NUMERICAL ANALYSIS OF THE EFFECTS OF ATMOSPHERIC PARAMETERS ON SPACE LAUNCH CENTER SAFETY

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**NUMERICAL ANALYSIS OF THE
EFFECTS OF ATMOSPHERIC PARAMETERS ON
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ABSTRACT

NUMERICAL ANALYSIS OF THE EFFECTS OF ATMOSPHERIC PARAMETERS ON SPACE LAUNCH CENTER SAFETY

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This thesis study aims at the problem of space launch center safety which is dependent on launch vehicle conceptual design and atmospheric parameters. Selection of the concept at the initial step of the satellite launch vehicle affects the project cost and space launch center safety directly. It is also important to model properties of atmospheric conditions and conduct safety analysis.

Successful mission for a launch vehicle depends on providing the total velocity requirement. In this study, launch vehicle performance that is necessary for a successful mission was modelled. Different launch vehicle configurations are compared using a parametric cost model and best alternative in terms of lifetime cost is selected to be used for the safety analysis of space launch center safety.

Atmospheric parameters, variation of temperature and pressure with altitude and winds with their deviations are modelled. Drop points of launch vehicle fragments under different atmospheric conditions are analyzed. In addition, seasonal variations of safety distance requirements due to secondary effects such as temperature, pressure and toxic pollutants are calculated.

Keywords: Launch vehicle, Launch center safety, Launch vehicle cost analysis, Synthetic wind profile, Safety distances

ÖZ

ATMOSFER PARAMETRELERİNİN UYDU FIRLATMA MERKEZİ GÜVENLİĞİNE ETKİLERİNİN NUMERİK ANALİZLERİ

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Bu tez çalışmasında uydu fırlatma sistemi konfigürasyonu ve atmosfer parametrelerine bağlı olan uydu fırlatma merkezi güvenliği ele alınmıştır. Uydu fırlatma sistemi projelerinin ilk adımı olan kavramsal tasarımda yapılan kavram seçimi proje maliyetine ve fırlatma alanı güvenliğine doğrudan etki etmektedir. Atmosfer koşullarının modellenmesi fırlatma alanı açısından ayrıca önemlidir.

Uydu fırlatma sistemlerinin görevlerini yerine getirebilmesi toplam hız ihtiyacının sistem tarafından sağlanabilmesine bağlıdır. Bu çalışmada görev için gereken uydu fırlatma sistemi performansı modellenmiştir. Fırlatma aracı alternatifleri parametrik bir maliyet modeli yardımı ile karşılaştırılmış ve toplam maliyet bakımından en iyi çözüm fırlatma merkezi güvenlik analizlerinde kullanılmak üzere seçilmiştir.

Atmosfer parametreleri, sıcaklık ve basıncın yüksekliğe bağlı değişimi ve rüzgarlar ile bunların ortalama değerlerinden sapmaları modellenmiştir. Kaza durumları için fırlatma aracı parçalarının farklı atmosfer koşullarındaki düşme noktaları incelenmiştir. Ayrıca sıcaklık, basınç ve toksik kirleticer gibi ikincil etkilerin mevsimsel koşullara bağlı güvenlik mesafeleri hesaplanmıştır.

Anahtar Kelimeler: Fırlatma aracı, Fırlatma alanı güvenliği, Fırlatma aracı maliyeti, Sentetik rüzgar profili, Güvenlik mesafeleri

To My Family

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Finally, I would like to thank my loved ones and friends, who have supported me throughout entire process and give me courage, both by keeping me cheerful and helping me going on. I will be grateful forever for your love and for the time spent together.

Ankara, January 2016

İbrahim Yaman

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Latin Letters

a	Semimajor axis
A	Azimuth
A_e	Nozzle exit area
A_w	Wind azimuth
b	Semiminor axis
CA, CC	Clocking angle
CMU, CMV	Conditional mean values of wind components
$CSDU$, $CSDV$	Conditional standard deviations of wind components
D	Drag force
e	Eccentricity
Ef	Average irradiance coefficient
f	Development cost degression factor
F	Thrust
$Fq(r)$	Angular irradiance coefficient
f_T, f_Y	Thrust and Side force
g_c,g_δ	Central and latitudinal component of gravity
g_0	Gravitational constant
i	Inclination
I_s	Specific impulse
m	Mass
M	Reference Mass
MR	Mass ratio
MU, MV	Mean values of wind components

p	Pressure or semilatus rectum (depending on context)
P_e	Static pressure at the nozzle exit
P_a	Ambient static pressure
$Q(r)$	Thermal radiation intensity
Q_m*	Mass of the released toxic material
r	Position
r_a	Apogee radius
r_p	Perigee radius
$R(U,V)$	Correlation coefficient of U and V
SDU, SDV	Standard deviations of wind components
SU, SV	Variances of wind components
SUV, SVU	Covariances of wind components
U	Zonal wind component
u_e	True exhaust velocity
v	Velocity
v_a,v_p	Apogee and perigee velocities
v_n,v_r	Normal and radial components of velocity
v_w	Wind velocity
V	Velocity or meridional wind component (depending on context)
V_c	Circular orbit velocity
V_e	Effective exhaust velocity
V_1, V_2	Required velocity increments
W	Wind speed

Greek Letters

α	Lagrange multiplier
β	Ballistic coefficient or conjugate of light path angle or launch azimuth or effective sideslip angle (depending on context)
γ	Flight path angle
δ	Geocentric Latitude

ΔV	Velocity change
ε	Structural coefficient or side slip angle (depending on context)
θ	True anomaly or Wind direction (depending on context)
λ	Payload ratio or geodetic longitude (depending on context)
μ	Earth's gravitational parameter or angle of attack (depending on context)
Ω	Right ascension of the ascending node
σ_x, σ_y, σ_z	Pasquill-Gilford dispersion coefficients
$\tau(r)$	Atmospheric transmission coefficient
ϕ	Flight path angle or launch site latitude (depending on context)
ω	Rotation rate of earth

Subscripts

0	Initial
f	Final
e	Exit or Exhaust
i	Initial
p	Propellant
pl	Payload
plf	Payload fairing
s	Structural
t	Total

LIST OF ABBREVIATIONS

AVUM	Attitude and Vernier Upper Module
CO	Carbon monoxide
DOC	Direct operations cost
FAA	Federal Aviation Administration
GEO	Geostationary orbit
GGUAS	Global Gridded Upper Air Statistics
GLOW	Gross lift-off weight
HCL	Hydrogen chloride
HTPB	Hydroxyl-terminated polybutadiene
ICAO	International Civil Aviation Organization
IDLH	Immediately Dangerous to Life or Health
IOC	Indirect operations cost
ISP	Specific impulse
LEO	Low earth orbit
LV	Launch vehicle
MEO	Medium earth orbit
NASA	National Aeronautics and Space Administration
NTO	Nitrogen tetroxide
TNT	Trinitrotoluen
TVC	Thrust vector control
UDMH	Unsymmetrical dimethyl hydrazine
US1976	The US Standard Atmosphere, 1976
WYr	Work year

CHAPTER 1

INTRODUCTION

In spaceflight, launch vehicle is a rocket powered vehicle which is used to carry a payload beyond the atmosphere. There are many launch vehicle examples which have been used to carry manned or unmanned spacecraft to suborbital or orbital trajectories. Atlas and Delta families of United States of America, Soyuz and Proton launchers of Russia and Ariane series and new launch vehicle VEGA of Europe are known examples.

A launch vehicle with or without boosters, which consists of single or multiple stages, must provide the necessary velocity increment to the spacecraft in order to carry it from launch site to the target orbit. Total required velocity consists of several components and it can be as high as many kilometers per second which requires controlled burn of tons of propellant carried on board.

Accessing space is an expensive task due to requirement of using highly technological equipment and materials. Even small launch vehicles which are only capable of carrying light payloads to orbit cost millions of dollars. Many countries have been carrying out independent space programs through government funding in order to have an independent space launch capability.

Most space launch vehicles trace their heritage to ballistic missiles developed for military use during the 1950s and early ’60s. Those missiles were based on the ideas first developed by Konstantin Tsiolkovsky from Russia, Robert Goddard from the United States of America, and Hermann Oberth from Germany. It was first recognized by Tsiolkovsky that rockets with separate stages are needed to achieve orbital velocities.

Although built as a weapon of Second World War, the German V-2 rocket with its technical name Aggregat-4 later served as the predecessor of some of the launch vehicles used in the early space programs. This liquid propellant rocket is the world’s first long range ballistic missile. During 1950s, Jupiter intermediate range ballistic missile was developed which was a derivative of V-2 rocket. Following Jupiter and first two intercontinental ballistic missiles Atlas and Titan of United States, five F-1 engine on Saturn V launch vehicle is used in Apollo Program to carry humankind to the Moon in 1969. A representative image of V-2 is given in Figure 1.

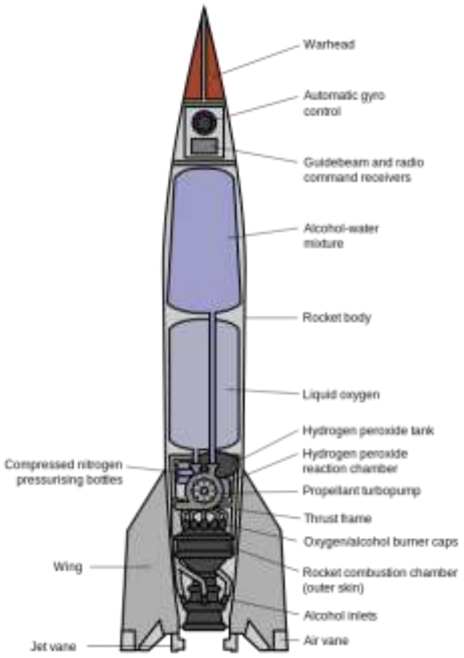

Figure 1 German V-2 Rocket Diagram (NASA)

Similarly in Soviet Union, an intercontinental ballistic missile R-7 was developed during the 1950s. Due to the fact that R-7 or Semyorka had a greater weight lifting capacity, Soviet Union had a significant advantage in terms of placing a payload into orbit. Sputnik 1 which is the first artificial Earth satellite carried by an unmodified R-7 in 1957. Also first human to Earth orbit, Yuri Gagarin, launched on a variant of R-7 (the Vostok) by Soviet Union in 1961. Another variant of this missile Soyuz was first used in 1966 and it is still operational with many improvements on the launch vehicle.

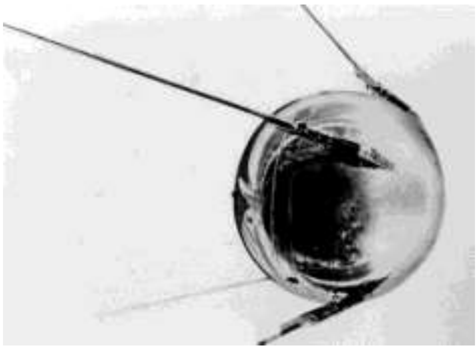

Figure 2 First Artificial Earth Satellite (Sputnik 1)

After the successful launch and operation of Sputnik 1 which was intended for atmospheric studies, application areas of satellites were extended. Sputnik 1 was followed by communication, reconnaissance, observation of Earth and outer space, global positioning and research satellites in addition to space stations. Due to increasing number of launches with different requirements in terms of carrying capacity, many launch vehicles were developed by different countries.

The only prevailing principle for the launch vehicle design was the maximum performance and the minimum weight. However, the new principle is to satisfy the mission requirements with the minimum cost. Design process of new launch vehicles becomes cost effective commercial operations instead of reputational space programs by the increased requirement for different payload carrying capacities. Since the primary goal of launch vehicle designers is to maximize the vehicle’s weight-lifting capability while at the same time providing an adequate level of reliability at an acceptable cost, a particular launch vehicle can be configured in several different ways, depending on its mission and the weight of the spacecraft to be launched. A comparative diagram of different launch vehicles is given in Figure 3.

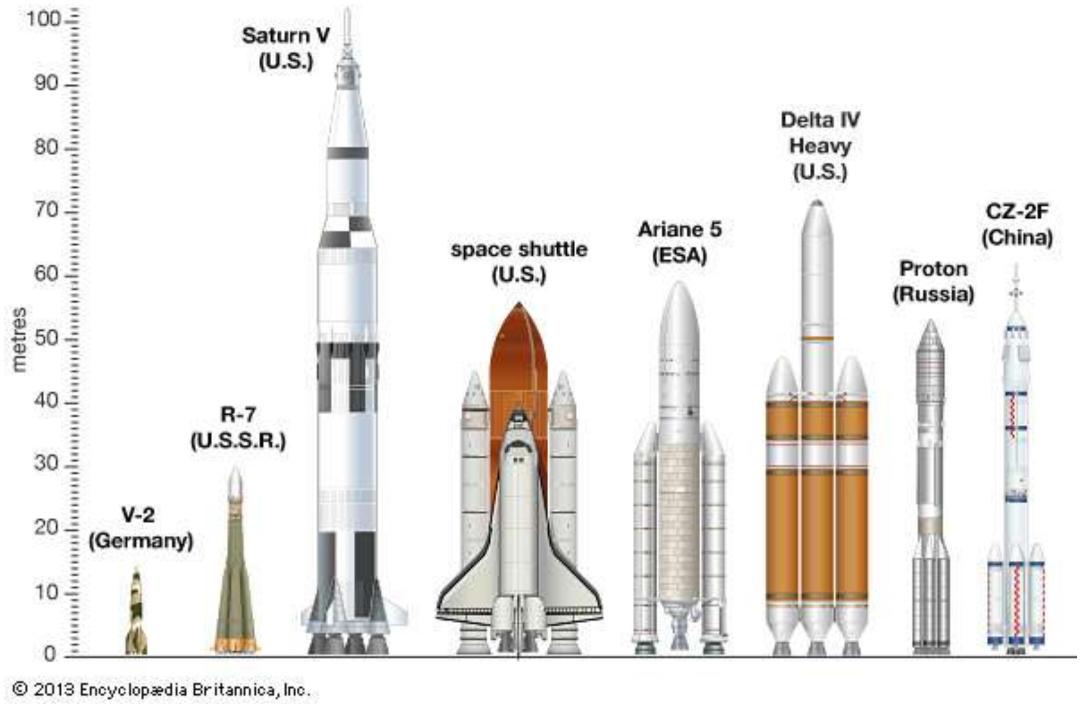

Figure 3 Comparison of Different Launch Vehicles (Britannica, Inc.)

Considering the increasing orbital launch activity during the last decade, it can be deducted that new launch vehicle development projects need to provide cost effective solutions. New launch vehicles need to decrease the cost of a launch to be able to take part in the competitive launch market. This is only possible by determining the cost items in the early phases, which are concept selection and conceptual design, of the project.

In addition to launch vehicle, space launch center also plays an important role in sending spacecraft to orbit as well as suborbital and orbital human flights. Space launch centers should have several technological infrastructures according to the mission requirements. Assembly, test and launch operations, filling of the propellants of both spacecraft and launch vehicle, operations of telemetry and tracking systems, analyses related with the mission and recovery of used systems if necessary are examples of main functions of a space launch center.

V-2, first rocket that can reach to space, initially launched from Peenemünde in Germany during World War II. More than forty later launches of V-2 from White

Sands reaches to altitudes higher than 100 km. Baikonur Cosmodrome in Kazakhstan achieved the first orbital flight with Sputnik-1 in 1957 and the first human launch with Yuri Gagarin in 1961. As a response to early Soviets accomplishments, United States constructed Cape Canaveral Air Force Station in Florida. First human mission to moon surface with Apollo Program was conducted from Kennedy Space Center near Cape Canaveral in 1969.

