	Percent Error
Engine Development Cost	Stage 1	4,785.028	-	$1,613.033	UNKNOWN	-
	Stage 2	4,785.028	-	$1,613.033		-
	Sum	9,570.056	-	$3,226.066		-
Stage Development Cost	Stage 1	2,624.779	-	$884.813	UNKNOWN	-
	Stage 2	4,545.562	-	$1,532.309		-
	Sum	5,407.938	-	$1,823.016		-
Total Development Cost	Stage 1	8,014.447	-	$2,701.670	Expected under $2B	-
	Stage 2	10,091.966	-	$3,402.002		-
	Sum	18,106.413	-	$6,103.672		-
Engine Production Cost	Stage 1	55.066	-	$18.563	Less than $1M per engine	-
	Stage 2	5.814	-	$1.960		-
	Sum	60.880	-	$20.523		-
Stage Production Cost	Stage 1	133.990	-	$45.168	UNKNOWN	-
	Stage 2	16.504	-	$5.564		-
	Sum	-	-	-		-

Table 4.30 Falcon 9 Expendable Cost Results (cont.)

Variable		Calculate Value (WYr)	Actual Value (WYr)	Calculate Value (in $M, 2015 dollars)	Actual Value (in $M, 2015 dollars)	Percent Error
Total Production Cost	Stage 1	196.690	81.578	$66.304	$27.500*	141.11%
	Stage 2	23.009	-	$7.756	UNKNOWN	-
	Sum	-	-	-	UNKNOWN	-
Effective Total Production Cost Per Flight	Stage 1	19.669	-	$6.631	UNKNOWN	-
	Stage 2	23.220	-	$7.827	UNKNOWN	-
	Sum	42.889	-	$14.458	UNKNOWN	-
Prelaunch Costs		14.981	-	$5.050	UNKNOWN	-
Flight Operations Cost		1.598	-	$0.539	UNKNOWN	-
Direct Operations Cost		25.132	-	$8.472	UNKNOWN	-
Effective Refurbishment Cost Per Flight		0.143	-	$0.048	UNKNOWN	-
Cost Per Flight		82.049	-	$27.659	UNKNOWN	-
Price Per Flight		179.555	-	$60.528	UNKNOWN	-
Complete User Cost		183.658	128.745	$61.911	$43.400	42.65%

Table 4.31. Falcon 9 Reusable Sweep Inputs

Variable	Minimum Value	Maximum Value	Interval
$\Delta V_{fraction,\ stage\ 1}$*	28.26%	34.26%	1%
Payload mass (kg)	11,820	15,820	1000
Number of annual launches	10	40	5

*$\Delta V_{fraction,\ stage\ 2}$ = 100% - $\Delta V_{fraction,\ stage\ 1}$

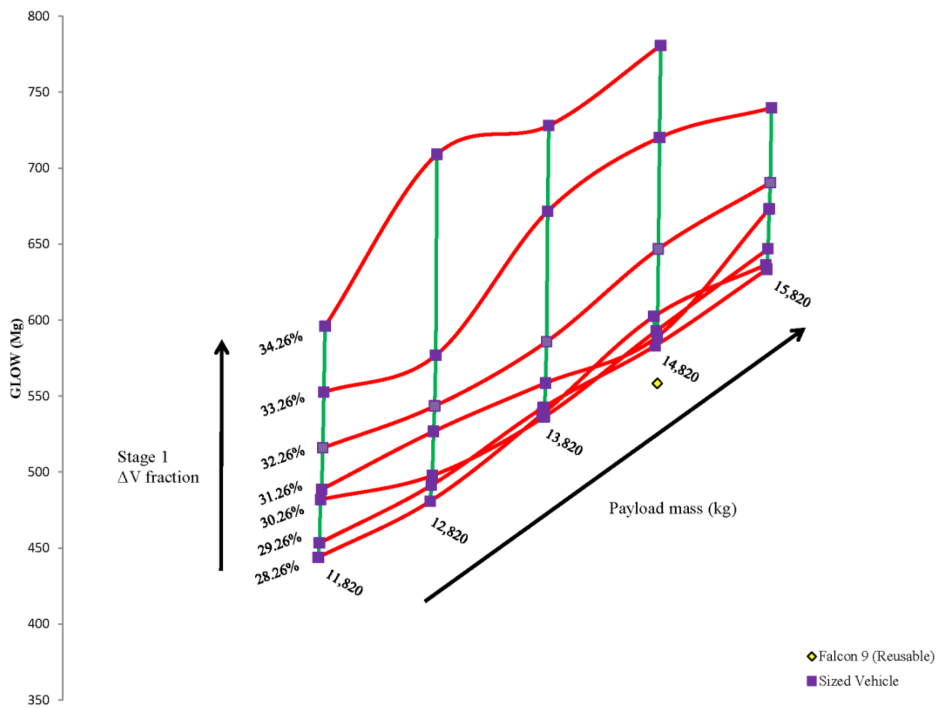

Figure 4.20 Rough Results for the Falcon 9 Reusable for GLOW vs. Delta V Fraction and Payload Mass

Figure 4.22 presents the complete user cost versus the payload mass and annual launch rate. This figure demonstrates the power increased launch rate of a partially reusable vehicle on the potential cost of space access. As the launch rate increases, the cost per launch drops significantly from $69.95M to $60.9M for a payload mass of 15,820 kg. The carpet plot begins to flatten out as the number of launches increases up to forty with a decrease of $0.78M, $0.57M, and $0.43M when the number of flights varies between 30 and 40 per year for a payload of 14,820 kg.

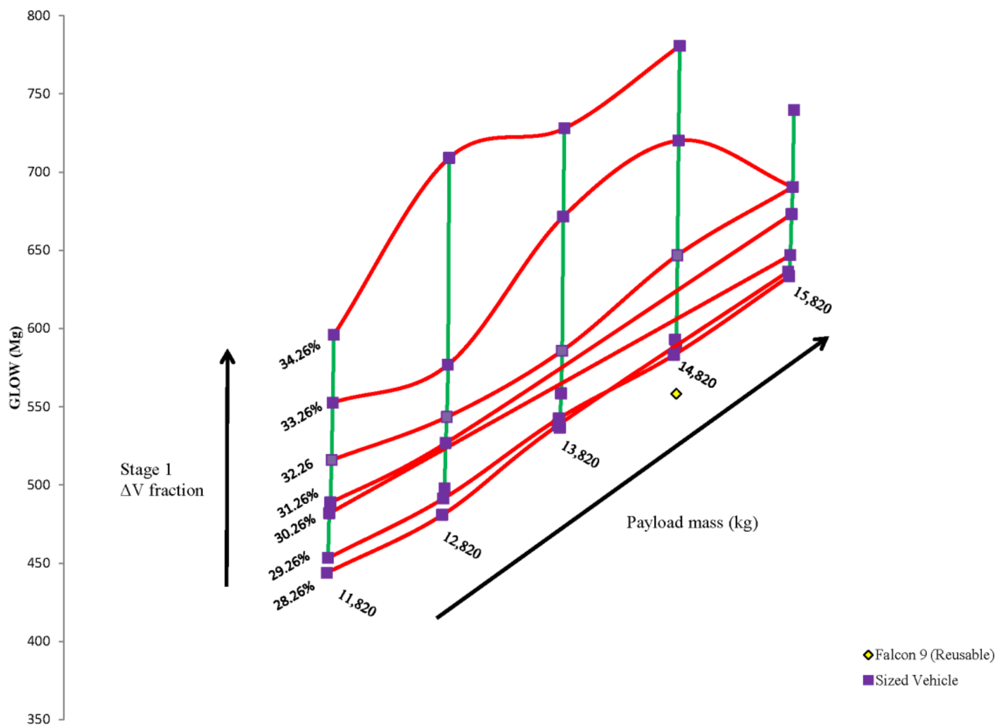

Figure 4.21 Smoothed Results for the Falcon 9 Reusable for GLOW vs. Delta V Fraction and Payload Mass