1 - Vandenberg AFB
2 - Edwards AFB
3 - Wallops Island
4 - Cape Canaveral/KSC
5 - Kourou
6 - Alcantara
7 - Hammaguir
8 - Torrejon AB
9 - Andoya
10 - Plesetsk
11 - Kapustin Yar
12 - Palmachim/Yavne
13 - San Marco Platform
14 - Baikonur/Tyuratam
15 - Sriharikota (SHAR)
16 - Jiuquan
17 - Xichang
18 - Taiyuan/Wuzhai
19 - Svobodny
20 - Kagoshima
21 - Tanegashima
22 - Woomera

Figure 4 Space Launch Centers of World (Braeunig, 2000)

Guiana Space Center which is located near the equator in Kourou of French Guiana is the Europe’s main space launch center. First Chinese human flight to orbit is conducted from the Jiuquan Satellite Launch Center in 2003, October. In 2004, a human is sent to space on the spacecraft SpaceShipOne by a privately funded company as a major breakthrough with the intention of future commercial spaceflights. SpaceShipOne was launched from a runway at Mojave Air and Space Port at California by a carrier plane taking off horizontally.

Depending on the location with respect to equator and target orbit, space launch centers can use the rotational speed of Earth as a benefit. Equatorial speed of 465 m/s provides an energy advantage to launch vehicles for geostationary orbits with an eastward launch. This energy advantage is not valid for highly inclined orbits such as SSO or Molynia. In this case being close to the poles reduces the disadvantage. This example shows the importance of the space launch center location related to the Earth's rotation.

Most of the energy provided to the satellite by the launch vehicle is used for the horizontal acceleration to be able to reach orbital speeds. Consequently, altitude of the launch center does not make a significant difference. Although lower density of atmosphere in high altitudes provides an advantage in terms of reduced drag and higher thrust due to less ambient pressure, transportation efforts to high altitudes prevents space launch centers being attractive when they are built at high elevations.

For the selection of space launch centers location, geography, climatic properties, proximity to existing infrastructure such as electricity and water, availability of different transportation means and many other factors are considered. It is possible to have more reusable launch vehicle missions in the future. Return ranges for space launch centers for reusable systems such as SpaceShipOne is also important. Recovering attempts of Falcon 9 first stage with an autonomous sea platform is also an example for available utilities of a space launch center.

Most important factor for a space launch center is safety. In addition to the remotely controlled or self-activating flight termination systems onboard the launch vehicles, proximity of launch centers to inhabited localities and pass of flight corridors over populated areas also primarily considered for safety. In case of a catastrophic failure of a launch vehicle, these factors play an important role on controlling the risks of harmful effects. In addition, having suitable drop zones for spent stages and fairing halves is also important. Eventually, most of the existing space launch centers are near to ocean or plain grounds to ensure human safety by ensuring that no LV parts are shed on inhabited areas by using suitable launch directions.

Figure 5 Near Ocean Aerial View of LC 39 at Kennedy Space Center

1.1. MOTIVATION

Most important point in launch vehicle design process is the fulfillment of the mission requirements. Mission requirements, especially the performance of the launch vehicle, depend on many factors. An accurate modeling of the launch vehicle performance or realistic predictions are required for the conceptual design studies due to the fact that at least the launch vehicle configuration and performance parameters of the stages are required to be known for simplest trajectory analyses. For the later phases of the project, it may be possible to repeat previous steps and make corrections on available items which is time consuming and costly. In addition to the launch vehicle performance which is directly related with the safety issues, some other factors such as cost, reliability, mission flexibility or time of preparation for operation can also be considered during the conceptual design. Considering such factors during the early phases of the project prevents correction requirements and decreases the risk of increased costs.

Cost of a launch vehicle is important for vehicle to be able to take part in the competitive launch market and for realization of the project as stated before. There are launch vehicle solutions considering only the weight distributions among the stages. It is possible to obtain the optimum mass of a launch vehicle under certain assumptions with help of mathematical equations directly. There are also many studies related to multi objective optimization of the trajectories of existing launch vehicles for maximizing their payload capacity. Although, previous studies on launch vehicle costs shows distinct relations between the costs and weights of the individual items on the vehicle, there is not a specific study considering cost and performance of a launch vehicle together during the conceptual design.

Space launch center safety is another important issue to be considered as a mission requirement which is directly related with the launch vehicle configuration. Most of the space launch centers are located close to plain grounds or major bodies of water. Depending on the location of the space launch center, launch vehicle expose to different climatic conditions and atmospheric properties during its operation. Although factors such as temperature, pressure and wind speeds have direct effect on launch vehicle performance, they are not considered in a detailed way during conceptual design studies. On the other hand, in case of adjacent inhabited areas, a bigger or smaller launch vehicle configuration may result in different dispersion footprints in case of a catastrophic failure near the space launch center. Due to the fact that vehicle configuration and launch center are closely interacting, selection of space launch center location without considering the safety issues during the conceptual design phase of the launch vehicle may cause costly design changes or even relocation of launch center. Within this context, considering initial safety analysis as a part of conceptual design phase is also important.

Main reason for the lack of considerations for different factors in recent studies may be the preference of existing stages of previous launch vehicles in order to cut development costs for new launch vehicle design projects. However this situation may not be applicable to countries that are developing their space programs recently like Turkey. In this thesis, developing a method that can be used for conceptual studies which is capable of considering launch vehicle performance and cost at the same time with a modular structure which allows consideration of additional factors in the future, generating a test case for a sun-synchronous orbit which is a possible target orbit for different purposes, obtaining alternative launch vehicle configurations and selecting one of the alternatives for the determination of space launch center safety distances by considering the effect of seasonal variations of atmospheric conditions is aimed.

In addition, main motivation of this study is

- To model the losses occurring during the flight in a way that they can be expressed in terms of equations and can be used in conceptual design studies,
- To develop a fast and accurate enough method to consider the launch vehicle performance and cost at the same time,
- To model the atmospheric parameters and their effect on dispersion footprints in case of a catastrophic launch vehicle failure
- To analyze the effects of atmospheric parameters and launch vehicle configuration on space launch center safety
- To improve the know-how of Turkey about the subject by considering the possible future projects.

1.2. LITERATURE SURVEY

Launch vehicle is a rocket that is used to transport satellites from the surface of Earth to outside of the atmosphere and put them in to the orbit. Since it is difficult to obtain high orbital velocities, multiple stages that are working in a serial manner are required on launch vehicles to reach higher speeds sufficient to provide centrifugal acceleration needed to balance Earth’s gravity.

Basic idea of staging a launch vehicle is first introduced by Tsiolkovsky in early 20th century after the demonstration of very first principles of reaction engines in 1883. In 1903, Tsiolkovsky published an article with the name "The Investigation of Space by Means of Reactive Devices" and outlined his theory of spaceflight and derive the basic formula for flight of a rocket with changing mass. This formula is still known as the ideal rocket equation or "Tsiolkovsky Equation". It shows that the acceleration of a rocket increases logarithmically as fuel in it burns and mass of the rocket decreases.

In 1916, Goddard, in his research “A method of Reaching Extreme Altitudes”, developed mathematical theories for rocket propulsion and proved that rockets can operate out of the atmosphere by the Newton’s 3rd law which states that action and reaction are equal and opposite. Goddard also outlined the possibility of travelling to the moon after conducting many experiments on solid propellant rocket motors and theorizing the liquid propellant mixtures. He was the first scientist that conceives the future of missiles and space vehicles and he contributed to realization of them directly.

Wernher Von Braun is also an important scientist in rocketry. He became involved in German rocket society in 1929 by working to develop ballistic missiles with the desire of building large rockets capable of spaceflight. His team developed the first V-2 rocket which was flown in 1942. This rocket, which has the technical name Aggregat-4, served as the ancestor of space programs in the United States of America and Soviet Union. Among many published works, Von Braun is an important pioneer as the chief of the design team of Saturn V launch vehicle which carried humankind to the moon surface with the Apollo Program.

After the first commercial communications satellite “Early Bird” placed in geosynchronous orbit by the United States in 1965, satellites began launching on vehicles that are not completely controlled by the governments. There was a market opportunity to provide launch service for the growing commercial communications satellites around the world since then.

European Arianespace Corporation first market Ariane launchers to the commercial customers in 1980. After Space Shuttle become operational in 1982, many existing expendable launch vehicle such as Atlas, Titan and Delta also became commercially available. Today there are many expendable launch vehicles operational in the world. Many new launch vehicle projects and concepts are advancing as the future solutions for today's competitive launch market. As United States and Russia are most active countries in space, vehicles developed by them are well suited for particular missions.

Many of the academic researches focused on optimization of the launch vehicle for a pre-defined flight trajectory or trajectory optimization for a given launch vehicle while many launch vehicle concepts are emerging as solutions as the future launchers. New challenge that confronts the industry is designing launch vehicles that fulfill the mission requirements while keeping the costs at minimum level as Ryan and Verderaime stated in 1993. In addition, (Koelle, 2005) discussed in his study that a launch vehicle program requires consideration of economic factors different from the previous projects that military requirements, technology demonstration or national prestige were the primary concerns. Effects of different factors on specific launch cost are also investigated in his study. Moreover, (Saravi, 2008) describes the importance of cost estimation in decision making. More than 70 % of the lifetime cost of a new design is fixed by the decisions during the early phases. It is important to guess and minimize the costs during the early phases of the project and as accurate as possible.

Space launch center safety is another important issue. Many orbital and suborbital launches conducted from the date back 1965 at which the commercial satellite Early Bird placed to orbit. Initial purpose of launch centers was military. Safety studies related with the launch centers started with the commercial use. In 1970, Knothe, discussed the requirement of range safety for space launches in his paper. Although previous studies show that objects inside the flight corridor are not safety concerns, falling nearby fragments of Apollo 11's first stage are reported by German vessel "Vegesack" in 1969, July. This incident shows that space launch center safety cannot be handled with simple statistics.

Thanks to advancements in methodology and computer programs, calculation of risk values peculiar to different missions became a reality. Many developments are accomplished since the drop of a V-2 rocket to a graveyard in Mexico which is launched from White Sands in 1947. The assessment of hazards imposed by the launching of large scale missiles requires an accurate flight simulation along with a consideration of the events following a failure (1970, Hammond). For an accurate flight simulation wind profiles and atmospheric properties of the launch site should be known precisely. Hammond and Geisinger's study, Reducing Safety Constraints through Vehicle Design, probability of impact and expected casualty calculations are mentioned. Effect of launch vehicle design on safety is emphasized and initial standards of safety analysis are established.

General methodology for the safety analysis is to divide the flight trajectory of the launch vehicle into small segments and relate these segments with different types of hazards according to the mission. Drop points of every possible fragment of the launch vehicle in case of a catastrophic failure are calculated and quantifying risk analysis are conducted accordingly. Launch Risk Analysis study of Baeker in 1977 includes the analysis procedure for assessing the risk associated with the launch of a space launch vehicle. This study is referenced in many of the later studies related with the space launch center safety as a guideline.

During the conceptual design phase, staging is a way to reduce the launch vehicle mass as introduced by Tsiolkovsky which is also related with the cost and safety. Staging means to throw away the no longer useful mass to be accelerated further to the orbital velocities. Types of the stages and their performances are the main factors which determine the mass distribution among stages. Staging that provides the minimum launch vehicle weight can be obtained by staging optimization. The problem of staging optimization has been solved under different assumptions and simplifications in the early ages of the space programs.

Coleman, in 1960, is developed a generalized method for determining the optimum weight distribution for multistage rockets. Variation of different structural ratios for stages is included in his study. All parameters included in the optimization were expressed in terms of stage weights by Coleman. Likewise, (Cooper, 1960) shows that for optimized staging the individual stage mass ratios depend only on the structure factors and are independent of the exhaust velocity and mission requirement. His analysis also leads to the definition of an effective exhaust velocity which includes the effect of structures on the propulsion system performance and used widely. Coopers's optimization schemes are as functions of the mission velocity requirement.

By the development of numerical methods for nonlinear optimization problems, staging solutions without certain assumptions have also been provided. For all these solutions on staging, formulization is based on the ideal velocity. In reality, ideal velocity can never achieved due to the losses such as gravitational losses, aerodynamic losses, engine transients, thrust losses due to atmospheric back pressure and steering losses arising from misalignment of thrust vector of stages for the maneuvering. Although application of different factors for the formulation of ideal velocity have been tried to be applied, exact determination of loses is impossible without detailed trajectory analysis. (Adkins, 1970) obtains formulations by including expressions for the average effects of drag for each stage. Although gravity losses are also included in the study, some assumptions for the total velocity are still required.

Since using a performance model for the launch vehicle is one of the key concerns, vehicle performance still needed to be determined before the detailed trajectory analyses. Velocity losses arising from different sources also required to be included for reasonable estimates on vehicle size and realistic calculations. For the loss estimations, information on properties and payload capacities, so the ideal velocities, of launch vehicles that are given in user manuals can be used.

Since generated new launch vehicle concepts need to consider not only the launch vehicle performance in terms of velocity but also additional factors such as cost, reliability etc., concept selection can be enriched by additional considerations to be able to select the best alternative and avoid later design changes to fulfill mission requirements. (Koelle, 2010) in Handbook of Cost Engineering included a model that can be used in a parametric way for space transportation systems. Model uses historical data from previous projects and extrapolates it for the future launch vehicles. Relation of launch vehicle components with different cost items are given as cost estimation relationships in his study.

Studies related with safety increased after mid-90s. In 1995 and 1997, Montgomery and Cole investigate the effect of falling fragments on people in the open in a detailed manner. Many space launches conducted at the last half of the century after the successful insertion of Sputnik-1 into orbit. Although launch vehicles are designed for high reliability, failures are common due to extreme operating conditions. Compilation of failure data and related assessments lead to valuable statistics on launch vehicle reliability. Launch Vehicle Historical Reliability study, which is conducted by Sauvageau and Allen in 1998, examines the US launches. According to the study more than 70 % of the failure reasons address the solid and liquid propulsion systems.

Recent studies are generally based on Licensing Regulations and Safety Requirements for the Operation of Launch Site (2000) which is published by Federal Aviation Administration (FAA). Flight Safety Code (2002) of Licensing and Safety Office of Australia explains the launch site safety approaches similarly. Both documents include the methodology of quantitative safety approaches and sample calculations for a new expandable space launch vehicles. However, mentioned methods are general and not applicable for some cases. It is required to modify existing methodology and conceptive examples for different launch scenarios.

After space shuttle Columbia disaster occurred in 2003, partially toxic debris fell over a 300 km^2 surface at Texas. Reason for this failure is the separation of thermal insulating foam from the main fuel tank at the initial phases of the flight. Although no civilian injury reported after the accident, safety related studies are increased after. Reliability is directly related with mission success and safety for both expandable and reusable launch vehicles. Reliability of a launch vehicle is required from the liftoff moment until the end of the mission with orbit insertion or disposal of the last stage to be able to answer the question “How much safe?”. Guikama investigated launch vehicle success rates by conducting Bayesian analysis for launch vehicle reliability in 2004. FAA defined key terms and performance standards for failure probabilities of new expandable launch vehicles in 2005.

Space launch center safety benefited from many different fields such as chemical industries and explosive production. Recent content evolved to include secondary influences such as pressure, thermal and toxic effects. Importance of safety increases with the advancements in civilian and military space applications. In addition to the reliability values of launch vehicles, wind profiles and atmospheric properties should be considered for credible assessments.

1.3. SCOPE

This thesis is organized in five chapters. Explanation of the chapters is as follows;

Chapter 1 includes the introduction of the study. Problem on launch vehicle concept selection and space launch center safety are explained and motivation for the solution is given. After presenting a short literature survey about the subject, scope of the thesis is stated.

Chapter 2 gives the classification of the launch vehicles. Recent developments on expandable launch vehicles are given in this chapter in addition to the future developments. This chapter also focuses on launch vehicle performance. Advantage of staging is examined. In addition, performance requirement of launch vehicles in terms of velocity and its components are explained. Optimum distribution of mass among stages for different scenarios is presented. Launch vehicle configuration for safety calculations is determined with help of the presented information and cost model that is provided in Appendix B.

Chapter 3 presents the atmospheric parameters (variation of pressure, temperature and density as well as their deviations from their mean values, mean and maximum wind speeds and prevailing wind directions) and generation of vector wind profile which depends on statistics.

Chapter 4 represents formation of the problem and solution. Comparisons of the safety zones for summer and winter conditions that are obtained using the randomly generated wind profiles are presented. This chapter also includes the safety zones calculations of secondary effects and their seasonal variations.

Chapter 5 comprises a conclusion for the whole study. This chapter also gives a summary on the contribution of the thesis and includes recommendations on future studies that can be appended.

In addition, computer codes that are used for the calculation of possible launch vehicle fragments' drop points and parametric cost model that is used for the determination of launch vehicle configuration for the safety calculations is presented in Appendix A and Appendix B, respectively.

CHAPTER 2

LAUNCH VEHICLE PERFORMANCE

A space flight system consists of three important sections which are essential for completing the mission. These are space section, transfer section and the ground section. Space section consists of the spacecraft and the payload. Ground section is responsible of controlling and monitoring the spacecraft and receiving and processing the flight data. Finally the transfer section provides the transport of payload into the space by means of a launch vehicle (LV). For space flight, launch vehicle is a single or multistage rocket that is used to carry a payload. In this case payload is usually an artificial satellite designed to operate in the Earth orbit but it is also possible to obtain sub-orbital or interplanetary trajectories for different missions.

2.1. CLASSIFICATION

Although there are more than ten operational launch vehicles in the market under the control of different countries, it is possible to define two main types of launch vehicles.

2.1.1. Reusable Launch Vehicles

They are designed to be recovered undamaged (intact) and used again for different missions after their operation with certain processes that are relatively cheaper when compared to production from the beginning. Most known example for reusable LV is the Space Shuttle (Figure 6). Boosters of the shuttle, which consist of several segments that are connected to each other, recovered from the ocean. In addition, orbiter section of the Space Shuttle lands to ground like an aircraft after its mission on the orbit completed. Space Shuttle is the only example for an orbital reusable LV. There are also other examples of suborbital reusable launch vehicles such as SpaceShipTwo. SpaceShipTwo is a suborbital launch vehicle which will be used in the near future for human space flight and space tourism. WhiteKnightTwo will carry SpaceShipTwo up to 16 km height and it will be dropped from the carrier plane (Figure 7). Rest of the flight will be propelled with the hybrid rocket propulsion system onboard of the SpaceShipTwo. This will provide a total of 6 minutes of weightless space flight to six passengers and two pilots.

Figure 6 Space Shuttle on Launch Pad (NASA)

Figure 7 SpaceShipTwo under WhiteKnightTwo (Virgin Galactic, 2013)

One recent example of recovering attempts is the booster landing tests of Falcon 9 launch vehicle. Objective of these tests, which starts at 2013, is to execute a controlled re-entry of the first stage of the launch vehicle and recovery. As of December, 2015, the eighth test yielded a successful recovery.

2.1.2. Expandable Launch Vehicles

These types of launch vehicles are designed for one time use and their parts usually break up during atmospheric re-entry after the end of their operation or break up at the time of hit to the surface of Earth. Most of the launch vehicles are expandable. One of the most recent examples for that kind of LV is European VEGA. VEGA made its maiden flight from Kourou launch site in 2012, January successfully. VEGA's first commercial flight conducted in 2013. In June of 2015, VEGA placed Sentinel-2A, Earth observation satellite, into its SSO target orbit successfully.

Figure 8 VEGA Launch Vehicle on Launch Pad (ESA, 2012)

Launch vehicles are often characterized by the amount of payload mass that they can carry into the Low Earth Orbit (LEO).

Sounding Rocket	Only suborbital flight
Small Lift Launch Vehicles	< 2000 kg into LEO
Medium Lift Launch Vehicles	2000 – 20000 kg into LEO
Heavy Lift Launch Vehicles	20000 – 50000 kg into LEO
Super Heavy Lift Launch Vehicles	> 50000 kg into LEO

Low Earth Orbit (LEO) is a circular orbit below an altitude of approximately 2000 km. Orbit altitude of 200 km or below gives rapid orbital decay to satellites due to molecular drag and sun pressure. LEO altitude is commonly accepted as between 160 – 2000 km.

It is also possible to classify launch vehicles according to the launch platform.

Land Launch (e.g., Dnepr, VEGA, Falcon 9)

Sea Launch (e.g., Zenit 2SL)

Air Launch (e.g., Pegasus, Virgin Galactic)

Different launch platforms have certain advantages. By sea launch, it is possible to use the rotation of Earth effect as maximum as possible which increases the available payload mass. In addition, since sea launch platforms can be located in the ocean near the equator, drop zones and flight corridor of launch vehicle is not a high priority problem. By air launch, it is possible to give LV an initial velocity and altitude which again increases the available initial energy and decreases the losses generated due to the atmospheric flight and gravity. By land launch on the other hand, management of launch campaign is easier, handling of much more propellant (generally cryogenic, hazardous or poisonous) is possible. Land launch is the most common method and there are 19 active launch sites around the world.

2.1.3. Recent Developments

Many small satellites have been launched with larger primary payloads. Orbital injection of these payloads depends on the target orbit and the mission of the main payload. This results in longer wait time for small payloads to find a common, suitable mission with larger payloads. In addition, stand-alone missions for small payloads are more expensive. To overcome mission compatibility difficulties and decrease the cost of stand-alone missions, many small lift launchers have been designed and many more are under development.

Satellites can be classified according to their sizes as follows;

Micro Sized Satellites	10 – 100 kg
Mini Sized Satellites	100 – 500 kg
Medium Sized Satellites	500 – 1000 kg
Large Satellites	> 1000 kg

There are more than 300 satellites that are in the range of micro to medium sized satellite operational in the low Earth orbit (LEO), excluding the medium (MEO) and elliptical orbits and geostationary orbit (GEO). Launch mass of about 550 of the total of around 700 operational satellites in LEO are less than 2000 kg and suitable to be launched by small lift launch vehicles.

VEGA, which is also a small lift launch vehicle, made its maiden flight in January 13, 2012. VEGA program starts in 90s. An initial study to expand the capabilities of Ariane launch vehicle family with a small launcher is started by European Space Agency (ESA) in 1998. Performance goal of VEGA is decided to be carrying a payload mass of 1500 kg to a circular orbit with an altitude of 700 km and with an inclination of 90^{o}. It has four different stages. First three stages of VEGA consist of solid propellant rocket motors P80 FW, Zefiro 23 FW and Zefiro 9 FW (from first to third). Restartable bipropellant (UDMH/NTO) AVUM (Attitude and Vernier Upper Module) including a propulsion module and an equipment module is the upper stage of the vehicle. Although performance goal of VEGA is defined clearly, it is capable of carrying payloads of 300 – 2500 kg to circular orbits with an altitude of 300 – 1500 km with different inclinations. Performance of VEGA for circular orbits with different inclinations and altitudes is given in Figure 9.

Total gross lift-off weight (GLOW) of VEGA is about 137 tons and height of the launch vehicle is more than 30 m. Main purpose of development is to decrease the launch cost for unit payload mass in addition to the development and ground operations' costs. Upper stage AVUM bases the Ukrainian bipropellant liquid rocket motor RD869. Properties of stages of VEGA are given in Table 1.

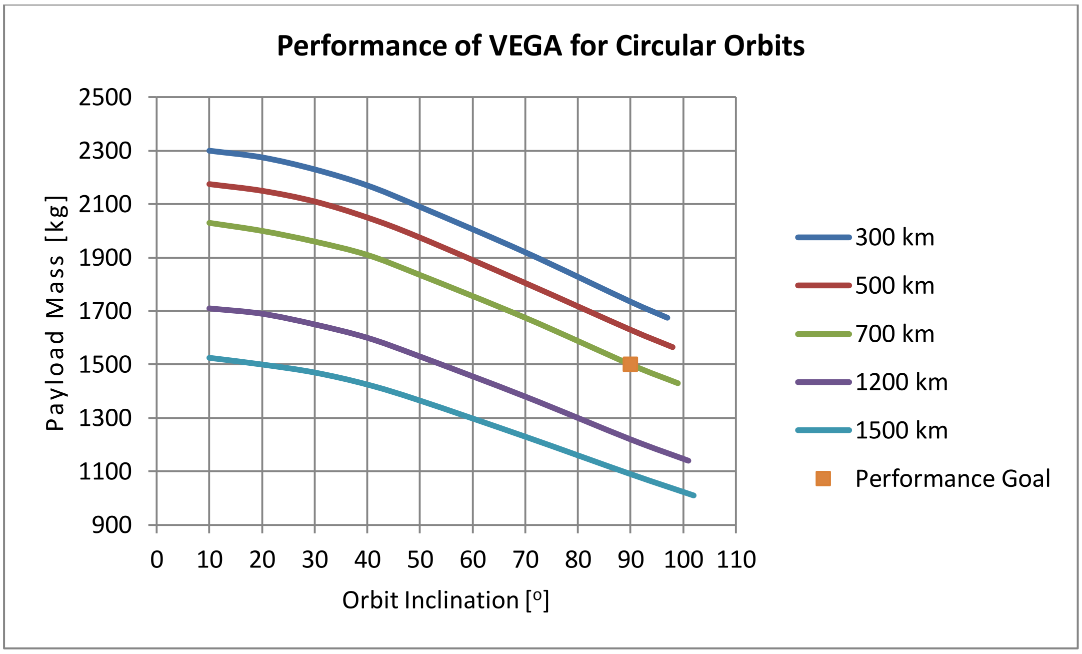

Figure 9 Performance of VEGA Launch Vehicle for Circular Orbits

Table 1 Stage Properties of VEGA Launch Vehicle

	P80 FW	**Z23 FW**	**Z9 FW**	**AVUM**
Diameter [m]	3.00	1.90	1.90	2.18
Length [m]	11.20	8.39	4.12	2.04
Propellant Mass [kg]	88365	23906	10115	550
Dry Mass [kg]	7431	1845	833	418
Total Mass [kg]	95796	25751	10948	968
Mass Ratio [%]	7.76	7.16	7.61	43.18
Average Thrust [kN]	2261	1196	225	2.45
Specific Impulse [s]	280	289	295	315.5
Burn Time [s]	106.8	71.7	117.0	667.0
TVC Capacity [^{o}]	± 6.5	± 7.0	± 6.0	NA

Another recent example of new launch vehicle is the Korean Naro-1 which is previously known as Korea Space Launch Vehicle. Naro-1 made its first successful flight in January 30, 2013 by carrying STSAT-2C into LEO. It is a two staged small lift launch vehicle using Russian liquid propellant engine RD-151 as the first stage and a solid rocket motor developed by National Space Agency of South

Korea, as the second stage. Total height of the vehicle is 33 m and the diameter is 3 m. It has a gross lift off weight of approximately 140 tons.

2.1.4. Future Projects

Orbital launch activity in 2010 is increased by 14% (from 74 annual launches to 84 attempted carrying 133 payloads, Space Report 2012). One of the two major developments back then is the first flight of the small lift launch vehicle VEGA in European spaceport Kourou (French Guiana). About half of the launches including all type of missions as well as GEO and MEO have been made by small launch vehicles for the past ten years. Distribution of payload masses in near-term manifest for non-geosynchronous orbits shows that most of the payloads which are planned to be launched in the near future can be carried to the orbit by small launch vehicles.

Table 2 Distribution of Payload Masses in Near Term Manifest (Non-GEO)

Payload Mass	2012	2013	2014	2015	Total	Percent of Total
< 200 kg	17	19	4	1	41	29.3 %
200 - 600 kg	8	11	11	0	30	21.4 %
601 - 1200 kg	6	9	4	19	38	27.1 %
>1200 kg	6	5	9	11	31	22.1 %
Total	37	44	28	31	140	100 %

Göktürk-1 which is a Turkish reconnaissance satellite weighting about 1100 kg is going to operate in SSO at about 700 km height in the near future and will be launched by Arianespace from the Guiana Space Center with VEGA as a fourth contract for the small lift vehicle.

ESA’s analyses on satellite payload mass trend show that the size of individual missions can be reduced by splitting payload complements. Nature of today’s mission objectives such as earth observation, scientific experiments and other engineering purposes would probably drive capacities of the satellites and the launchers in the future.

New trend for satellites is using constellations of smaller satellites for communication, scientific experiments or remote sensing. Turkey also has a goal of producing its own launch vehicle to be able to carry its own small payloads for different missions. There are many satellites that are going to be owned by Turkey given in “SSM Uydu ve Uzay Yol Haritası”. Due to the fact that these satellites can be carried stand alone to orbit with help of small launch vehicles, small launch vehicles are selected as a baseline for the conceptual design studies.

2.2. LAUNCH VEHICLE PERFORMANCE

For a space mission, performance of launch vehicle can be expressed in terms of the total delta-v which is the total of the velocity increment given to the payload by means of the propulsive tasks. It is a scalar quantity and it is independent of the mass of the launch vehicle or the payload in addition to the amount of thrust or the total duration of burns. It is particularly useful for conceptual design studies due to the fact that it is easy to implement the relation of the weight of the vehicle to the mission requirements such as orbit altitude, orbit inclination and launch site location.

The Tsiolkovsky rocket equation, which is also known as ideal rocket equation, derived by Russian rocket scientist Konstantin Eduardovich Tsiolkovsky and published in 1903. Equation shows that the delta-v of a rocket or a stage of a LV is directly proportional to the specific impulse of the propulsion system and the logarithm of the fuelled to empty mass ratio of the vehicle. It is possible to apply the principle of impulse and momentum to obtain the ideal rocket equation.

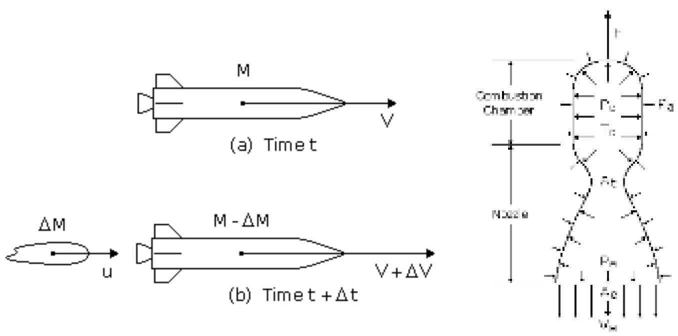

Figure 10 Conservation of Momentum and Derivation of Thrust

Newton's 3rd Law states that action and reaction are equal and opposite. By defining mass flow rate as $\dot{m}$ in kg/s and effective exhaust velocity as V_e in m/s without considering the effects of atmospheric pressure and the pressure in the exhaust stream;

$$F = \dot{m}V_e \tag{2.1}$$

$$\dot{m} = \frac{\mathrm{d}m}{\mathrm{d}t} \tag{2.2}$$

$$\frac{\mathrm{d}V}{\mathrm{d}t} = \frac{F}{m} \tag{2.3}$$

Combining Equations (2.1) and (2.2)

$$\frac{\mathrm{d}V}{\mathrm{d}t} = V_e \frac{\mathrm{d}m}{\mathrm{d}t}\frac{1}{m} \tag{2.4}$$

$$\mathrm{d}V = V_e \frac{dm}{m} \tag{2.5}$$

Integration of velocity from zero to V as the mass changes from an initial mass of m_0 to final mass of m_f;

$$\int_0^V \mathrm{d}V = V_e \int_{m_0}^{m_f} \frac{\mathrm{d}m}{m} \tag{2.6}$$

$$V = V_e \ln\left(\frac{m_0}{m_f}\right) \tag{2.7}$$

When effective exhaust velocity V_e, which depends on chemical nature of the propellants, assumed to be constant, there is no need to know the variation of the thrust. V is the maximum velocity that a rocket can attain without considering the effects of atmospheric drag or gravity.