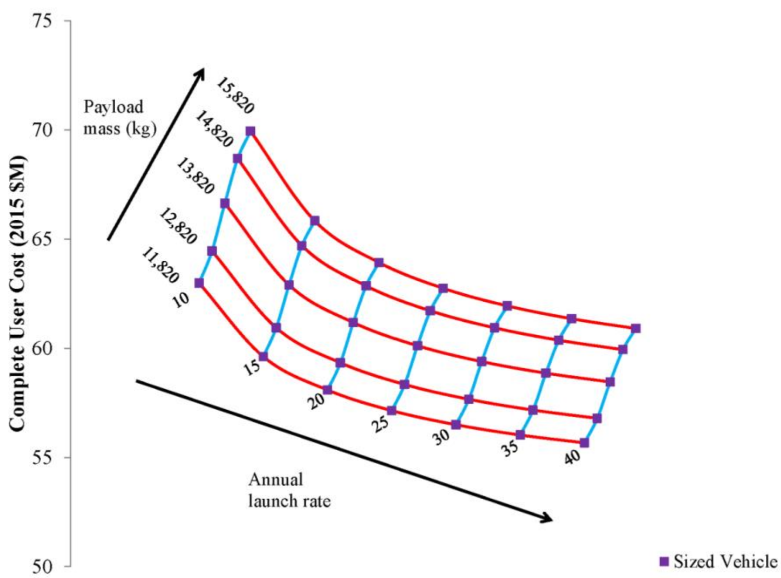

Figure 4.22 Falcon 9 Reusable Complete User Cost vs. Payload Mass and Annual Launch Rate

Chapter 5. Conclusions and Future Work

The first two chapters of this thesis provide a brief history of launch vehicles, the need for a flexible, intuitive software for conceptual launch vehicle design, and presents the literature search used in order to determine what features should be incorporated into the new tool as well as develop a best practice approach for the design process. The third chapter explained the breakdown of the software into various modules which handle one section of the sizing process and how modules could be replaced to add new features such as transfer stage or first stage boostback sizing to the design. The software was verified as a complete system in chapter four with its abilities to:

1. Take initial vehicle design parameters from the user such as the payload mass and geometry, engine properties, and desired orbital altitude, and use these inputs to size a space a fully expendable, fully expendable with combination ascent-to-orbit and transfer stage, and partially reusable with first stage boostback space launch vehicle;

2. Simulate the ascent-to-orbit trajectory for the vehicle and revise the input flight path if it results in a final orbit altitude greater than ±5% of the target orbit;

3. Adjust the initial estimate for ΔV_{losses} until the estimate is between 105% and 110% of the actual value;

4. Simulate the RTLS trajectory for a first-stage boostback vehicle by automatically determining when in the flight the three required burns should begin and end; and

5. Revising the initial estimate for $\Delta V_{boostback}$ if the RTLS simulation depleted the propellant reserved for the boostback maneuvers prior to completing a landing.

Launch vehicle design is a broad field and there are still many features that could be added to the system in the future. Among the various possibilities, the author recommends the following features to be incorporated in the next major iteration:

1. **Refining ascent-to-orbit-trajectory analysis and optimization.**
   The current trajectory analysis requires the user to input a starting trajectory path. As discussed in Section 4.3.2, the input flight path must be semi-feasible for the sized vehicle configuration in order for the trajectory simulation to be successful. This results in vehicles whose design configuration that deviate significantly from the baseline design occasionally required an increase to the ΔV_{losses} when a significant reshaping of the trajectory path would suffice instead. The increase in

ΔV_{losses} leads to an increase in GLOW which may not have been as large should the trajectory have been optimized to minimize losses.
An improved Trajectory Module should no longer require an input flight path but instead auto-generate an initial flight path and then optimize it for the specific vehicle configuration that is being sized.

2. **Coupling the ascent-to-orbit and boostback trajectories for overall optimization.**
   As first introduced in Section 3.14.2, the shape of the trajectory for a fully expendable fully and a partially boostback-reusable vehicle are different. Studies such as the one in reference [127] discuss the sensitivity in GLOW from the staging conditions when attempt to recovery the first stage of a launch vehicle, and this was seen in Figure 4.20 and Figure 4.21. Coupling and refining the two trajectory simulations has the potential to further open the solution space of viable boostback launch vehicles.

3. **Sizing of parallel stage launch vehicles.**
   Launch vehicles that utilizing parallel staging have been in use from the very first launch vehicle, the Sputnik Launch Vehicle (also known as the 8K71), to the upcoming Falcon Heavy from SpaceX and the Space Launch System from NASA. Parallel staging provides a unique set of advantages and disadvantages, and offers many unique design features such as cross-feed which offer the potential of greatly increased performance.

4. **Adding a basic structural analysis.**
   With the current software setup the user must provide a maximum dynamic pressure limit which the vehicle should not exceed. A Structures Module that utilizes low-order estimation techniques should be able to calculate this limitation based on the vehicle's design. Using a maximum dynamic pressure value specific to the sized vehicle has the potential to refine the solution space generated to ensure all of the vehicles included are truly valid designs.

5. **Improvements to the Geometry Module.**
   As noted in the case study, all of the vehicles sized by the system were physically smaller than their actual counterparts. An improved Geometry Module would correct this.

6. **Additions and improvements to the Systems Mass Check Module.**
   As discussed during its explanation in Section 3.10, the Systems Mass Check Module is based on a series of MERs provided by Akin in reference [100].

Additional MERs should be added in the future in order to generate estimates for any thermal protection systems or aerodynamic surfaces which will be used in the future by launch vehicles such as Blue Origin's New Glenn and SpaceX's BFR.

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Appendix A. History of Rocketry

A.1 Origins

The history or rocketry extends back well past the dawn of the space age. In fact, rockets have been around for nearly 1,000 years. It is unknown exactly when rockets first came about, but historical evidence points to their origin in China. It is likely that the concept of rockets emerged from gunpowder being loaded into bamboo tubes and tossed into fires. If the tube was sealed at only one end, it is possible that instead of exploding into pieces the hot gasses would fly out the open end of the bamboo and propel the bamboo in the opposite direction. This was likely first adapted into what became the earliest fireworks and into the first true rockets at a much later date [18].

Whatever the specific origins were, it is known that the first confirmed rockets appeared in 1232 AD. In his book *Rockets: Sulfur, Sputnik, and Scramjets*, Peter Macinnis writes about how the Chinese city of Kai-fung-fu was defended by using 'arrows of fire', rockets that used black powder explosions as a propulsion system, and how it wasn't long afterwards that rockets were used elsewhere. They spread across Asia towards the Middle East where they were used as early as 1258 in Baghdad, and eventually into Europe where they found favor as celebratory fireworks instead of weapons of war. The Italians in particular were known for their fireworks in the late 1300s, and it is in fact from Italy were the word "rocket" originates [18], [192].

Rockets eventually found their way into common use in European military armaments in the early 1800s thanks to a man by the name of William Congreve. Congreve developed a strong interest in fireworks and rockets, and used family connections to get access to firing ranges where he could test out the rockets he was developing. He eventually developed four rockets for military use ranging from a light three pounds to a noticeably larger twenty-four pound warheads. After an uphill political battle, Congreve eventually found favor with some of England's military leaders and his rockets were used in several battles against Napoleon and the United States. After these engagements, the use of rockets as weapons gained a significant amount of popularity and quickly spread into the militaries of other major Europeans powers and then into Russia [18], [192].

While it appears most of these rockets consisted of a single stage, the plans for multi-stage rockets have appeared in books and letters for several centuries. In 1650, the Polish general Kazimierz Siemienowicz published *The Complete Art of Artillery* (also called *The Great Art of Artillery*), and inside was an early concept of multi-staged rockets [192], [193], [194]. Several two-stage rocket designs were also published in 1741 by Francois Freizer. These designs bear particular historical significance because of the inspiration they provided to one Edward Boxer in 1855. At the time, rockets with ropes on them, called "rescue rockets" were being used as a means to rescue sailors from sinking ships. Boxer dramatically improved rescue rockets with inspiration from Freizer's plans, and these improved rescue rockets eventually saw widespread use in Britain [192].