It is necessary to take the surface integral of pressure over the whole inner surface and the nozzle in order to find the effective exhaust velocity of a rocket; (F is the force tending to accelerate the rocket)

$$F = \oint p\mathrm{d}A \tag{2.8}$$

At any point in nozzle, upstream pressure is higher than the downstream pressure;

$$\mathrm{d}F = pA - \left(p - \mathrm{d}p\right)A \tag{2.9}$$

where A is the cross sectional area and dp/dx is the pressure gradient. It is possible to find the force tending to accelerate the gasses and the rocket as follows; (u_e is the true exhaust velocity)

$$F_G = \oint p\mathrm{d}A - P_e A_e = \dot{m}u_e \tag{2.10}$$

$$F_R = \oint p\mathrm{d}A - P_a A_e \tag{2.11}$$

By substituting $\oint p\mathrm{d}A$ into the equations 2.10 and 2.11, effective exhaust velocity can be found.

$$V_e = u_e + \left(\frac{P_e - P_a}{\dot{m}}\right)A_e \tag{2.12}$$

To obtain the maximum thrust, the exit pressure P_e should be equal to the ambient pressure P_a, which is also known as the atmospheric pressure. For an ascending rocket or a stage, P_a continuously drops. By considering staging, one should optimize the nozzle length and the exit area A_e, respectively. In vacuum, $P_a = 0$ and in order to obtain $P_e = P_a$ for optimum expansion, infinitely long nozzle extensions are required which is impractical due to additional mass of nozzle. Use of extendable nozzle sections is also an option to prevent separation in nozzle and decrease the performance loss.

Total impulse, I_T is defined as the thrust force F integrated over time t. For constant thrust and negligible transients, total impulse can be approximated as $I_T = Ft$. Specific impulse, I_s or Isp is the total impulse per unit weight of propellant consumed. It is a merit of performance and shows how effectively the propellant mass is used to generate thrust.

For constant propellant flow rate;

$$I_S = \frac{\int_0^t F\mathrm{d}t}{g_0 \int_0^t \dot{m}\mathrm{d}t} \tag{2.13}$$

Using the definition of exhaust velocity;

$$V_e = \frac{I_S}{g_0} = \frac{F}{\dot{m}} \tag{2.14}$$

The Tsiolkovsky Rocket Equation obtained as follows;

$$\Delta V = I_S g_0 \ln\left(\frac{m_0}{m_f}\right) \tag{2.15}$$

Ideal rocket equation shows the maximum attainable velocity for a rocket without considering any other effects than propellant mass and characteristics such as drag and gravity.

2.2.1. Staging

Staging means to throw away the no longer useful structural mass like empty propellant tanks during the flight to minimize the mass to be further accelerated. When the effect of increasing mass fraction m_0/m_f on the total attainable velocity ΔV is investigated, it can be seen that increasing the mass fraction after some point will have no effect on the velocity gain. In addition, it is an important fact that a rocket can travel faster than the exhaust velocity.

Specific impulse of various propulsion types such as mono-propellant or bi-propellant liquid, solid, hybrid or nuclear propulsion have certain technological limitations. For Space Shuttle main engine when operating in vacuum, measured specific impulse is 453 s and this specific impulse is equivalent to an effective exhaust velocity of 4440 m/s.

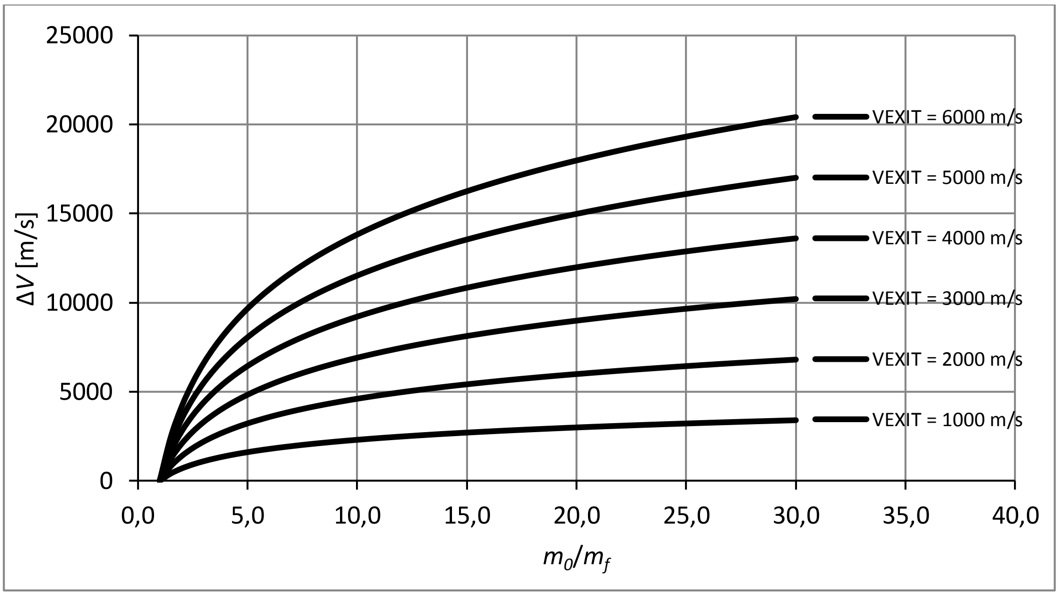

Figure 11 Total Attainable Velocity w.r.t. Mass Fraction

Since maximum velocity with a single staged vehicle is limited, staging is a solution to reach higher orbital velocities which is the case in most of the launch vehicles. Assuming a single staged launch vehicle equipped with a rocket engine having a specific impulse of 300 s, one should obtain the effect of staging on the attainable ΔV for the hypothetical LV as follows;

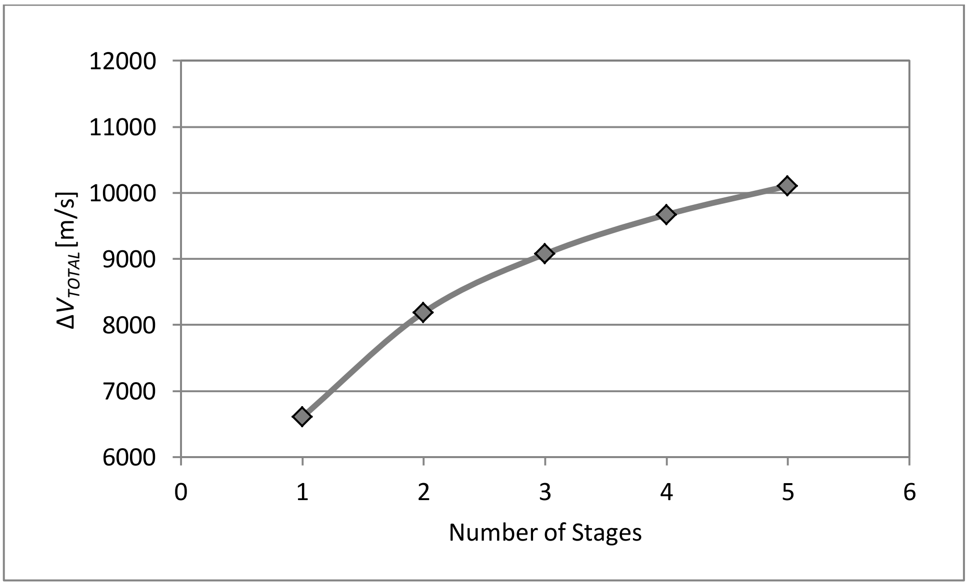

Figure 12 Effect of Staging on Total ΔV_{TOTAL}

This figure is obtained by equal division of propellant mass among the stages which is never the real case. Increasing the number of stages beyond some point is not efficient. Details of the example launch vehicle is given in Table 3 where m_s, m_p and m_t are the structural, propellant and total mass of the rocket motors used, respectively. By staging, cost and complexity of the system increases and reliability decreases due to the increased number of interstage structures and stage separation mechanisms. Percent increase of total ΔV by adding just one more stage to the single staged launch vehicle is 24.0 % while increasing the number of stages from four to five increases the total ΔV only by 4.5 %.

Table 3 Properties of Hypothetical Multistage Launch Vehicles

Single Stage	**Stage 1**
m_s [kg]	1500
m_p [kg]	13500
m_t [kg]	15000
m_p/m_t	0,90
m_i [kg]	15100
m_f [kg]	1600
ΔV [m/s]	**6606.1**

Two Stages	**Stage 1**	**Stage 2**
m_s [kg]	750	750
m_p [kg]	6750	6750
m_t [kg]	7500	7500
m_p/m_t	0,90	0,90
m_i [kg]	15100	7600
m_f [kg]	8350	850
ΔV [m/s]	**1743.5**	**6447.1**

Three Stages	**Stage 1**	**Stage 2**	**Stage 3**
m_s [kg]	500	500	500
m_p [kg]	4500	4500	4500
m_t [kg]	5000	5000	5000
m_p/m_t	0,90	0,90	0,90
m_i [kg]	15100	10100	5100
m_f [kg]	10600	5600	600
ΔV [m/s]	**1041.4**	**1735.7**	**6298.2**

As a result, staging allows higher ΔV for a certain technological level. It is important to note that, dividing the propellant mass equally among the stages is not the optimal solution in terms of the LV gross lift off weight. In addition, staging increases the production cost of the launch vehicle and decreases the reliability due to the increased number of parts and complexity.

2.2.2. Velocity Requirement

A launch vehicle with or without boosters, which consists of multiple stages, must provide the necessary velocity increment to the payload in order to carry it from launch site to the target orbit. Total required velocity consists of several components. It is possible to define the velocity requirement of a launch vehicle as follows;

$$\Delta V_{required} = \Delta V_{orbit} + \Delta V_{rotation} + \Delta V_{losses} \tag{2.16}$$

It is possible to calculate the required orbital velocity ΔV_{orbit} with the related orbital parameters. It is also possible to determine the velocity gain or loss arising from the rotation of earth $\Delta V_{rotation}$ depending on the launch site latitude and altitude and inclination of the target orbit. Final term ΔV_{losses} includes several loss terms such as gravity loss, thrust loss due to atmospheric back pressure, drag force losses, steering losses, losses due to propellant residues on the stages, cant angle loss if there is any and other losses. Since the gravity loss is the most dominant term, it is more useful to divide ΔV_{losses} into two main sections as follows;

$$\Delta V_{losses} = \Delta V_{gravity} + \Delta V_{other} \tag{2.17}$$

In literature, there are many statements that $\Delta V_{gravity}$ can be assumed as 1500 m/s. Although there is not any direct method for estimation of other losses ΔV_{other}, some suggestions propose a value between 500 – 1500 m/s which is not exact and not accurate enough for conceptual design studies. Gravity can also be calculated by taking the integral of gravity force acting on the launch vehicle over the trajectory but the problem is that the trajectory cannot be determined exactly without detailed analyses.

Gravity loss and other losses mentioned before highly depends on the trajectory followed by the LV but it is possible to create an initial estimation by examining the properties of existing small lift launch vehicles with a statistical approach and this initial estimation will be accurate enough for conceptual design studies.

2.2.3. Orbital Velocity

Velocity at any point on an orbit can be found by using the energy equation. Shape of the orbit only depends on the initial velocity and the distance from the center of the earth.

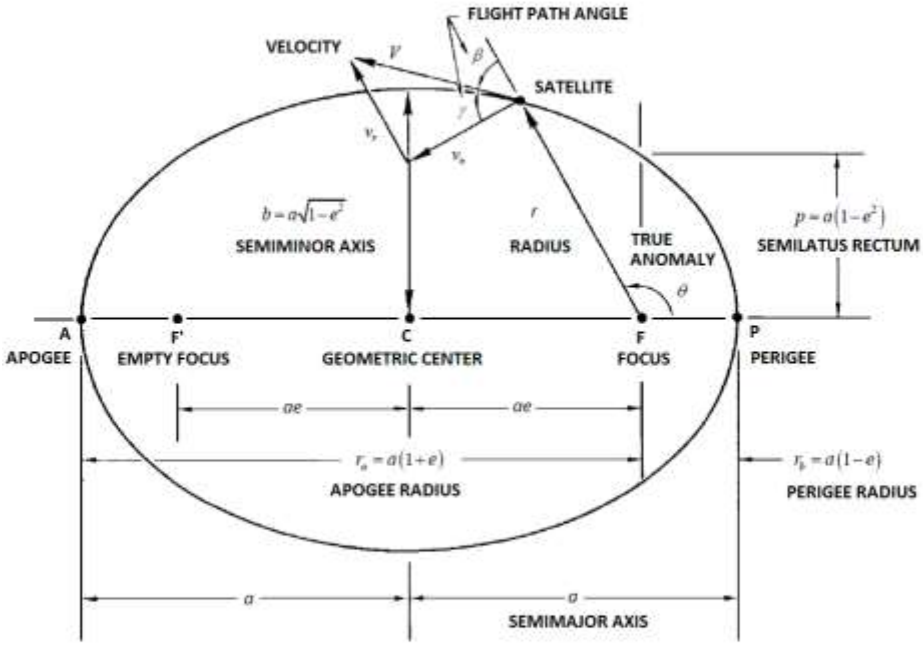

Figure 13 Orbital Parameters on Ellipse Geometry (Chobotov, 2002)

The orbital parameters which are given in Figure 13 can be defined as follows;

Semimajor Axis	$a=\left(r_a+r_p\right)/2$	(2.18)
Semiminor Axis	$b=a\sqrt{1-e^2}$	(2.19)
Eccentricity	$e=\left(r_a-r_p\right)/\left(r_a+r_p\right)$	(2.20)
Apogee Radius	$r_a=a\left(1+e\right)$	(2.21)
Perigee Radius	$r_b=a\left(1-e\right)$	(2.22)
Semilatus Rectum	$p=a\left(1-e^2\right)=b^2/a=r_p(1+e)=r_a(1-e)$	(2.23)
Flight Path Angle	$\gamma=\pi/2-\beta=\cos^{-1}\left(\frac{v_n}{V}\right)=\tan^{-1}\left(\frac{v_r}{v_n}\right)$	(2.24)
True Anomaly	θ	

The radial component of velocity can be found by taking the derivative of radius vector;

$$v_r = \sqrt{\frac{\mu}{p}} e \sin\theta \tag{2.25}$$

The normal component v_n of velocity V can be found as follows;

$$v_n = r\dot{\theta} = r\left(\frac{H}{r^2}\right) = \sqrt{\frac{\mu}{p}}\left(1 + e\cos\theta\right) \tag{2.26}$$

Finally, since radial and normal components of velocity is perpendicular, velocity at any point on orbit;

$$V = \sqrt{\frac{\mu}{p}\left(1 + e^2 + 2e\cos\theta\right)} \tag{2.27}$$

Maximum and minimum velocity points in an orbit occur at apogee and perigee points which are given by the following equations.

$$v_a = \sqrt{\frac{\mu}{p}}(1 - e) \tag{2.28}$$

$$v_p = \sqrt{\frac{\mu}{p}}(1 + e) \tag{2.29}$$

Where v_a is the maximum velocity at apogee and v_p is the minimum velocity at perigee. Since the specific mechanical energy in an orbit is constant, it is also possible to find the velocity at any point in an orbit by using the energy equation as

$$V = \sqrt{\mu\left(\frac{2}{r} - \frac{1}{a}\right)} \tag{2.30}$$

For circular orbits $r = a$ and orbital velocity becomes a simple term. Elliptic orbits intersect with circular ones and can be used to transfer between two circular orbits if they are on the same orbital plane. Hohmann Transfer is the most energy efficient way to transfer between two circular orbits. Using the impulsive maneuver assumption energy required to transfer between two circular orbits can be calculated. For h_1 and h_2 being the altitude of circular orbits, V_1 and V_2 being the required velocity increments, apogee and perigee velocities are as follows;

$$v_a = V_c^2 - V_2 \tag{2.31}$$

$$v_p = V_c^1 - V_1 \tag{2.32}$$

By using the previously defined values of semilatus rectum *p*, eccentricity *e* and apogee and perigee velocities of the elliptical transfer orbit v_a and v_p, it is possible to obtain transfer velocities as follows;

$$V_1 = \sqrt{\frac{2r_a\mu}{r_p(r_a + r_p)}} - \sqrt{\frac{\mu}{r_p}} \tag{2.33}$$

$$V_2 = \sqrt{\frac{\mu}{r_a}} - \sqrt{\frac{2r_p\mu}{r_a(r_a + r_p)}} \tag{2.34}$$

Total velocity requirement for the Hohmann Transfer is the sum of these two velocities which can be used to calculate the total velocity requirement of an orbit transfer between two circular orbits.