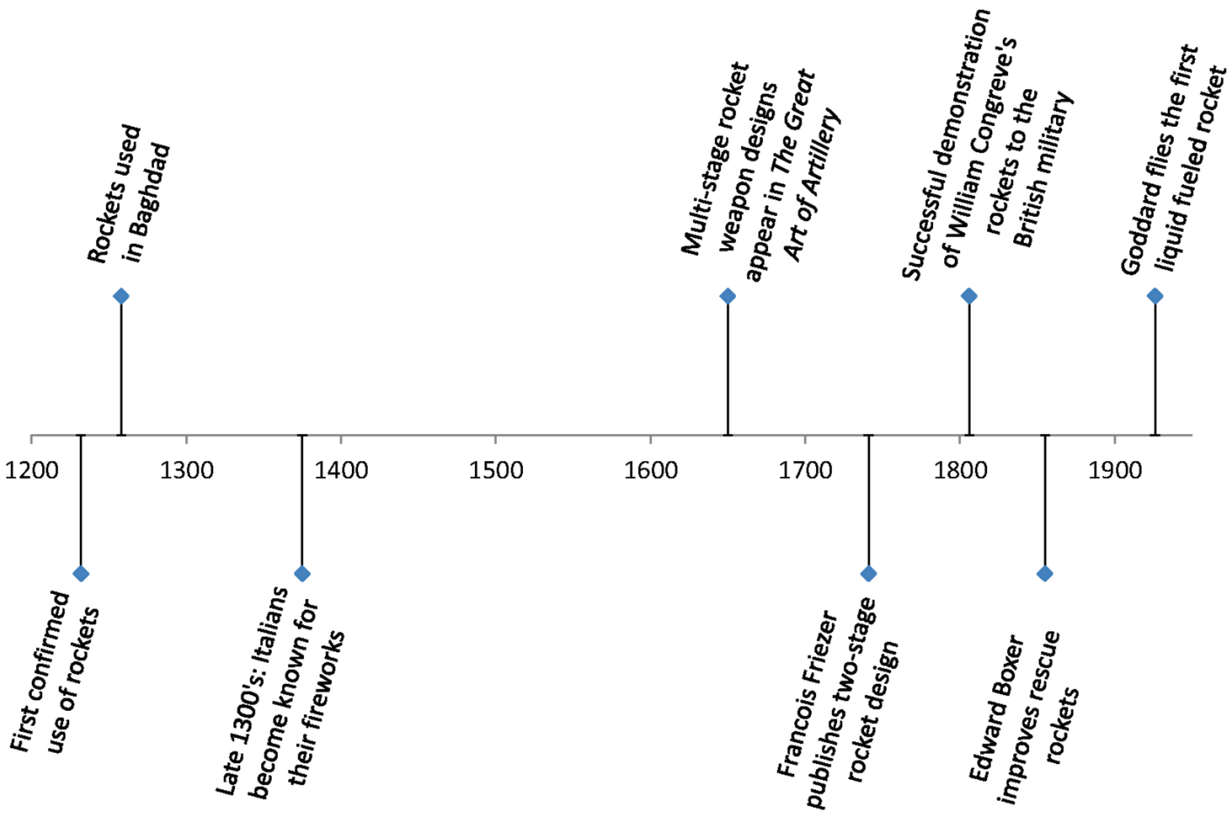

Figure A.1. Timeline of Early Rocket History

A.2 The Fathers of Modern Rocketry

Modern engineering requires a team-based approach. No single individual can complete massive projects like those that took mankind to the moon and established semi-permanent space stations in orbit around the planet. However, sometimes a lone individual, or series of individuals, instigate the creative process and cause the ripples which lead to such amazing feats.

While there are numerous individuals and teams which helped bring mankind to its current spacefaring state, history tends to recognize three individuals as being the fathers of modern rocketry: Konstantin Tsiolkovsky, Robert Goddard, and Hermann Oberth.

Konstantin Tsiolkovsky

Konstantin Tsiolkovsky (1857 – 1935) was born in a village near Moscow. Although his formal education ended in his early teenage years, Tsiolkovsky was a passionate individual who was self-taught in mathematics, astronomy, physics, and chemistry through studying in libraries, attending public lectures, and performing his own experiments. Having been inspired by the works of Jules Verne, Tsiolkovsky dedicated himself towards the goal of space access [18].

While it was not widely known for some time, Tsiolkovsky is the first person recorded as proposing the use of rockets to access space in his work "Investigating Space with Reaction Devices", which was finished in 1898 but not published until 1903 in *Scientific Review* [72]. In *To a Distant Day*, Chris Gainor explains

> *This work, one of the greatest foundations of astronautics, contained the first proposal for the use of liquid-fueled rockets as devices for venturing into space, including rockets that used the combinations of liquid hydrogen and liquid oxygen, kerosene and liquid oxygen, alcohol and liquid oxygen, and methane and liquid oxygen* [18]*}.*

"Investigating Space with Reaction Devices" contained far more than just the proposal of using rockets to reach space: Tsiolkovsky also mathematically explained the speeds that would be required, how these liquid propellants could provide the power to generate these speeds, and much more. Although he initially disregarded staging, Tsiolkovsky embraced the idea in his later works as a means to help shed dead weight to make the rocket-powered vehicle something that could be feasibly built [195], [18].

Unfortunately "Investigating Space with Reaction Devices" was not widely circulated due to the Russian government at the time confiscating that issue of *Scientific Review* due to the contents of another article. Furthermore, because Tsiolkovsky self-published most of his work and Russia's turmoil in the early 1900's isolated a majority of the country from the rest of the world most of his work remained in relative obscurity. It wasn't until Goddard and Oberth began receiving international recognition that the Soviet government began re-publishing Tsiolkovsky's works and he gained international recognition [18].

A complied collection of Tsiolkovsky's work may be found in NASA Technical Translation F-15571 (see reference [72]).

Goddard

Robert Goddard (1882 – 1945) is known today as the "father of modern rocket propulsion". Similar to Tsiolkovsky, Goddard read Jules Verne's books on traveling to space at a young age and was enamored by the idea [196], [192].

While he was studying at Worcester Polytechnic Institute in 1909, Goddard realized that rockets were the only practical means of transportation to space. In 1914 he began working on solid-fueled rockets, but by 1919 he had made the switch to liquid propellants. It was in that year that he completed "A Method of Reaching Extreme Altitudes", which was published by the Smithsonian the following year [73]. Goddard received a significant amount of negative responses to the idea of leaving Earth by using rockets. While this response darkened his view on the press, it did not deter him in his research. Goddard completed the design and fabrication of the first liquid fueled rocket, and successfully launched it on March 16, 1926, reaching an altitude of 41 feet and traveling 184 feet downrange in approximately 2.5 seconds. This rocket can be seen in Figure A.2 [18], [192].

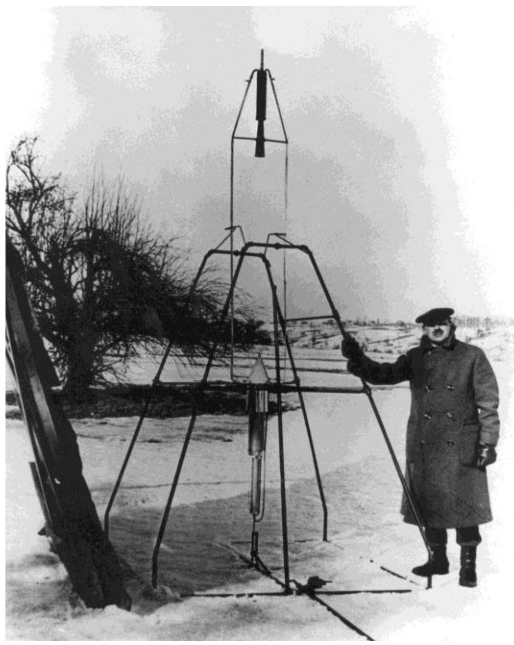

Figure A.2. Goddard next to his first liquid-fueled rocket [196]

In the nearly two decades between this flight and his death, Goddard gathered a team and together they continued his research into rocketry. His rocket designs evolved to the modern shape seen today, as is evident by the updated version of his work was published by the Smithsonian as "Liquid-propellant Rocket Development" in 1936 [197]. Additionally, he explored ideas of staged rockets, exotic propulsion sources such as solar, ion, and electric propulsion, aerobraking, manned space missions, and using cameras on spacecraft [18], [192].

A collection of Goddard's papers, diaries, notebooks, and newspaper clippings, including his famous paper "A Method of Reaching Extreme Altitudes", are hosted online and may be found in references [198] and [73].

Oberth

Hermann Oberth (1894 – 1989) is another remarkable figure in the history of modern rocketry. While Oberth was inspired by the fictional works of Jules Verne at a young age like both Tsiolkovsky and Goddard, he initially chose to study medicine at the University of Munich. His love with space travel never ceased though, and during his time as a field medic in World War I he designed a rocket that could carry an explosive warhead for the German military. The design was ultimately rejected but was the start of Oberth's shift into rocketry [199], [18].

After World War I ended, Oberth moved to Gottingen University and then later to the University of Heidelberg. During these years he studied mathematics and physics, and began work on a two-stage rocket design. He wrote a dissertation on rockets and space travel, but his work was rejected. Oberth decided to get the work published as a book, and in 1923 was successful in doing so although he had to pay the printing cost himself. The ninety-two page book titled *Die Rakete zu den Planetenraumen* (*By Rocket to Interplanetary Space*) \ [74] contained technical information on rockets and space travel, a two-stage liquid-fueled rocket design, a section describing an interplanetary spacecraft, and an appendix dedicated to Goddard's work which he had received during correspondence with Goddard not long before publishing the book [199], [18].

Oberth continued his work in the following decades and launched his first rocket in May 1931. He worked with a number of notable figures, including Wernher von Braun who was one of Oberth's assistant. Today Oberth is remembered for performing research in rocketry and space travel similar to Tsiolkovsky and Goddard without being aware of the details of their works and for spreading interest in manned space travel [199], [18].

Appendix B. Aristo Martin Space Slide Rule

In reference [83] Wood cites three references at the end of his chapter on staging and sizing. The second reference states

> 2. Stoiko and Wurth, Space Rule Handbook, published in 1960 by Martin Co., and intended for inclusion in a later book by Stoiko not known to be published in 1964.

This author of this thesis tried without success to find this "Space Rule Handbook" by Martin Co. and the book that was supposed to be published by Stoiko, but did discover images from the Autumn 2007 edition of *Slide Rule GAZETTE* [200] on a website for slide rules. In this issue Rod Lovett explains a little about the slide Martin Space Rule and provides several worked examples. He writes:

> To my knowledge only one [slide] rule was designed specifically for solving problems that are frequently encountered in several spaceflight technological areas. This rule, the Aristo 80123, was designed by engineers at Martin's Space System Division (now part of Lockheed Martin) and manufactured by Aristo in 1962.
>
> Now, of course, replaced by computers, this rule enabled the user to determine very rapidly individual booster stage sizes and takeoff weights of single- and multi-stage boosters, as well as mission velocity requirements and associated flight parameters for many ballistic, orbital and interplanetary problems.

Lovett provides some images of the slide rule which are reproduced in Figure B.1 through Figure B.3. Interestingly the slide rule here has a copyright date listed on it of 1962 while Wood mentions the handbook was published internally at Martin in 1960.

While this did not add anything to the work presented in the thesis, it is an interesting historical note and shows that launch vehicle sizing resources and tools came in a number of different forms over the past century. It also shows, unsurprisingly, that engineers at Martin put together enough information on the sizing of space launch vehicles to make their own sizing methodology and from that develop a sizing tool.

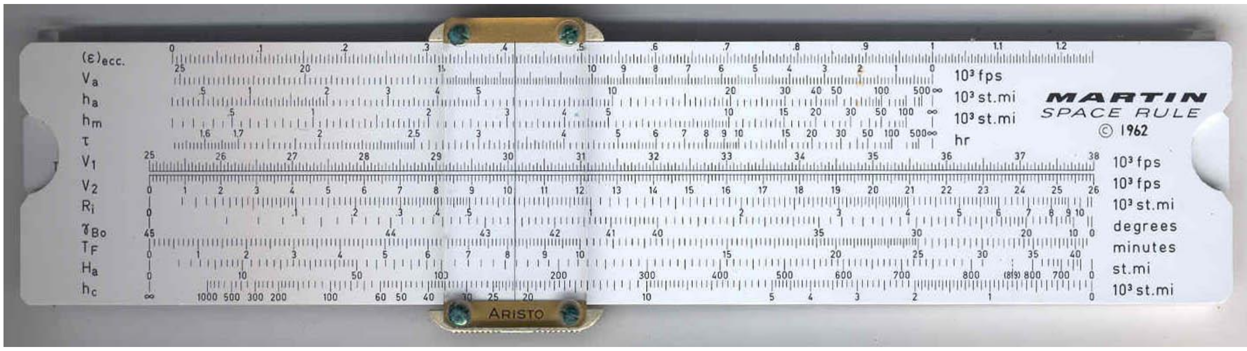

Figure B.1 Martin Slide Rule Front [200]

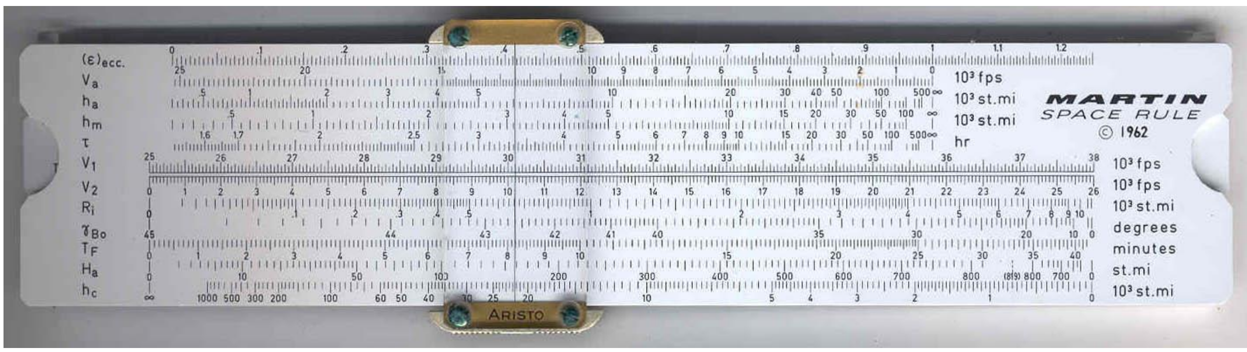

Figure B.2 Martin Slide Rule Back [200]

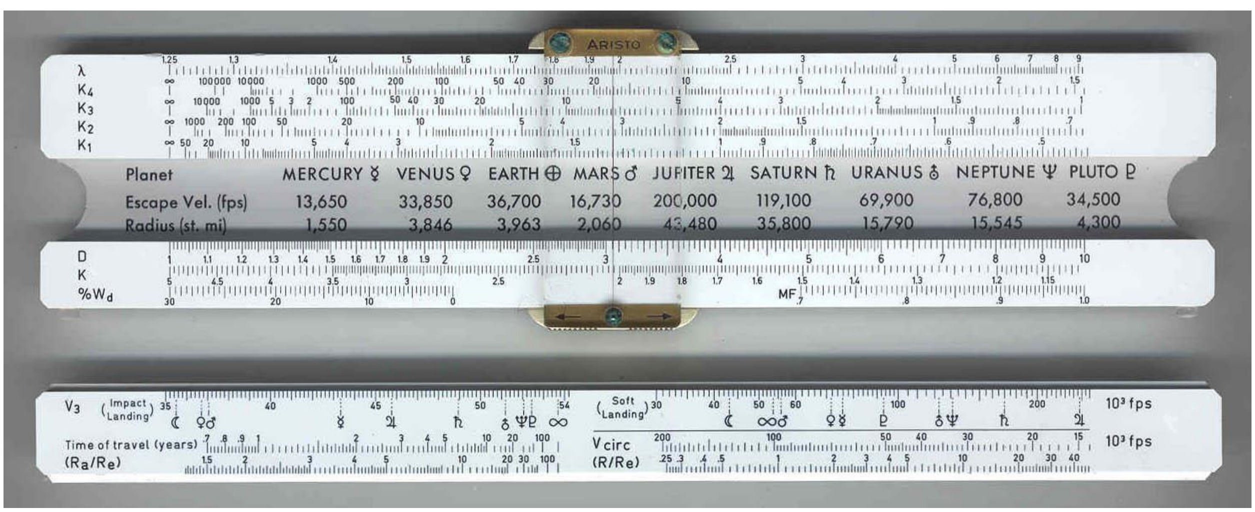

Figure B.3 Martin Slide Rule Gutter [200]