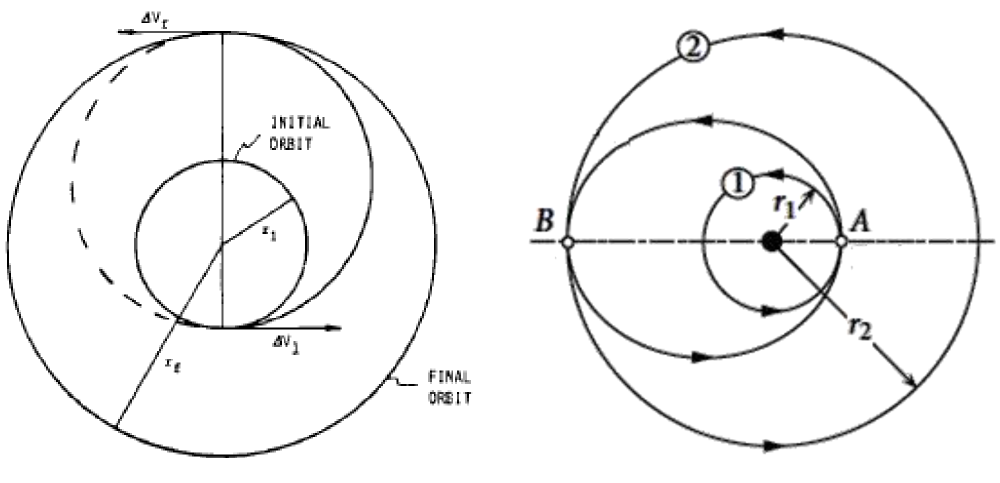

Figure 14 Hohmann Transfer (Chobotov, 2002)

Classification of orbit can be done according to the eccentricity.

Table 4 Orbit Types According to Eccentricity

e	Orbit Type
0	Circle (a = r)
< 1	Ellipse (a > 0)
1	Parabola (a → ∞)
> 1	Hyperbola (a < 0)

It is obvious that orbital velocity decreases with increasing mean altitude. In addition, increasing true anomaly between $0^o – 180^o$ decreases the orbital velocity of an object in an elliptical orbit. Variation of velocity for circular orbits with respect to altitude and variation of velocity for elliptic orbits with respect to true anomaly is given in Figure 15 and 16, respectively.

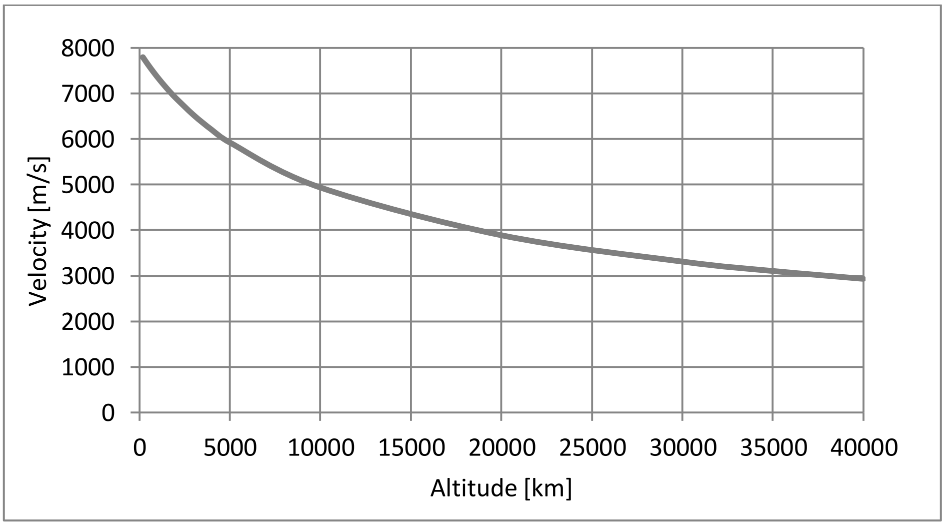

Figure 15 Variation of Orbital Velocity w.r.t. Altitude (Circular Orbit)

Orbital velocity ΔV_{orbit} is the largest term in $\Delta V_{required}$ and it is easy to calculate this value independent of the launch site location or the type of the launch vehicle. It is also possible to determine required orbital velocity for different elliptical orbits with ease with the help of the previously explained equations.

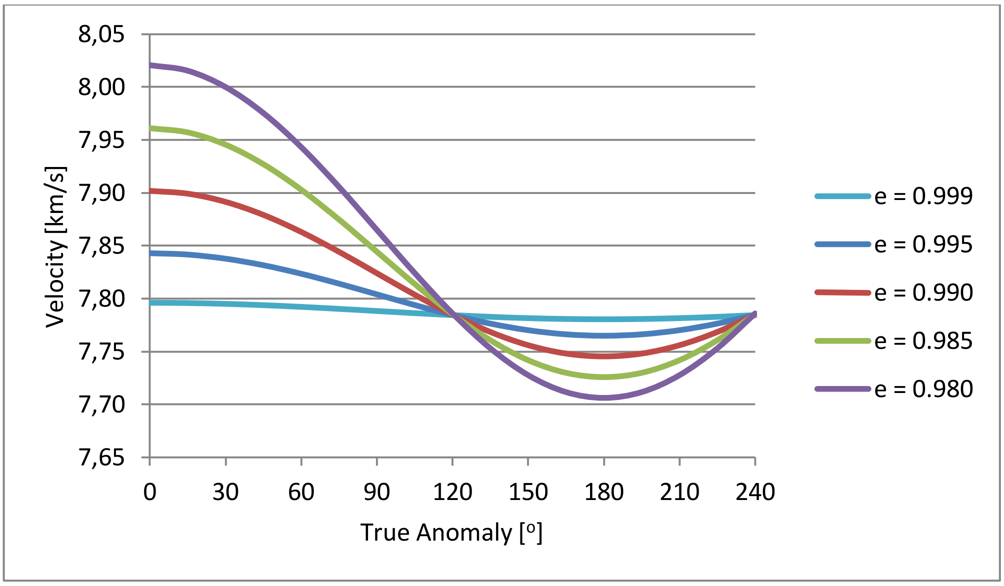

Figure 16 Variation of Orbital Velocity w.r.t. True Anomaly (Elliptic Orbits)

2.2.4. Sun-Synchronous Orbits

In order to define an orbit, six quantities, called orbital elements, are required. These are the length of semi major axis a, eccentricity e, inclination i, argument of perigee ω, time of perigee passage τ and right ascension of the ascending node Ω. It is also possible to replace the time of perigee passage with the mean anomaly MA which is a uniform varying angle at some arbitrary time t.

First three orbital elements a, e and i are dimensional elements. They specify the shape and size of the orbit and give the position of an object in the orbit. Remaining orbital elements are related with the orientation of the orbit in the space.

Orbital Plane of sun-synchronous orbits (SSO) rotates at the same period as the Earth's solar orbit. In other words, SSOs are orbits with secular rate of the right ascending node equal to the right ascension rate of the mean sun. For an orbit to be sun-synchronous, the following equation must be satisfied;

$$\left(\frac{d\Omega}{dt}\right)_s = -\frac{3}{2} n J_2 \left(\frac{R}{p}\right)^2 \cos i = \dot{\alpha} = 0.9856 \frac{\deg}{day} \tag{2.35}$$

where n is the orbit mean motion, R is the earth radius, J_2 is the Jeffery constant for earth and $\dot{\alpha}$ is the right ascension rate of the mean sun. For a certain altitude, inclination of an orbit to be sun-synchronous is fixed. Orbit inclination increases with increasing altitude. Change of inclination with respect to altitude of a circular sun-synchronous orbit is given in Figure 17.

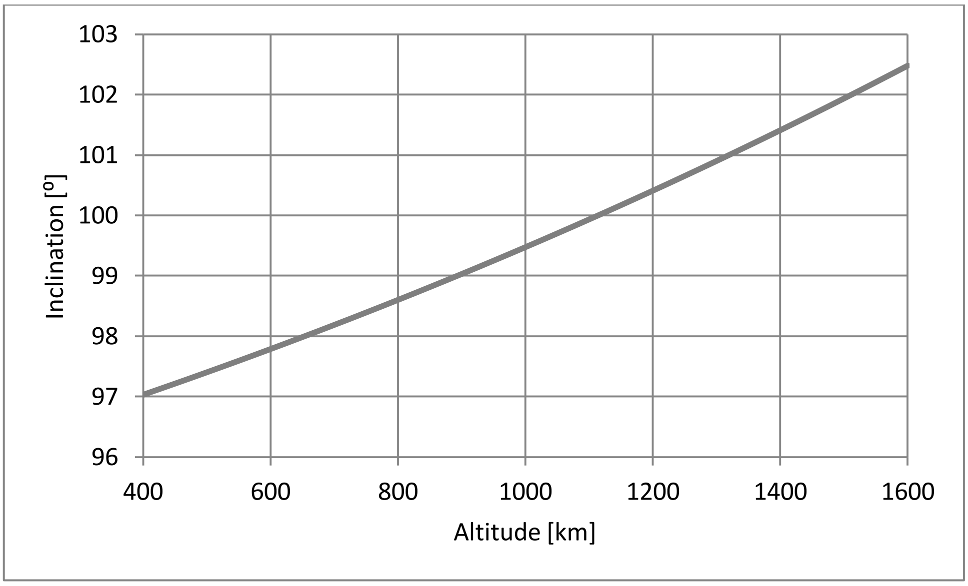

Figure 17 Variation of Inclination for SSO

2.2.5. Elliptic Orbits

Elliptic orbit is an orbit with an eccentricity value less than 1. As previously stated in orbital velocity section, velocity on an elliptic orbit can be calculated using Equations (2.27), (2.28) and (2.29). Maximum and minimum velocity points in an elliptic orbit occur at apogee and perigee points. In order to determine ΔV_{orbit} it is required to know the true anomaly.

2.2.6. Rotational Gain

The angle between the north direction and the projection of the initial orbit plane onto the launch location is called the launch azimuth β. In other words, launch azimuth is the compass heading during the launch. If the latitude of a launch site is higher than the inclination of the target orbit, the orbit cannot be reached with a direct launch. Launch azimuth required to reach an available orbit can be calculated using spherical trigonometry. For i being the target orbits inclination, ϕ is the launch site latitude. Launch azimuth β can be calculated using the following equation:

$$\beta = \sin^{-1}\left(\frac{\cos i}{\cos\phi}\right) \tag{2.36}$$

Since inclination, which is greater than the launch site latitude, gives an argument that is greater than 1 to the inverse sine function that is out of domain. Previous equation shows the limitation on the inclination mathematically. It is important to note that there are two solutions to the launch azimuth when the inclination is not equal to the launch site latitude or their sum is not equal to 180°.

During the launch, earth is rotating and effect of this rotation must be corrected for the azimuth. Geometry necessary for this correction is given in Figure 18. Angle between the rotating vector and the Earth rotation vector is the corrected azimuth β_{rot} actually headed during the launch.

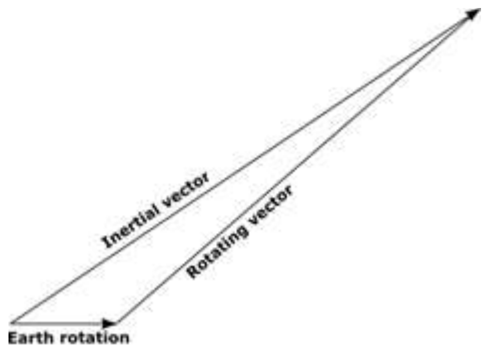

Figure 18 Effect of Earth Rotation

Depending on the inclination of the target orbit, effect of earth's rotation can be a loss or gain. For a known orbital speed and launch site rotation speed, $\Delta V_{rotation}$ can be calculated as using the following equations:

$$\vec{v}_{inertial} = v_{orbit}(\sin\beta_{inertial}, \cos\beta_{inertial}) \tag{2.37}$$

$$\vec{v}_{rot} = \vec{v}_{inertial} - \vec{v}_{earth} = v_{orbit}(\sin\beta_{inertial}, \cos\beta_{inertial}) - v_{equator}(\cos\phi, 0) \tag{2.38}$$

$\Delta V_{rotation}$ is the difference between magnitudes of the orbit velocity of the target orbit and the rotational vector. It can be either a positive or a negative value depending on the launch site latitude, launch direction and the orbit inclination.

2.2.7. Gravity Losses

By defining all the external forces acting on a launch vehicle during the flight in a simple model, gravity losses can be calculated. Using the conservation of momentum which is previously explained and equilibrium of forces acting on launch vehicle;

$$m\frac{\mathrm{d}v}{\mathrm{d}t} = F - D - mg\sin\gamma \tag{2.39}$$

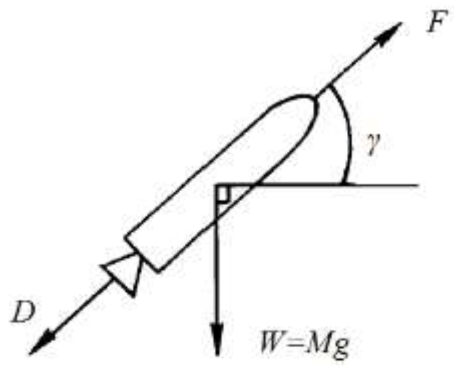

Figure 19 Simple Force Balance on a Launch Vehicle

Since drag force *D*, and losses due to drag is not the area of interest, it can be assumed to be equal to zero during the calculation of gravity losses. By multiplying remaining terms with d*t/m* it is possible to obtain *dv* as follows;

$$\mathrm{d}v = -u\frac{\mathrm{d}m}{m} - g\sin\gamma\mathrm{d}t \tag{2.40}$$

By integrating the obtained equation from t_1 to t_2, from the beginning of the burn to the end of the burn, gravity losses during the flight can be calculated as follows;

$$\Delta V_{gravity} = -\int_{t_1}^{t_2} g \sin\gamma dt \tag{2.41}$$

$\Delta V_{gravity}$ cannot be calculated easily due to the fact that both g and γ are functions of the time and they depend on the trajectory followed which requires a case specific solution. In addition, a launch mission may include coast phases on transfer orbits or delays between the ignitions of different stages. One common value for $\Delta V_{gravity}$ used for conceptual design studies of small lift launch vehicles is 1500 m/s. For an existing vehicle, this value can be calculated and compared with the proposed magnitude. However, it is difficult to find altitude and flight path angle profiles of launch vehicles in literature. For Cyclone4, gravity loss for the first 280 s of the flight is 1443 m/s in a LEO mission.

2.2.8. Other Losses

Other losses, which are stated as ΔV_{other} in the required total velocity, include losses arising from the atmospheric back pressure during the phases of the flight with active thrust, losses due to drag and steering requirements and losses occurring because of the inefficiencies at engine transients. It is not possible to directly calculate the included terms due to the fact that most of them are directly related to the trajectory flown which also depends on the mission itself. Although it is possible to use estimation for other losses term as 500 – 1500 m/s for most of the modern launch vehicles, a statistical approach can be used to obtain accurate enough values for conceptual design studies.

When the existing small-lift launch vehicles are investigated, one can obtain related performance graphs that can be used to make estimation on the sum of ΔV_{losses} term. By calculating the ideal ΔV of a launch vehicle with the properties of the stages, difference between the ideal ΔV and the orbital velocity can be found as the losses with the consideration of target orbit and the launch site location. This calculation can be repeated for different inclinations and altitudes which are also given in performance graphs as well as for different launch vehicles. An average on the calculated loss of different launch vehicles can be expressed as simple mathematical relations of target orbits altitude and inclination.

Using the tabulated data for VEGA given in Table 5, ideal ΔV of the system for the circular orbit with an altitude of 700 km and an inclination of 90^o can be calculated.