Appendix C. Launch Vehicle Design Reference Table

Table C.1. List of launch vehicle design literature found during this thesis

Source Full Name	Author/ Developer/ Lead Editors	Book/ Paper/ Thesis/ Other	Year Published	Method (Analytic, Graphical, Empirical)	# of Y's	# of N's	# of Discussion or Perform.
Design Methodologies for Space Transportation Systems [98]	Walter E. Hammond	Book	2001	Analytic	6	4	4
Aerospace Vehicle Design, Volume II: Spacecraft Design [83]	K. D. Wood	Book	1964	Both Analytic and Graphical	7	4	3
Space Planner's Guide [99]	U.S. Air Force (Harney)	Book	1965	Graphical	7	7	0
Space Propulsion Analysis and Design [201]	Ronald W. Humble; Gary N. Henry; Wiley J. Larson	Book	1995	Analytic	5	9	0
Handbook of Astronautical Engineering [69]	Heinz Hermann Koelle, et. al.	Book	1961	Both Analytic and Graphical	6	7	1
Introduction to Space Flight [202]	Francis J. Hale	Book	1994	Analytic	2	6	1
Fundamentals of Space Systems [203]	Pisacane, et. Al.	Book	2005	Analytic	4	7	2
Aerospace Vehicle Design [204]	Gerald Corning	Book	1964	Analytic	5	9	0
Manned Spacecraft: Engineering Design and Operation [119]	Purser; Faget; Smith	Book	1964	Analytic	3	6	5
Integrated Design for Space Transportation System [11]	Suresh; Sivan	Book	2015	Analytic	5	6	3
Astronautics [85]	Ulrich Walter	Book	2008	Analytic	4	9	1
Elements of Rocket Propulsion [65]	Sutton; Biblarz	Book	2010 (8th ed.)	Analytic	3	10	1

Table C.1. List of launch vehicle design literature found during this thesis (cont.)

Source Full Name	Author/ Developer/ Lead Editors	Book/ Paper/ Thesis/ Other	Year Published	Method (Analytic, Graphical, Empirical)	# of Y's	# of N's	# of Discussion or Perform.
Spacecraft Systems Engineering [205]	Peter Fortescue	Book	2003	Analytic	4	7	3
Orbital Mechanics: Theory and Applications [75]	Tom Logsdon	Book	1998	Analytic	4	10	0
Space Mission Analysis and Design [116]	Wiley Larson	Book	1999	Analytic	4	6	3
Elements of Space Technology [70]	Rudolf X. Meyer	Book	1999	Analytic	3	10	1
Flight Performance Handbook for Powered Flight Operations [86]	Joseph White	Book	1962	Both Analytic and Graphical	5	9	0
Design Guide to Orbital Flight [206]	Jensen, et. al.	Book	1962	Both Analytic and Graphical	4	9	1
Flight Performance Handbook For Orbital Operations [207]	Wolverton, et. al.	Book	1961	Both Analytic and Graphical	4	10	0
Launch Systems [208]	John Keesee	Lecture	2003	Analytic	1	13	0
Launch Vehicle Design lectures [209]	Robert Stengel	Lecture	2016	Analytic	2	11	1
Rocket Propulsion class notes (MIT course # 16.512) [210]	Manuel Martinez-Sanchez	Lecture	2005	Analytic	4	9	1
Space Flight Handbook - Volume 2 - Lunar Flight Handbook Part 3 - Mission Planning [211]	Martikan, F.	Book	1963	Analytic	2	12	0
Space Vehicle Design [212]	Michael D. Griffin; James R. French	Book	2004	Analytic	3	10	1
The Standard Handbook for Aeronautical and Astronautical Engineers [213]	Jerry Jon Sellers	Book	2002	Analytic	1	12	1

Table C.1. List of launch vehicle design literature found during this thesis (cont.)

Source Full Name	Author/ Developer/ Lead Editors	Book/ Paper/ Thesis/ Other	Year Published	Method (Analytic, Graphical, Empirical)	# of Y's	# of N's	# of Discussion or Perform.
Understanding Space: An Introduction to Astronautics [214]	William J. Astore; Robert B. Giffen; Wiley J. Larson	Book	2004	Analytic	2	11	1
Modern Engineering for Design of Liquid-Propellant Rocket Engines [215]	Dieter K. Huzel; David H. Huang	Book	1992	Analytic	2	12	0
AIAA Aerospace Design Engineers Guide [216]	AIAA	Book	2012	Analytic	3	10	1
Georgia Tech Lecture Series on Space Design [9]	Alan Wilhite	Lecture	Unknown	Analytic	5	6	3
University of Munich Lecture Series [217]	Unknown	Lecture	2011	Analytic	2	11	1
University of Maryland Lecture Series [100], [218]	David Akin	Lecture	2016	Analytic	7	5	2
Space Transportation Analysis and Design [219]	R. Hartunian and B. Boss	Technical Report	1993	Analytic	4	9	1
Preprints of Papers to be Presented at the Unguided Rocket Ballistics Meteorology Conference [220]	Multiple	Conference	1967	Analytic	2	12	0
A Method of Reaching Extreme Altitudes [73]	Robert H. Goddard	Technical Report	1919	Analytic	1	13	0
A Study of Air Launch Methods for RLVs [221]	Marti Sarigul-Klijn; Nesrin Sarigul-Klijn	Technical Report	2001	Analytic	1	12	1
An Approach for Calculating the Cost of Launch Vehicle Reliability [222]	Zachary C. Krevor; Alan Wilhite	Technical Report	2007	Analytic	1	9	4
An Overview of the Characterization of the Space Launch Vehicle Aerodynamic Environment [223]	John A. Blevins; John R. Campbell, Jr.; David W. Bennett; Russ D. Rausch; Reynaldo J. Gomez; Cetin C. Kiris	Technical Report	2014	Analytic	0	12	2

Table C.1. List of launch vehicle design literature found during this thesis (cont.)

Source Full Name	Author/ Developer/ Lead Editors	Book/ Paper/ Thesis/ Other	Year Published	Method (Analytic, Graphical, Empirical)	# of Y's	# of N's	# of Discussion or Perform.
Applying Monte Carlo Simulation to Launch Vehicle Design and Requirements Verification [224]	John M. Hanson; Bernard B. Beard	Article	2012	Analytic	1	13	0
Ascent Optimization for a Heavy Space Launcher [225]	C. Ponssard; K. Graichen; N. Petit; J. Laurent-Varin	Conference	2009	Analytic	2	12	0
Bio-Inspired Computing for Launch Vehicle Design and Trajectory Optimization [226]	Sundaram Suresh; Hai-Jun Rong; Narasimhan Sundararajan	Conference	2009	Analytic	4	10	0
Capabilities and Applications of the Program to Optimize Simulated Trajectories (POST) [227]	G. L. Brauer; D. E. Cornick; R. Stevenson	Manual	1977	Analytic	1	13	0
Comparison of Space Launcher Equations of Translation Using a Full Quaternion [228]	A. Vachon; A. Desbiens; E. Gagnon; C. Berard	Technical Report	2001	Analytic	1	13	0
Cost Estimates of Near-Term, Fully Reusable Space Access Systems [229]	James Michael Snead	Technical Report	2006	Analytic	1	13	0
Cost Estimating Module, Space Systems Engineering, version 1.0 [230]	Lisa Guerra	Other	2008	Analytic	0	13	1
Design Objectives for a Commercial Launch Vehicle with Integrated Guidance System [231]	Nickolay Mykola Zosimovych; Anatoly Voytsytskyy	Article	2014	Analytic	1	13	0
Development Of A Mass Estimating Relationship Database For Launch Vehicle Conceptual Design [232]	R. Rohrschneider	Technical Report	2002	Analytic	1	13	0
Economic Model of Reusable vs. Expendable Launch Vehicles [233]	James R. Wertz	Conference	2000	Analytic	1	13	0

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Cost Estimating Module, Space Systems Engineering, version 1.0 [230]	Lisa Guerra	Other	2008	Analytic	0	13	1
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Source Full Name	Author/ Developer/ Lead Editors	Book/ Paper/ Thesis/ Other	Year Published	Method (Analytic, Graphical, Empirical)	# of Y's	# of N's	# of Discussion or Perform.
Efficient Solid Rocket Propulsion for Access to Space [234]	Filippo Maggi; Alessio Bandera; Luciano Galfetti; Luigi T. De Luca; Thomas L. Jackson	Article	2010	Analytic	0	14	0
Electronic Propulsion Upper-Stage for Launch Vehicle Capability Enhancement [235]	Gregory E. Kemp; John W. Dankanich; Gordon R. Woodcock; Dennis R. Wingo	Technical Report	2007	Analytic	1	12	1
Estimating the Life Cycle Cost of Space Systems [236]	Harry W. Jones	Conference	2015	Analytic	1	12	1
Alternate Vehicles for Engine/Vehicle Optimization [237]	G. Venkatasubramanyam; James A. Martin	Conference	1993	Analytic	3	11	0
Future low-cost space transportation system analysis [124]	Dietrich E. Koelle; H. Hermann Koelle	Article	1979	Analytic	2	12	0
Launch Vehicle Design Features for Minimum Cost	Ron Portz	Conference	2004	Analytic	1	12	0
Multi-criteria genetic optimisation of the manoeuvres of a two-stage launcher [238]	Fernado Alonso Zotes; Matilde Santos Penas	Article	2010	Analytic	2	12	0
Multidisciplinary Aerospace System Design: Principles, Issues and Onera Experience [239]	Defoort, at. al.	Article	2012	Analytic	1	13	0
Optimal Conceptual Design of Two-Stage Reusable Rocket Vehicles Including Trajectory Optimization [240]	Takeshi Tsuchiya; Takashige Mori	Article	2004	Analytic	1	13	0

Table C.1. List of launch vehicle design literature found during this thesis (cont.)

Source Full Name	Author/ Developer/ Lead Editors	Book/ Paper/ Thesis/ Other	Year Published	Method (Analytic, Graphical, Empirical)	# of Y's	# of N's	# of Discussion or Perform.
Optimum Stage-Weight Distribution of Multistage Rockets [241]	J. J. Coleman	Technical Report	1960	Analytic	2	12	0
Propulsion and Staging Considerations for an Orbital Sortie Vehicle [242]	Robert F. Stengel	Article	1987	Analytic	1	13	0
Simple mass and size estimation relationships of pump fed rocket engines for launch vehicle conceptual design [243]	B. T. C. Zandbergen	Article	2015	Empirical	0	14	0
Some Considerations for the Preliminary Structural Design of Liquid-Fueled Boosters [244]	C. J. Meissner	Conference	1960	Analytic	1	13	0
Space Technologies, Volume I: Spacecraft System [245]	L. H. Abraham	Technical Report	1965	Analytic	4	10	0
Verifying Launch Vehicle Conceptual Designs Using First Principles and Historical Trends [246]	J. Simmons; J. Black	Technical Report	2011	Analytic	2	12	0

Appendix D. Stage Diameter vs. Length Trade

As mentioned in Chapter 3.9, the stage diameter calculated by the system is simply the minimum diameter based on the vehicle's payload and propulsion requirements, and may not be the optimal diameter for the vehicle. There can be many reasons for wanting a stage diameter larger than what is required. This Appendix explores the trade of the stage's diameter vs. the stage's length for an equal volume requirement.

The volume of a cylinder is

$$V_{cylinder} = \frac{\pi * d^2 * l}{4} \tag{D.1}$$

If there are two cylinders of equal volume and the diameter of the second cylinder is the diameter of the first cylinder plus some number, then

$$d_2 = d_1 + x \tag{D.2}$$

and

$$V_1 = V_2$$

$$\frac{\pi * d_1^2 * l_1}{4} = \frac{\pi * d_2^2 * l_2}{4}$$

$$\frac{\pi * d_1^2 * l_1}{4} = \frac{\pi * (d_1 + x)^2 * l_2}{4}$$

$$d_1^2 * l_1 = (d_1 + x)^2 * l_2$$

$$l_2 = \frac{d_1^2}{(d_1 + x)^2 * l_1} \tag{D.3}$$

As can be seen in Equation (D.3), increasing the diameter has a significant effect on the length. As an example, the Saturn V's second stage, the Saturn-II, has a propellant volume requirement of 1,329 m^3. If the diameter was set at 6 m, this would turn into a length requirement of 47.00 m for the propellant tank stack. If the diameter grows to 7 m, the tank stack length drops to 34.53 m, a decrease of 26.5%. If the diameter increases to 8 m, the tank stack decreases further to 26.44 m, 23.4% lower the length a 7 meter diameter and 43.7% from the 6 meter diameter. Results for a diameter from 6 meters to 11 meters is displayed in Figure D.1 and Table D.1 Diameter vs. Length Trade. As seen in the plot, the affect decreases exponentially as the stage diameter continues to increase. Note that the table assumes the tanks are a cylinder with flat ends; the length would increase slightly when the hemispherical tanks are taken into account.