Table 5 Ideal ΔV for VEGA Launch Vehicle

1500 kg - 700 km / 90^o		
1st Stage	m_s [kg]	7431
	m_p [kg]	88365
	m_t [kg]	95796
	Isp [s]	280
2nd Stage	m_s [kg]	1845
	m_p [kg]	23906
	m_t [kg]	25751
	Isp [s]	289
3rd Stage	m_s [kg]	833
	m_p [kg]	10115
	m_t [kg]	10948
	Isp [s]	295
4th Stage	m_s [kg]	418
	m_p [kg]	550
	m_t [kg]	968
	Isp [s]	315,5
	m_{pl} [kg]	1500
	m_{plf} [kg]	490

Ideal ΔV	
m_{i1} [kg]	135453
m_{f1} [kg]	47088
Isp_1 [s]	280,0
ΔV_1 [m/s]	2902,3
m_{i2} [kg]	39657
m_{f2} [kg]	15751
Isp_2 [s]	289,0
ΔV_2 [m/s]	2617,8
m_{i3} [kg]	13416
m_{f3} [kg]	3301
Isp_3 [s]	295,0
ΔV_3 [m/s]	4058,0
m_{i4} [kg]	2468
m_{f4} [kg]	1918
Isp_4 [s]	315,5
ΔV_4 [m/s]	780,3
ΔV_{TOTAL}	**10358,4**

Orbital velocity for a circular orbit at an altitude of 700 km is 7504.4 m/s. Launch site latitude of European spaceport Kourou for VEGA launch vehicle is 5.2^{o}N. Considering the inclination of the target orbit, which is equal to 90^o, rotational speed of the earth becomes a loss instead of a gain with a magnitude of 14.3 m/s. Assuming a 1500 m/s gravity loss for the launch vehicle, velocity required to reach to the target orbit can be found as 9018.7 m/s. Difference between the ideal ΔV_{TOTAL} = 10358.4 m/s and 9018.7 m/s gives the value of other losses, ΔV_{other} = 1339.7 m/s. This loss term, that is 12.9% of the ideal ΔV, is only valid for the investigated target orbit. One can repeat the previous calculation to obtain a relation between the altitude and the losses for each LV in the area of interest, namely small-lift launch vehicles.

Variation of ΔV_{other} and the payload capacity of Athena II launch vehicle, which is also a small-lift launch vehicle, with respect to altitude for orbits with an inclination of 90° is given in Figure 20. It can be seen from the figure that ΔV_{other} is represented with a second order equation with a fit value of R^2 = 0.9989, which is a high value. Variation of ΔV_{other} for different inclinations for Athena II launch vehicle can be determined by repeating the previous calculations.

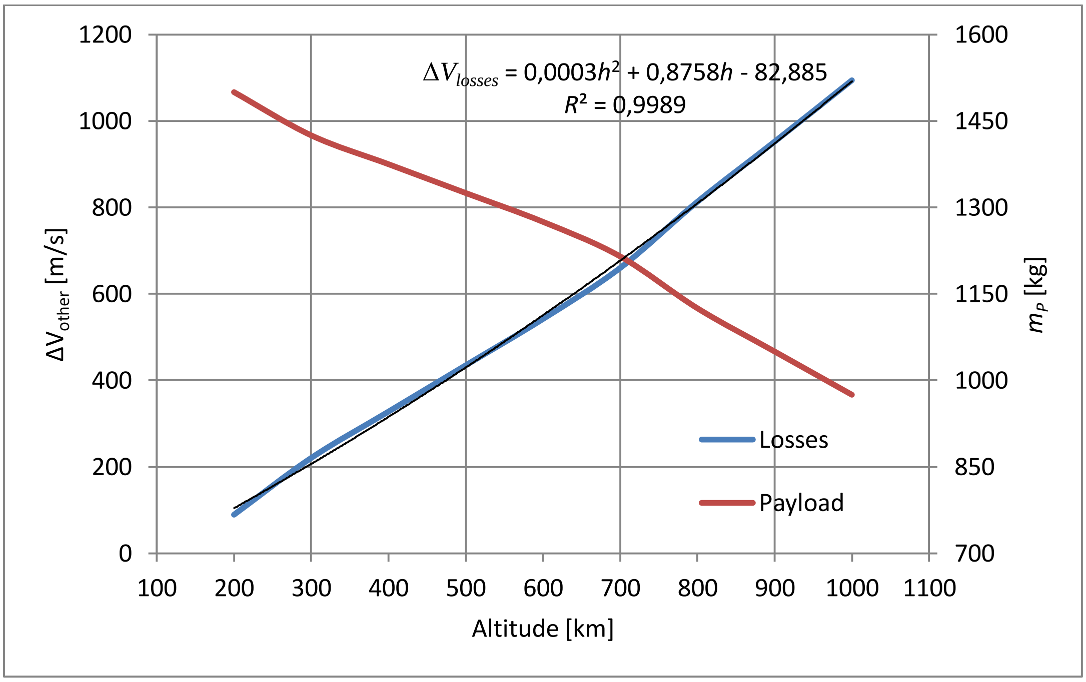

Figure 20 Athena II Launch Vehicle Losses

By determining ΔV_{other} for other small-lift launch vehicles and taking the averages by considering the common orbit altitudes and inclinations, one can determine mathematical equations accurate enough for conceptual studies. Figure 21 shows averages of ΔV_{other} variations for different inclinations.

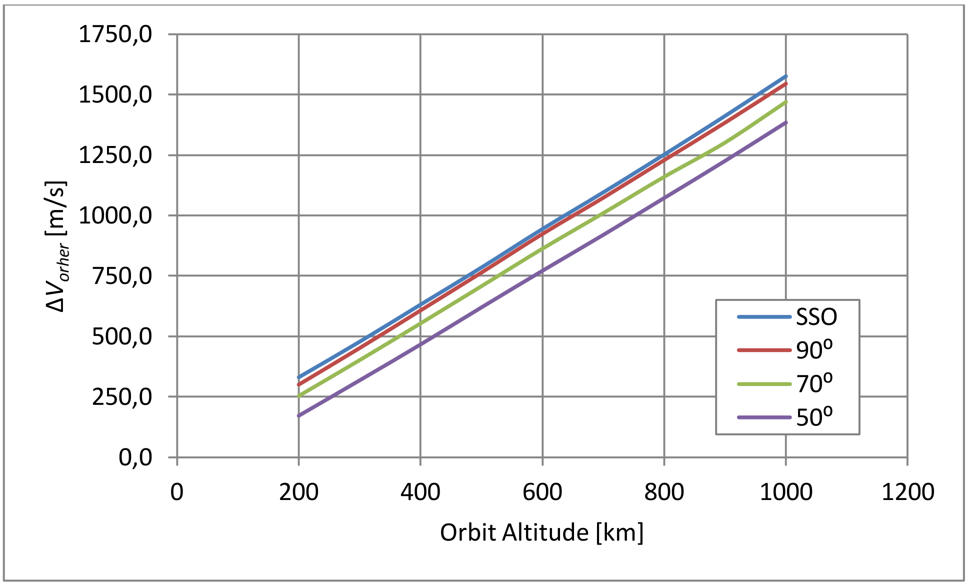

Figure 21 Variation of ΔV_{other} for Different Inclinations

Obtained second order polynomials are easy to implement on any calculation that can be used in conceptual studies. This kind of approximation gives accurate results and does not depend on an initial estimation. Instead, it considers the properties of existing launch vehicles which any new design will also be similar to. Small-lift launch vehicles Athena I and II, Atlas 3B, Kosmos, Minotaur, Rockot, Start, Taurus 2110 and VEGA are used to obtain the relation between ΔV_{other} and the target orbit. Obtained result is independent of the launch site latitude due to the fact that the rotation effect of earth is subtracted from the ideal ΔV values.

2.2.9. Optimum Mass Distribution

Simplest form of the Tsiolkovsky rocket equation which gives the ideal velocity increment that can be obtained by a single stage rocket is given in Equation (2.15). For m_0 and m_f being the initial and final masses, respectively, mass ratio can be defined as follows;

$$MR = \frac{m_0}{m_f} \tag{2.42}$$

Distribution of masses along different parts of a rocket can be used to define a payload ratio λ which can be used to find the optimum mass distribution of mass among stages for minimum liftoff weight.

$$m_0 = m_p + m_s + m_{pl} \tag{2.43}$$

where m_p is the mass of the propellant available for the propulsion, m_{pl} is the payload mass and m_s is the structural mass of the vehicle including all other necessary masses like tanks, engines, structures or flight computer to build and operate a LV. If complete use of propellant is assumed, which is never the real case due to residues remaining after the operation or unburned propellant because of different reasons, final mass of the vehicle m_f becomes the sum of structural masses and payload mass.

$$m_f = m_s + m_{pl} \tag{2.44}$$

Payload ratio λ and structural coefficient ε of a vehicle under the no ullage is defined in the following equations.

$$\lambda = \frac{m_{pl}}{m_0 - m_{pl}} = \frac{m_{pl}}{m_p + m_s} \tag{2.45}$$

$$\varepsilon = \frac{m_s}{m_p + m_s} = \frac{m_f - m_{pl}}{m_p + m_s} \tag{2.46}$$

For the complete use of the loaded propellant, mass ratio of the vehicle becomes a function of payload and structural ratios.

$$MR = \frac{m_p + m_s + m_{pl}}{m_s + m_{pl}} = \frac{1 + \lambda}{\varepsilon + \lambda} \tag{2.47}$$

It is obvious from the mass ratio that small structural ratios are advantageous but attainable advantage of using lighter materials is limited due to technological development level of materials and consequently structures. Vehicle staging is another way to improve the performance of a launch vehicle as explained previously. It is required to analyze multistage vehicle with sequential operation in order to be able to determine the optimum mass distribution among stages. For a multistage vehicle, subscript *n* can be used to define the stages.

m_{0_n} initial mass of nth stage with the upper stages and payload

m_{f_n} final mass of the nth stage with the upper stages and payload

m_{s_n} structural mass of the nth stage alone

m_{p_n} propellant mass of the nth stage

Considering this definition, payload for every stage is the main payload plus the weight of any stage above the stage in consideration. Initial mass of every stage on a vehicle with this definition is as follows;

$$m_{0_n} = m_{p_n} + m_{s_n} \, \mathrm{m}_{0_{n+1}} \tag{2.48}$$

Payload and structural ratios of each stage in addition to the mass ratio in terms of these two for a multistage vehicle is given in the following equations.

$$\lambda_n = \frac{\mathrm{m}_{0_{n+1}}}{\mathrm{m}_{0_n} - \mathrm{m}_{0_{n+1}}} \tag{2.49}$$

$$\varepsilon_n = \frac{m_{s_n}}{\mathrm{m}_{0_n} - \mathrm{m}_{0_{n+1}}} = \frac{m_{f_n} - \mathrm{m}_{0_{n+1}}}{\mathrm{m}_{0_n} - \mathrm{m}_{0_{n+1}}} \tag{2.50}$$

$$MR_n = \frac{\mathrm{m}_{0_n}}{m_{f_n}} = \frac{1+\lambda_n}{\varepsilon_n + \lambda_n} \tag{2.51}$$

Applying the rocket equation to each stage and sum of the velocity increments of the stages gives the total ΔV of a multistage launch vehicle.

$$\Delta V_{TOTAL} = \sum_{n=1}^{N} \Delta V_n = \sum_{n=1}^{N} g_0 I_s \ln MR_n \tag{2.52}$$

For the simplest case, where the structural mass ratio ε and specific impulse I_s of all stages are equal, maximum velocity for a given payload mass and initial vehicle mass can be obtained by setting $\lambda_n = \lambda$ for all stages as

$$\lambda = \frac{(m_{pl} / m_{01})^{1/N}}{1 - (m_{pl} / m_{01})^{1/N}} \tag{2.53}$$

where N is the number of stages on the launch vehicle. However, stages with identical performance are not practical due to several reasons such as different structural requirements of stages or even the expansion ratios of the engines. For fixed I_s but varying ε, maximum velocity for the given initial vehicle mass occurs when the following payload ratio condition is satisfied.

$$\lambda_n = \frac{\alpha \varepsilon_n}{1 - \varepsilon_n - \alpha} \tag{2.54}$$

α used in the above equation is the Lagrange multiplier obtained by the ratio of the payload mass to the initial mass.

$$\frac{m_{pl}}{m_{01}} = \prod_{n=1}^{N} \frac{\lambda_n}{1 + \lambda_n} = \prod_{n=1}^{N} \frac{\alpha \varepsilon_n}{1 - \varepsilon_n - \alpha + \alpha \varepsilon_n} \tag{2.55}$$

Most applicable case for the payload ratio is the variable performance variables I_s and ε.

$$\lambda_n = \frac{\alpha \varepsilon_n}{g_0 I_{s^n} (1 - \varepsilon_n) - \alpha} \tag{2.56}$$

Similarly, related Lagrange multiplier can be found using Equation (2.57).

$$\frac{m_{pl}}{m_{01}} = \prod_{n=1}^{N} \frac{\lambda_n}{1 + \lambda_n} = \prod_{n=1}^{N} \frac{\alpha \varepsilon_n}{(1 - \varepsilon_n)(g_0 I_{s^n} - \alpha)} \tag{2.57}$$

Knowing the value of payload ratio λ_n, mass ratio of each stage on the launch vehicle can be found simply by using the Equation (2.51). In order to obtain the minimum gross lift-off weight for the desired payload mass m_{pl} and known performance variables I_s and ε,

$$\lambda_n = \frac{1-\varepsilon_n MR_n}{MR_n - 1} \tag{2.58}$$

where mass ratio for stages are

$$MR_n = \frac{1+\alpha g_0 I_{s\,n}}{\alpha \varepsilon_n g_0 I_{s\,n}} \tag{2.59}$$

Finally, Lagrange multiplier α for the minimum gross lift-off weight depending on the final velocity requirement can be found by the relation,

$$\Delta V = \sum_{n=1}^{N} g_0 I_{s\,n} \ln\left(\frac{\alpha g_0 I_{s\,n} + 1}{\alpha \varepsilon_n g_0 I_{s\,n}}\right) \tag{2.60}$$

To be able to determine the mass distribution for different cases, it is required to obtain the roots of the Equations (2.55), (2.57) and (2.60) numerically. It is also important to note that given equations are not directly applicable to the launch vehicle configurations with boosters or parallel stages. It is also not possible to directly implement the effect of fairing jettisoning without certain assumptions. A numerical solution is required to find the related Lagrange multiplier. It is impractical for launch vehicles with boosters and it is complex for consideration of fairing jettisoning. Application of optimum mass distribution with Lagrange multiplier is not useful and is not preferable in finding the mass distribution of launch vehicles. In addition, optimum mass distribution does not always provide the most effective solution in terms of LV costs which will be explained in the following sections.

CHAPTER 3

ATMOSPHERIC PARAMETERS

Atmosphere can be defined as the gaseous envelope surrounding the earth which starts from the surface and reaches altitudes up to 1000 km. Below 50 km, atmosphere can be assumed as a homogeneous ideal gas. From Earth surface to 10 km and between the altitudes of 45 km and 95 km temperature drops while for 10 to 45 km and 95 to 400 km temperature increases. For altitudes above 400 km, temperature of the atmosphere is nearly constant.

Launch vehicle design is strongly dependent on atmospheric parameters such as temperature, pressure and density. In addition to the calculation of aerodynamic forces, atmospheric parameters are required for determination of structural and thermal loads. Atmospheric constituents, viscosity, solar radiation and molecular oxygen density are other examples of models' contents. Many global and regional atmosphere models are developed using direct measurements and statistical approaches by different organizations. International Civil Aviation Organization (ICAO), Committee on Space Research (COSPAR) and Committee on Extension to the Standard Atmosphere (COESA) are the examples of related organizations.

There are two main types of atmosphere models. Reference atmosphere models are generally specialized for specific geographic regions of earth while standard atmosphere models' coverage is global. It is also possible to classify the models according to the covered altitude interval. For example, Jacchia J70, a thermosphere model, covers the altitudes between 90 km and 2500 km. This model is intended for the drag calculation of operational satellites in orbit. Jacchia J70 is not suitable for design studies of launch vehicles and sounding rockets due to the lack of coverage of lower layers of the atmosphere.