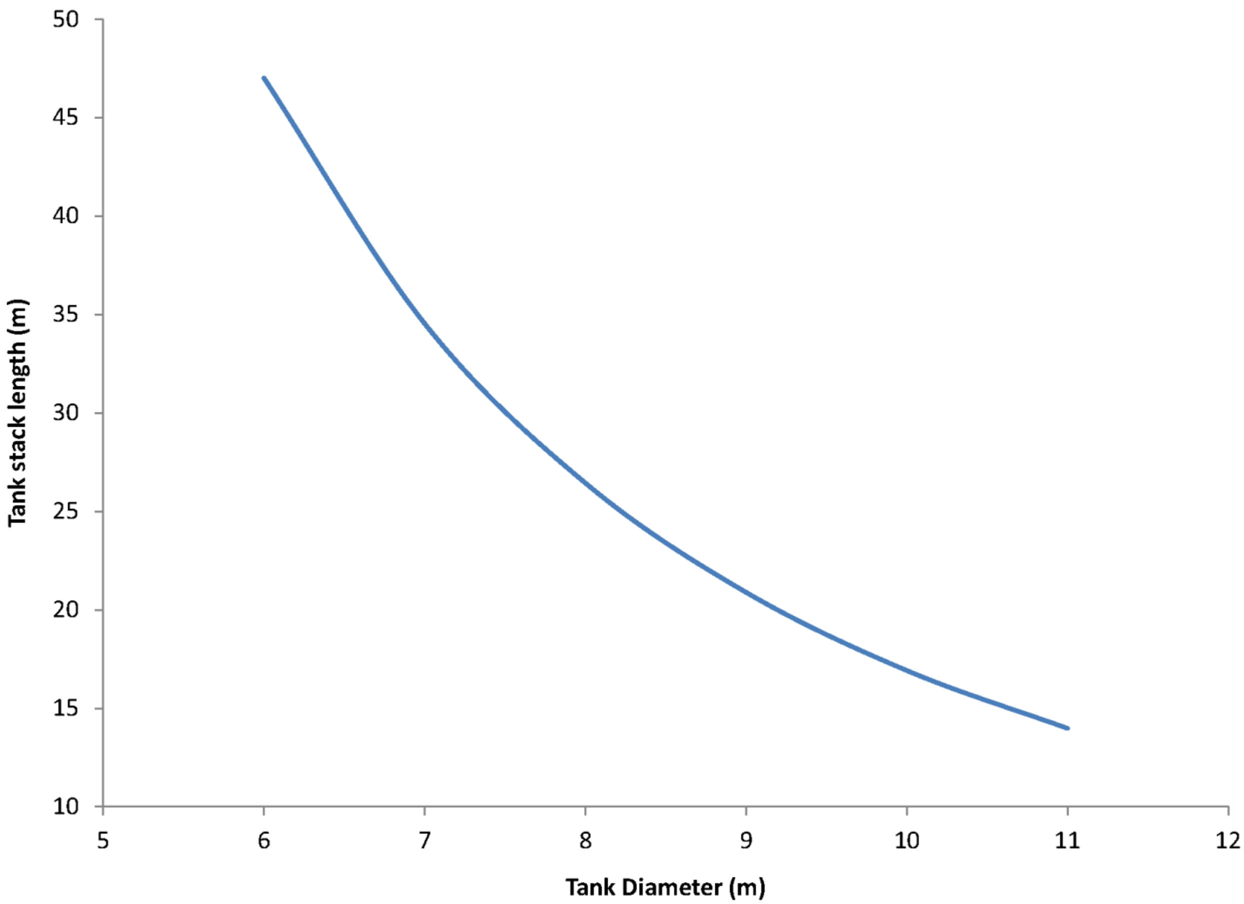

Figure D.1 Diameter vs. Length Trade

Table D.1 Diameter vs. Length Trade

Volume Requirement	1,329	1,329	1,329	1,329	1,329	1,329
Diameter	6	7	8	9	10	11
Length	47.00	34.53	26.44	20.89	16.92	13.98

Appendix E. WYr Conversion Table

The table below is a copy of Table 1-V from the *Handbook of Cost Engineering* [125] and is used to convert from WYr to USD, Euros, or Yen.

Table E.1 WYr to Currency Conversion

Year	USA*) US$	Europe**) Euro (ECU/AU)	Japan***) Million Yen
1961	27000	18900	-
1962	28000	20000	-
1963	29000	21000	-
1964	30000	22000	-
1965	31000	23200	-
1966	32300	24400	-
1967	33200	25700	-
1968	34300	27400	-
1969	36000	29100	-
1970	38000	31000	-
1971	40000	33050	-
1972	44000	35900	-
1973	50000	38700	-
1974	55000	43600	-
1975	59500	50000	-
1976	66000	55100	-
1977	72000	60500	-
1978	79700	65150	-
1979	86300 #	71800	-
1980	92200	79600	-
1981	98770	86700	-
1982	105300	92400	-
1983	113000	98300	-
1984	120800	104300	14.6
1985	127400	108900	15.2
1986	132400	114350	15.8
1987	137700	120000	16.4
1988	143500	126000	17.1
1989	150000	133000	17.6

Table E.1 WYr to Currency Conversion (cont.)

Year	USA*) US$	Europe**) Euro (ECU/AU)	Japan***) Million Yen
1990	156200	139650	18.1
1991	162500	145900	18.6
1992	168200	151800	19.0
1993	172900	156800	19.5
1994	177200	160800	20.0
1995	182000	167300	20.5
1996	186900	172500	21.0
1997	191600 ##	177650	21.5
1998	197300	181900	22.0
1999	203000	186300	22.6
2000	208700	190750	23.2
2001	214500	195900	23.8
2002	222600	201200	24.4
2003	230400	207000	25.0
2004	240600	212800	25.6
2005	250200	219200	26.3
2006	259200	226300	26.9
2007	268800	234800	27.5
2008	278200	243600	28.2
2009	286600	252700	29.0
2010	296000	261000	28.9
2011	303400	268800	30.4
2012	312000	275500	31.2
2013	320000	285000	32.0
2014	328700	292400	32.8
2015 est.	337100	301200	33.6

*) Established with application of NASA's official annual cost escalation factors
**) Based largely on official ESA annual cost growth values, valid for aerospace Industry in West-Europe
***) Basis data from "The Aerospace Industry Year Book", Society of Japanese Aerospace Companies
= USAF Reference Value
= NASA 1997 effective WYr cost from 533 contracts

Appendix F. User's Guide

As discussed in Chapter 3, the software is broken up modules, and each individual module performs one part of the sizing. The modules are connected in a single python file which calls and loops through modules as necessary in order to perform the sizing process. For example, "sizeExpendable.py" is the main module used for sizing fully expendable launch vehicles without transfer stages. Someone who wishes to size a new launch vehicle should not access use this but instead run "sizeExpendableSingleSize.py" or "sizeExpendableMulti.py".

The first step to sizing a new vehicle is to create the input file. The input file contains all of the variables necessary to size a specified vehicle configuration. These variables are arrays if the information relates to more than one stage (for example, the stage's $I_{sp,\ mean}$) or floats if it doesn't (for example, the target orbit altitude). The tables in this Appendix detail the required and optional inputs in order to size a launch vehicle using the software.

To size a single vehicle with this input file, "sizeExpendable.py" needs updated to read the new file. This is done by changing the input file names on lines 42 and 43. No further changes are necessary to run the software.

Sizing multiple vehicles in a single run is more complex. "sizeExpendableMulti.py" is the file used in order to size multiple launch vehicles. This file is made up of two functions. In order to use the software to run multiple vehicle configurations in a single run, changes are needed to both functions.

The main function is where the user will input the variables they wish to vary during the sizing process. There are three steps to adding or removing a variable from the list:

1. Create a new array to hold the range of values to vary.

2. Update the variable "allLength" with the length of the newly created array, and create an empty array with a name based on variable being varied and a length equal to the length of "allLength". As an example, the author used "all_mPayload" when varying payload capacity.

3. Update the loop which assigns all of the variables to be varied. In the last iteration of this software, the loop looked like Figure F.1. The loop will need to be updated with:

    a. An outer for loop based on the new variable that is being varied;
    b. An additional term to determine the index number used;
    c. The newly created "all_..." array which will hold all of the new variables used.

When this loop runs, it will assign every possible configuration into the "all_..." Each index number corresponds to a single configuration.

```
    for mPayloadIndex in range(len(m_payload)):

        for dVIndex in range(len(delta_V_fraction)):

            for seTwoIndex in range(len(pi_seTwo)):

                for seOneIndex in range(len(pi_seOne)):

                    #Don't forget to update index as well.  The general formula is
                    #   index of the first + sum of (next level index * product of all lengths below it).
                    #   I.e., for 3 for loops, it's
                    #   index of innerMostLoop + (indexMiddleLoop * len(innerMostArray)) + (indexOuterLooper * [len(innerMostArray) * len(middlerArray)])
                    #   BUT remember some variables are dependent on each other, like delta V fraction. In this case you should only have 1 for loop for them and 1 set of (indexLoop * ...))
                    index = (
                            (seOneIndex) +
                            (seTwoIndex * len(pi_seOne)) +
                            (dVIndex * len(pi_seOne) * len(pi_seTwo)) +
                            (mPayloadIndex * len(pi_seOne) * len(pi_seTwo) * len(deltaVOne))
                            )


                    all_piSeOne[index] = pi_seOne[seOneIndex]
                    all_piSeTwo[index] = pi_seTwo[seTwoIndex]
                    all_dVOne[index] = deltaVOne[dVIndex]
                    all_dVTwo[index] = 1 - deltaVOne[dVIndex]
                    all_mPayload[index] = m_payload[mPayloadIndex]
```

Figure F.1 Snapshot of the Loop Used to Assign Variables

Under this for loop is a variable called "params". This variable takes all of the data to be varied and compacts it in order to send it to the second function. "params" also needs updated with the new variable new.

The second function is what actually runs the sizing code. It takes "params" from the main function and unzips it into the variables the user wants to vary. The function then loads the data from the input file. Each variable of the variables that got unzipped from params then needs to be assigned to the correct variable name from the input file in order to overwrite it with the correct value. Both unzipping information from "params" and overwriting the input variables must be updated manually by the user.

Once the data has been overwritten, the function runs the sizing code. Once every vehicle configuration has been sized, the resulting data is compiled and passed back to the main function as a single variable. The main function unpacks this information and assigned the resulting vehicle and trajectory data to the Ordered Dictionaries. The vehicle and trajectory data are then passed to the Output Module, and everything is written to Excel.

It is important to note that sizing a large number of vehicle configurations can take a significant amount of time. In particular, sizing a first stage boostback vehicle can range anywhere from seventeen seconds to five minutes or more depending on the provided inputs values. The software is programmed to determine how many CPU cores are available and utilize all of them, so before running multiple configurations be sure all other work is saved and the computer is not needed for any other tasks. **Be sure to also set the computer to not sleep or power off if no input is provided.**

Table F.1 Required Inputs

Variable Name	Description
h_launch	Altitude of the launch site (m)
latitude	Latitude of the launch site (deg)
alt_orbit	Altitude of the parking orbit (m)
m_payload	Mass of the payload (kg)
d_payload	Diameter of the payload interface (m)
l_payload	Length of the payload (m)
delta_V_losses	Initial estimate for the delta V losses in (m/s)
n_stages	Desired number of stages
delta_V_fraction	Split of the delta V required among the different stages. The sum of these values should equal one
pi_se	Fractional mass of the structure and equipment
Isp_mean	Average specific impulse a stage will experience during flight (s)
T_eng	Thrust per engine (N)
Throttle	How low the engine can be throttled to. If the engine cannot be throttled, the value equals 1
m_eng	Mass of a single engine (kg)
l_eng	Length of a single engine (m)
A_e	Exit area of an engine's nozzle (m^2)
engGap	Required spacing between engines (m)
FTO	Fuel-to-oxidizer ratio
rho_oxidizer	Density of the oxidizer in kg/m^3
rho_fuel	Density of the fuel in kg/m^3
T_W_min	Minimum launch thrust-to-weight ratio
T_W_max	Maximum launch thrust-to-weight ratio
hturn	Altitude to begin first pitch-over maneuver (m)
qMax	Maximum allowable dynamic pressure (Pa)

Table F.1 Required Inputs (cont.)

Variable Name	Description
gMax	Maximum allowable g-load
X_CG_pLAlone	CG location for the payload from the base of the payload to the front end of the launch vehicle (m)
dFPA_plan	Array of values for $\dot{\gamma}$ (rad/s)
T_dFPA	Array of time values to begin dFPA_plan (s)
nLaunches	Number of launches per year
nVehiclesProduced	Number of vehicles produced
nAnnualVehicleProd	Annual vehicle production
nFlightsPerVehicle	Number of flights a vehicle performs. For expendable vehicles, this is equal to 1
f0_prod	System's engineering cost factor for production
f1	Development standard factor
f3	Team experience factor
stageLF_ForceLowerBound	Boolean to determine whether or not to use the lowest possible learning factor for stage production. This should only be set to "True" when the company has a strong focus on cost engineering
f4_ops	Learning cost factor for operations
relative_development_schedule	Percentage of how on time the project is. 100% corresponds to on time
f7	Program organization cost factor
f8	Country/region cost factor
f9	Subcontractor cost factor
f10	Past experience cost factor
f11	Commercial development cost factor
f_v	Launch vehicle type cost factor
f_c	Vehicle assembly and integrator mode cost factor
oxidizerCostPerKG	Cost of the oxidizer (WYr/kg)
fuelCostPerKG	Cost of the fuel (WYr/kg)
propellant_type	Type of propellant used (storable or cryo)

Table F.1 Required Inputs (cont.)

Variable Name	Description
caseLetter	Procurement cost case
outputFileName	Name for the Excel file to output all of the data to
runMassMarCheck	Boolean to determine whether or not to exclude data that fails the check from the Systems Mass Module
insurancePercentCost	Percent of the cost-per-flight to add for insurance
delta_V_addBurns	Additional ΔV to be used for transfer stages (m/s)
delta_V_bb_req	ΔV required for boostback maneuvers (m/s)
vehicleLifecycleLife	Years for which the launch vehicle that is developed will be flown for before the design is replaced (years)
bigX	Array of grid fin lengths to use for calculating the grid fin drag (m)
t_MAC	Thickness of a grid fin section at the mean aerodynamic chord (m)
c_MAC	Length of the mean aerodynamic chord (m)
sweep_LE	Sweep of the leading edge for the grid fin sections (rad.)
n_GF	Number of grid fins
b	Span of each grid fin section (m)

Table F.2 Optional Inputs

Variable Name	Description
t_fuel	Thickness of the fuel tank in m.
t_fuelInsul	Thickness of the fuel tank's insulation in m.
r_f_pipe	Radius of the pipe that runs from the fuel tank through the oxidizer tank, if applicable, in m.
t_oxidizer	Thickness of the oxidizer tank in m.
t_oxInsul	Thickness of the oxidizer tank's insulation in m.
l_stageGap	Spacing requirement between the top of a stage and the nozzle exit of the next stage

Appendix G. Programmer's Guide

As discussed in Chapter 3, the software is broken up modules, and each individual module performs one part of the sizing. The modules are connected in a single python file which calls and loops through modules as necessary in order to perform the sizing process. For example, "sizeExpendable.py" is the main module used for sizing fully expendable launch vehicles without transfer stages. As documented in the User's Guide, the user does not access or run anything from this file, but instead runs "sizeExpendableSingleSize.py" or "sizeExpendableMulti.py". Someone making modifications to the program, however, needs to access this.

The first step in designing new or improving existing modules is determining what specifically the modules do and where in the programming logic it should be placed. As an example, when creating the modules necessary for sizing first stage boostback vehicles, the NS diagram in Figure 3.1 was consulted. The new modules would need to:

1. Size the first stage to have propellant reserved for ascent-to-orbit and the boostback return;

2. Include landing hardware in the first stage's secondary structure and equipment mass estimate; and

3. Simulate a boostback return;

4. Output this data appropriately.

This means modifications needed made to the

1. Sizing Module;

2. Mass Module;

3. Trajectory Module; and

4. Output Module.

In order to keep each type of vehicle (fully expendable, fully expendable with transfer stage, and first stage boostback), this also means a new version of "sizeExpendable.py" and "sizeExpendableMulti.py" which calls the correct modules for the type of vehicle.

As a general rule, changes to the Output Module should be kept at a minimum. The Output Module is designed such that all of the vehicle sizing and trajectory data are passed to it in ordered dictionaries. An ordered dictionary is essentially a collection of variables that can be accessed using a keywords. The Output Module takes the ordered dictionary, run through each keywords in it, and writes the appropriate data to Excel. The module is setup such that keywords can be added or removed without needing to modify the module to recognize the keyword.

As an example, consider the possibility of adding in a Structures Module. One useful feature of such a module would be to estimate the maximum allowable dynamic pressure for the vehicle. This calculation would require information about the vehicle's geometry and would be utilized by the trajectory simulation, so it would be placed between the Geometry Module and the Trajectory.

Once the module has been created and inserted into "sizeExpendable.py", an update is needed to the ordered dictionary that is returned. First, an ordered dictionary should be created to store just the structures data like shown in Figure G.1.

```
structuresData = OrderedDict([
        ("Fuel Tank Thickness (m)" , fuelTankThick),
        ("Oxidizer Tank Thickness (m)", oxTankThick),
        ("Maximum dynamic pressure sustainable (Pa)", qMaxSus),
        ("Maximum dynamic pressure allowable (Pa)", qMaxAllow)
        ])
```

Figure G.1 Example Storage of Structures Data

The vehicleData ordered dictionary would then be updated to look like Figure G.2.

```
vehicleData = OrderedDict([
        ("Inputs" , inputData),
        ("Delta V outputs" , deltaVOutputs),
        ("Launcher sizing outputs" , launcherSizingOutputs),
        ("Propulsion module outputs" , propulsionModOutputs),
        ("Propellant sizing outputs" , propellantSizingOutputs),
        ("Geometry outputs"          , geometryOutputs),
        ("Structures outputs"        , structuresData),
        ("Mass module outputs"       , massModuleOutputs),
        ("Cost outputs"              , costOutputs),
        ("Stability data"            , stabilityData)
        ])
```

Figure G.2 Update to vehicleData

The Output Module would require no update as it would automatically read the new key in vehicleData and extract the information from structuresData appropriately.