3.1. US STANDARD ATMOSPHERE 1976

The US Standard Atmosphere, 1976 (US1976) is a revision of the previous version (US1962) and generated with the additional knowledge of the upper atmosphere. This common model depends on rocket data for the altitudes above 50 km. It is identical with ICAO Standard 1962 up to 32 km and ISO Standard up to 50 km altitude. Maximum altitude for the main tables of atmospheric properties is 1000 km. This model represents the variation of pressure, temperature, density and viscosity of the earth's atmosphere over a range of altitudes. This model also discusses the variations of real values of these parameters from the averaged ones. Although seasonal and regional differences of the parameters are mentioned in the model, it is specifically developed for 45° north latitude.

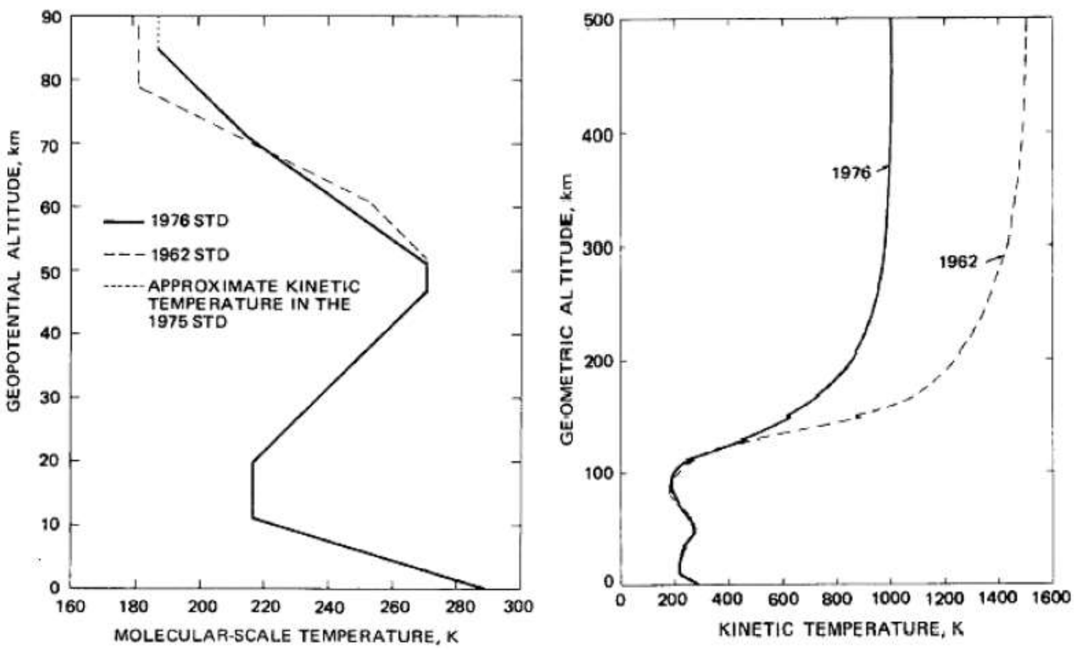

Figure 22 Variation of Temperature in US1976 and US1962 (NASA)

Variation of temperature and pressure is modelled linearly for different layers of atmosphere in US1976. On the other hand, air density must be calculated using the ideal gas law. Variation of parameters and their initial values with respect to altitude is given in Table 6. Mesopause and thermosphere are the following layers of the Earth's atmosphere.

Table 6 Properties of Atmosphere Layers in US1976

	Altitude from MSL [m]	Static Pressure [Pa]	Standard Temperature [K]	Temperature Lapse Rate [K/km]
Troposphere	0	101325.00	288.15	-6.50
Tropopause	11000	22632.10	216.65	0.00
Stratosphere	20000	5474.89	216.65	1.00
Stratosphere	32000	868.02	228.65	2.80
Stratopause	47000	110.91	270.65	0.00
Mesosphere	51000	66.94	270.65	-2.80
Mesosphere	71000	3.96	214.65	-2.80

Although US1976 is applicable for most cases of trajectory simulations with three degrees of freedom, insufficient variability data makes this model inapplicable for safety analysis. US1976, which is developed for 45^{o} (average latitude of US), is not accurately compatible for other locations and does not include information about wind profiles.

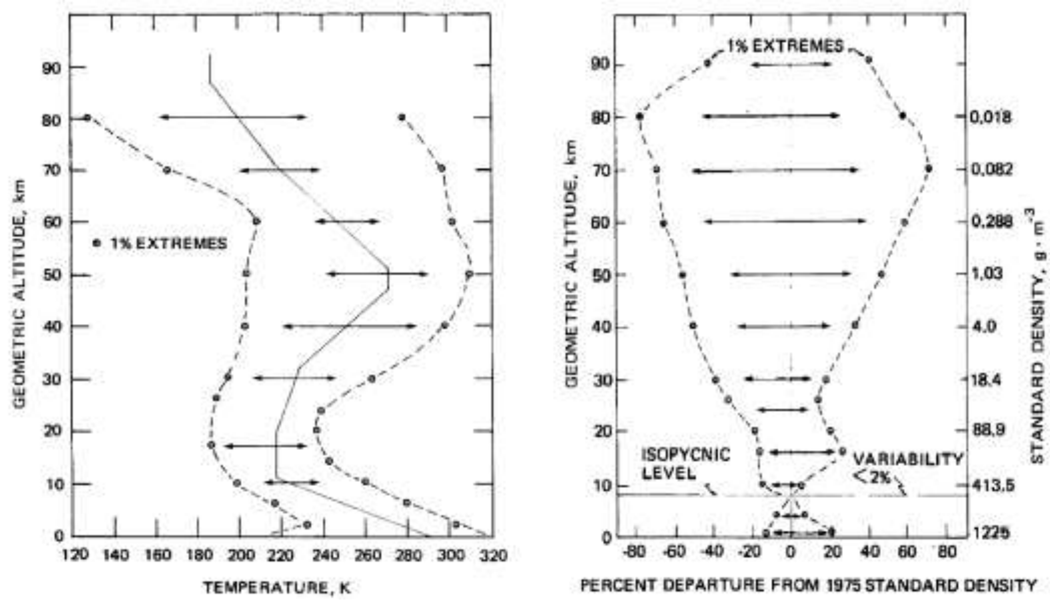

Figure 23 Temperature and Density Variability in US1976 (NASA)

3.2. NRLMSISE-00

NRLMSISE-00 (US Naval Research Laboratory Mass Spectrometer and Incoherent Scatter Radar Exosphere 2000) is an empirical model of the atmosphere. Although primary purpose of this model is to help satellite orbital decay predictions, it extends from earth surface to high altitudes. This model is based on sounding rockets, satellite and incoherent scatter radar measurements. Year and day, time of the day, geographical location and solar properties are the additional inputs to geodetic altitude for the operation of the global model. Comparison of temperature and density variations for different seasons is given in Figure 24.

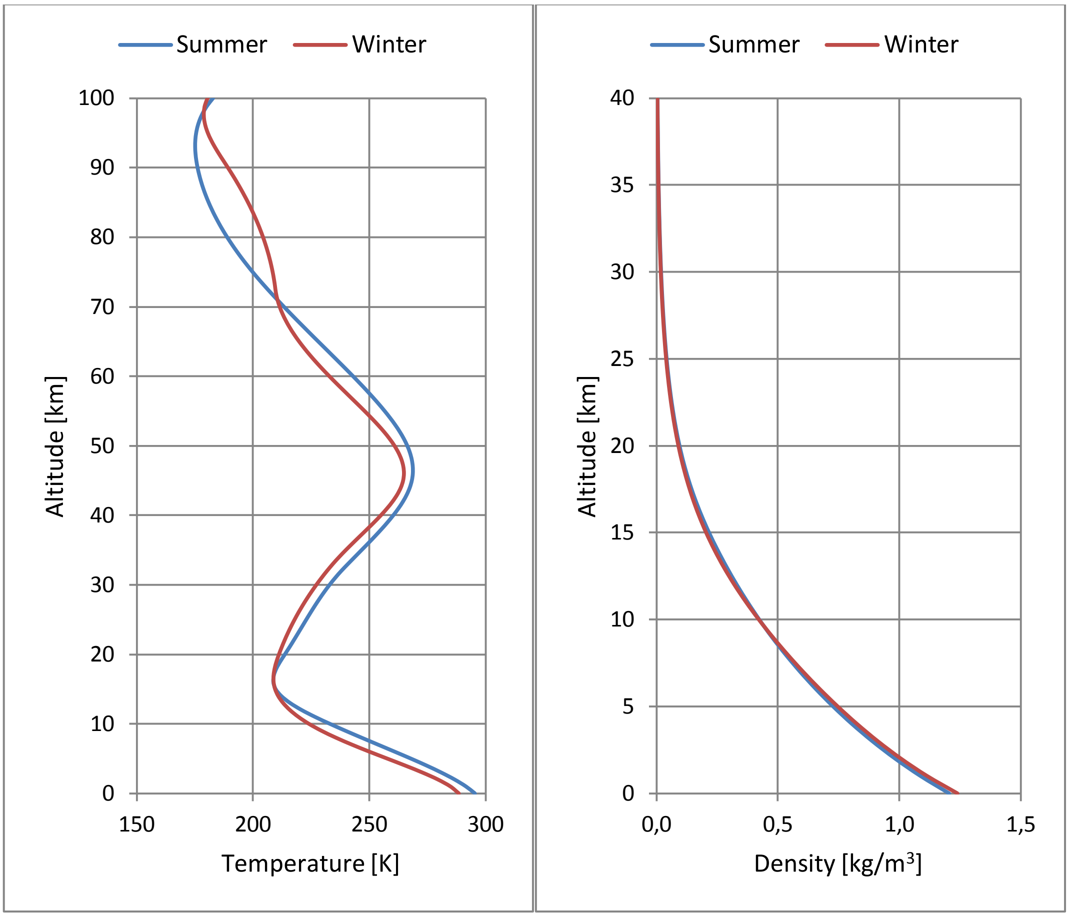

Figure 24 Temperature and Density Variation at Different Seasons

Results of this model are more compatible with sounding balloon measurements for low altitudes when compared to US1976. Similar to US1976, this model also does not include information for deviations of the parameters from their mean values and information for wind profiles which are required for safety analysis.

3.3. GLOBAL GRIDDED UPPER AIR STATISTICS (GGUAS)

Climatological data is better than using standard atmosphere model which assumes a worldwide state for atmospheric properties when data are a function of geographical position and seasons. The source of the GGUAS data set was the European Centre for Medium-Range Weather Forecasts and is based on twice daily sounding balloon measurements of 12 years period. GGUAS data set defines the atmosphere for each month of the year with a 2.5 degree spaced global grid and provides a 73 x 144 resolution. Mean and standard deviation of different parameters for 15 pressure levels are provided in the data set. Provided parameters are altitude, temperature, density, dew point temperature and wind components.

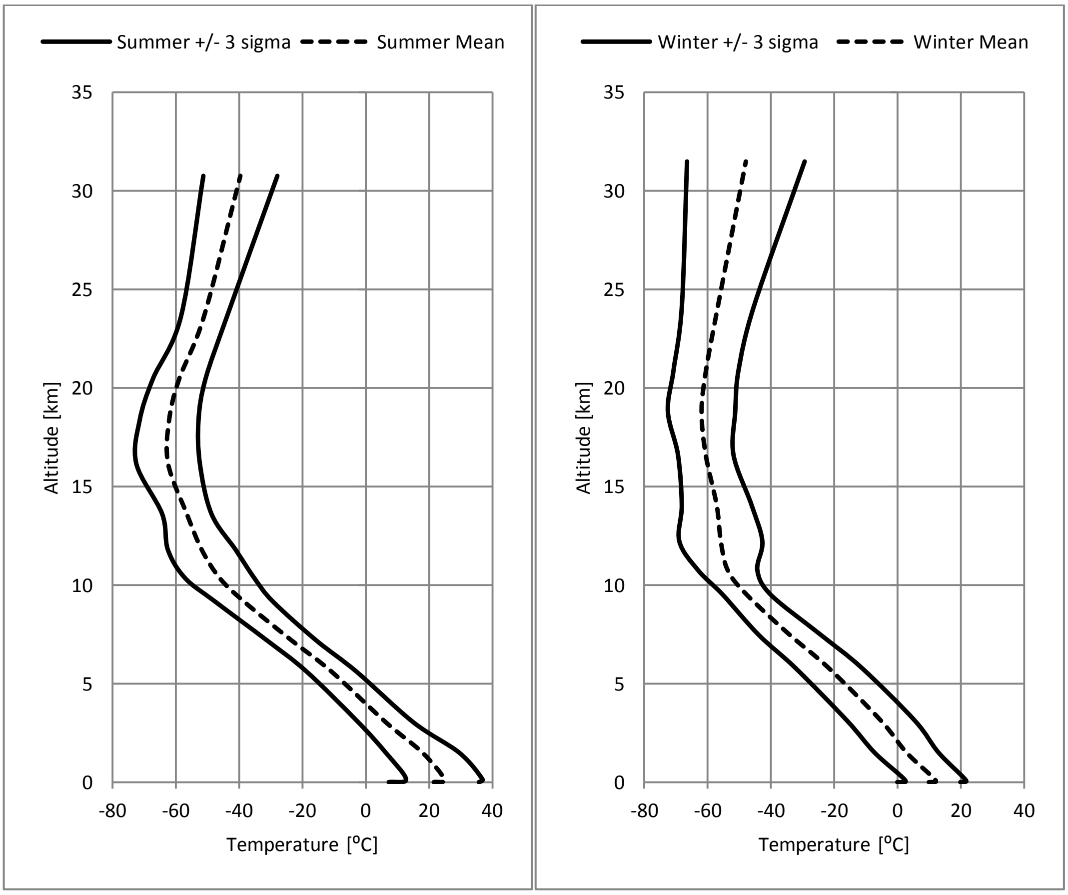

Figure 25 Variation of Temperature at Different Seasons

Temperature variations of summer and winter conditions for the selected location are compared in Figure 25. GGUAS also includes the variations of density with the standard deviations. Variation of density and percent deviations of density from the mean values are given in Figure 26.

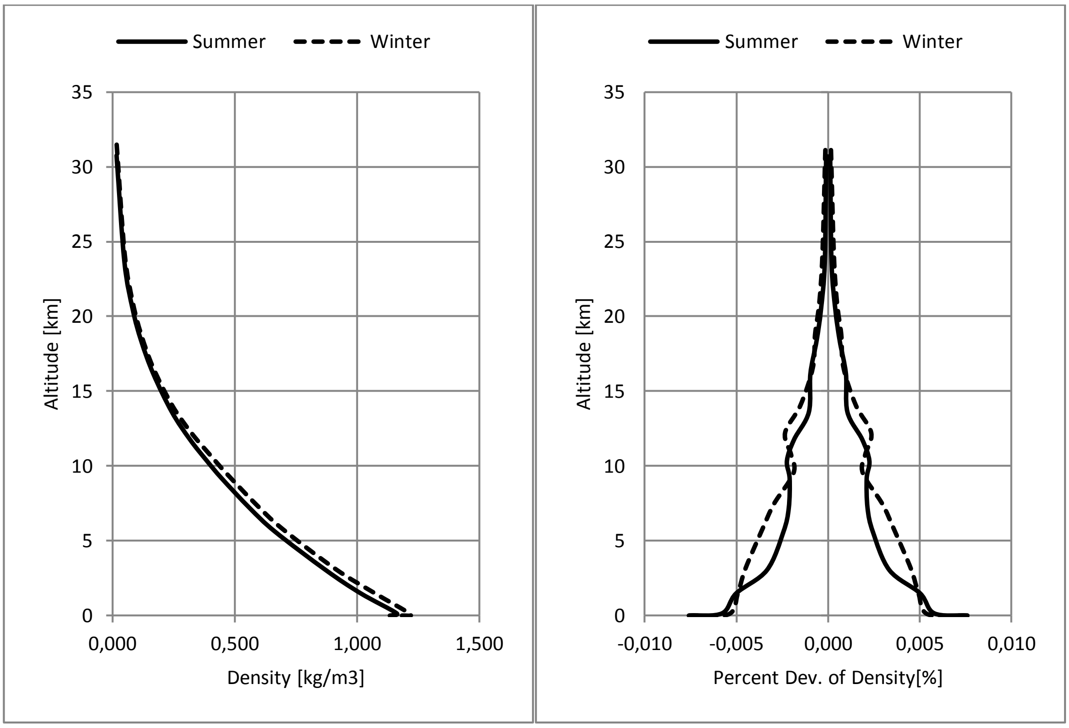

Figure 26 Density Variations and Percent Deviations

There are eight radiosonde stations in Turkey as an essential source of meteorological data for high altitudes. Locations of these stations at which twice daily measurements are taken is shown in Figure 27.

Radiosonde measurements of Isparta (37.45° N, 30.33° E) and GGUAS statistics (grip point located at 37.50° N, 30.00° E) are compared from surface (about 900 mbar) up to the altitude of 30 km (about 10 mbar) which is the general ceiling height for such measurements. Mean values of temperature for GGUAS are confirmative with the actual data. In addition, ±6σ values of GGUAS data covers most of the temperature extremes. Same conformation is also applicable for the density variation. Eventually, comparison of the GGUAS statistics with actual data shows that using climatological data is better in terms of validity.

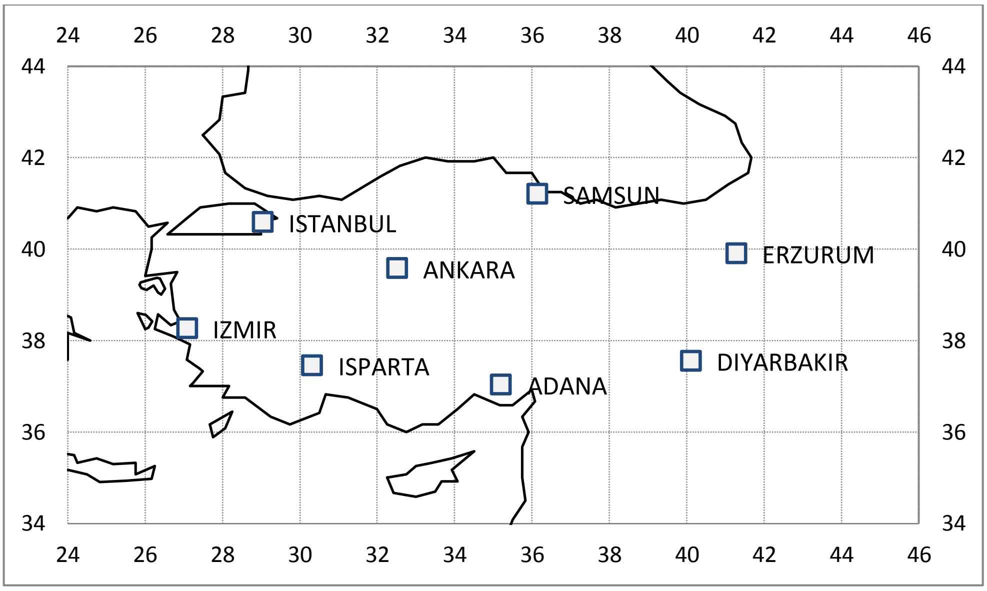

Figure 27 Radiosonde Stations of Turkey

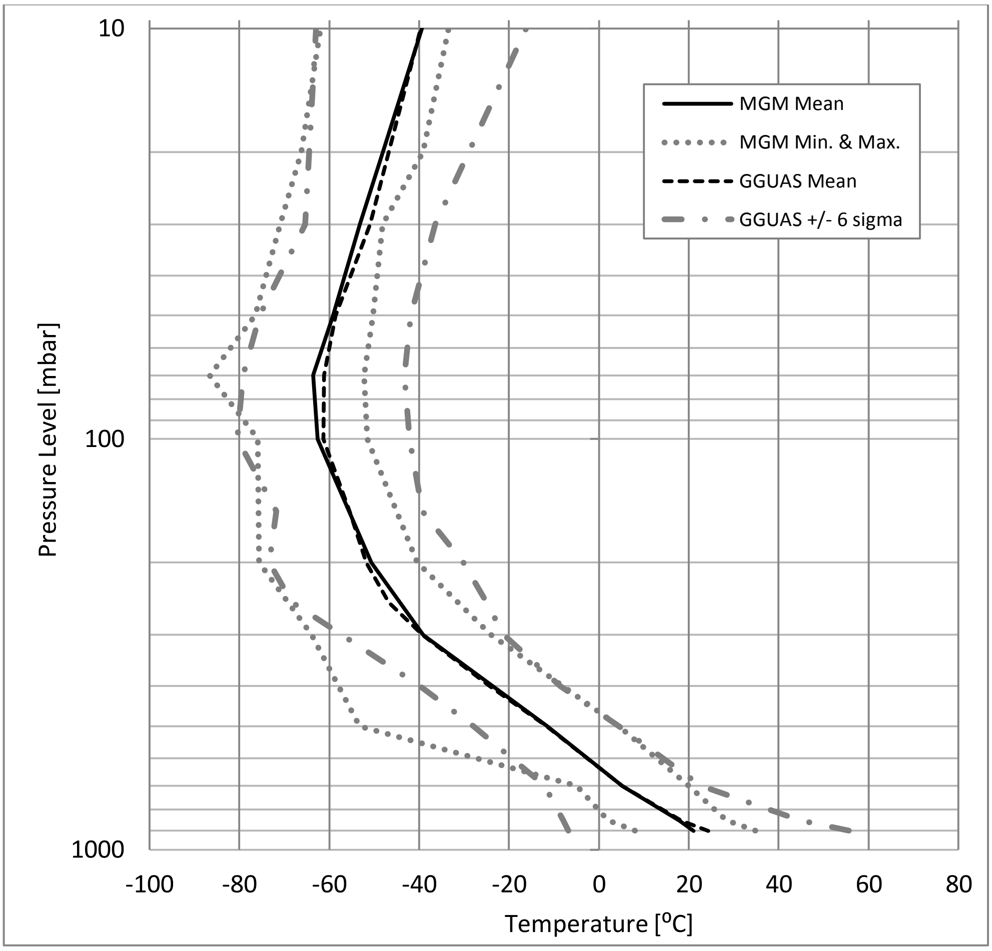

Figure 28 Comparison for GGUAS and Actual Values of Isparta

3.4. NASA MONTHLY VECTOR WIND PROFILE MODEL

An accurate knowledge of atmospheric parameters is necessary for design activities of space vehicles. Wind data is required for both launch vehicle design and safety analysis. Winds are three dimensional motions of the air and their variations are function of seasons and geographic location of space launch center. NASA provided guidelines on natural environmental conditions for different geographical locations on several references. It is the common practice to use synthetic wind profiles during initial phases of the design studies.

A technique to develop a synthetic vector wind profile model of interest for launch vehicle applications is presented in NASA's Monthly Vector Wind Profile Model. This model uses coefficients of correlations with means and standard deviations in a statistical manner to obtain wind profiles. It is based on the concept that the wind components of vectors at two different altitudes have a quadravariate normal probability distribution function. For a given wind vector, the conditional distribution of the wind components at another altitude is bivariate normal.

3.4.1. Coordinate Systems

Standard meteorological coordinate system has been chosen for wind statistics due to the fact that statistical tools are already available in Cartesian form. Wind measurements are obtained in terms of direction and magnitude for meteorological purposes. Wind direction is expressed in degrees clockwise from north and is the direction from which the wind is blowing. Wind magnitude is the scalar quantity and is referred as scalar wind or wind speed. The meteorological coordinate system is given in Figure 29. Zonal and meridional components define Cartesian form while W and θ define polar form. Variables of the system are defined as follows;

- W Wind speed or scalar wind [m/s]
- θ Wind direction measured cw from north
- U Zonal wind component, positive from west to east [m/s]
- V Meridional wind component, positive from south to north [m/s]

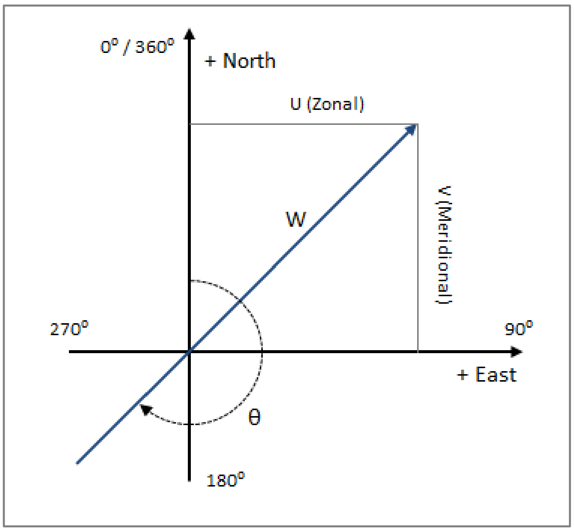

Figure 29 The Meteorological Coordinate System

3.4.2. Statistical Model and Wind Profile Construction

It is assumed that wind components at two different altitudes, U_1, V_1 and U_2, V_2 respectively, are quadravariate normal distributed. Mean values of the wind components, their standard deviations and correlation coefficients are required to obtain the distribution. There are 14 different parameters required for the definition.

For a given wind vector (U_1, V_1) at a reference altitude H_1, the distribution of wind vector (U_2, V_2) at any other altitude H_2 is conditional bivariate normal. Parameters required for the conditional bivariate normal distribution are the conditional means, conditional standard deviations and the conditional correlation coefficient.

$$f\left\langle U_2, V_2 \middle| U_1^*, V_1^* \right\rangle = \frac{f\left\langle U_1, V_1, U_2, V_2 \right\rangle}{f\left\langle U_1^*, V_1^* \right\rangle} \tag{3.1}$$

Statistical parameters for quadravariate normal distribution are four mean values, MU_1, MV_1, MU_2 and MV_2, four standard deviations, SDU_1, SDV_1, SDU_2 and SDV_2 and four correlation coefficients of same variables between altitudes H_1 and H_2, $R(U_1,U_2)$, $R(V_1,V_2)$, $R(U_1,V_2)$ and $R(V_1,U_2)$ in addition to the two coefficients for different variables at the same altitude $R(U_1,V_1)$ and $R(U_2,V_2)$.

Wind profile construction is a straight forward application of the statistical model. First step is to define number of specific wind vectors and determine the clocking angle measured counterclockwise from the centroid to the desired probability ellipse at a fixed reference altitude. For 12 specific wind vectors clocking angle will be 30°. Probability ellipse of 99.73 % is selected which corresponds to the 3.0 σ. Wind vector components from the centroid of the probability ellipse can be obtained by using the following equations.

$$U_1^* = MU_1 + RS\cos(CA) \tag{3.2}$$

$$V_1^* = MV_1 + RS\sin(CA) \tag{3.3}$$

$$RS = \frac{1}{A/\sqrt{-2\ln(1-P)}} \tag{3.4}$$

$$A^2 = \frac{1}{1-R(U_1,V_1)^2}\left[\begin{array}{l}\left(\dfrac{\cos(CA)}{SDU_1}\right)^2 + \left(\dfrac{\sin(CA)}{SDV_1}\right)^2 \ldots \\ \ldots - \dfrac{2R(U_1,V_1)\cos(\mathrm{CA})\sin(\mathrm{CA})}{SDU_1 SDV_1}\end{array}\right] \tag{3.5}$$

An example probability ellipse is given for MU_1 = 8.9 m/s, MV_1 = -1.3 m/s, SDU_1 = 5.4 m/s, SDV_1 = 6.0 m/s and $R(U_1,V_1)$ = 0.2 in Figure 30.

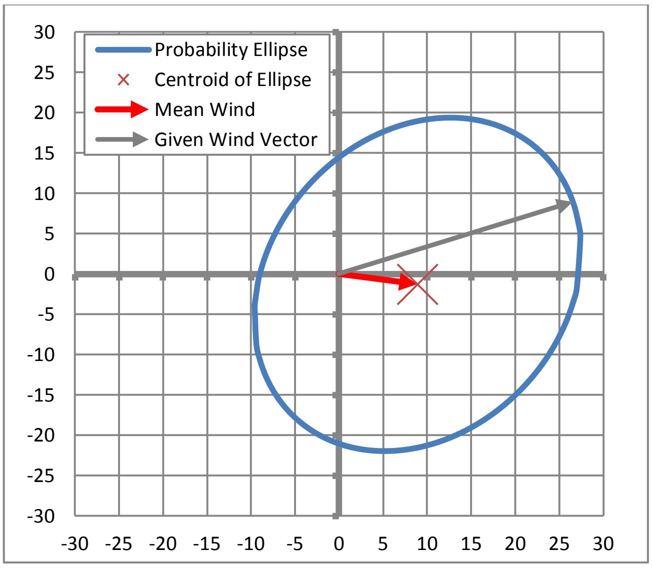

Figure 30 Example Probability Ellipse

Second step is to compute the five parameters for the conditional bivariate normal probability distributions for all altitude levels below and above the reference altitude. For each of the specific wind vector $\{U_1^*, V_1^*\}$ at the reference altitude, the conditional mean vector $\{CMU_2, CMV_2\}$ can be calculated using the equations;

$$CMU_2 = MU_2 + \frac{(T_1 + T_2)}{1 - R(U_1, V_1)^2} \tag{3.6}$$

$$T_1 = \left[R(U_1, U_2) - R(U_1, V_2) R(U_1, V_1)\right](U^*_1 - MU_1)(SDU_2 / SDU_1) \tag{3.7}$$

$$T_2 = \left[R(U_1, V_2) - R(U_1, U_2) R(U_1, V_1)\right](V^*_1 - MV_1)(SDU_2 / SDV_1) \tag{3.8}$$

$$CMV_2 = MV_2 + \frac{(T_3 + T_4)}{1 - R(U_1, V_1)^2} \tag{3.9}$$

$$T_3 = \left[R(V_1, U_2) - R(V_1, V_2) R(U_1, V_1)\right](U^*_1 - MU_1)(SDV_2 / SDU_1) \tag{3.10}$$

$$T_4 = \left[R(V_1, V_2) - R(V_1, U_2) R(U_1, V_1)\right](V^*_1 - MV_1)(SDV_2 / SDV_1) \tag{3.11}$$

The conditional standard deviations $CSDU_2$ and $CSDV_2$ and the conditional correlation coefficient can be calculated from the symmetric covariance matrix Sigma. Note that for the calculation of standard deviations and correlation coefficient, components of specific wind vector are not required.

$$\begin{aligned}\text{Sigma}(1,1) = SU_2 &- SU_1U_2(SU_1U_2 \times SV_1 - SU_2V_1 \times SU_1V_1) / D \\ &- SV_1U_2(-SU_1U_2 \times SU_1V_1 + SU_2V_1 \times SU_1) / D\end{aligned} \tag{3.12}$$

$$\begin{aligned}\text{Sigma}(2,2) = SV_2 &- SU_1V_2(SU_1V_2 \times SV_1 - SV_1V_2 \times SU_1V_1) / D \\ &- SV_1V_2(-SU_1V_2 \times SU_1V_1 + SV_1V_2 \times SU_1) / D\end{aligned} \tag{3.13}$$

$$\begin{aligned}\text{Sigma}(1,2) = SU_2V_2 &- SU_1V_2(SU_1U_2 \times SV_1 - SU_2V_1 \times SU_1V_1) / D \\ &- SV_1V_2(SU_1U_2 \times SU_1V_1 + SU_2V_1 \times SU_1) / D\end{aligned} \tag{3.14}$$

$$D = SU_1 \times SV_1 - \left(SU_1V_1\right)^2 \tag{3.15}$$

$$CSDU_2 = \sqrt{\text{Sigma}(1,1)} \tag{3.16}$$

$$CSDV_2 = \sqrt{\text{Sigma}(2,2)} \tag{3.17}$$

$$CR(U_2, V_2) = \frac{\text{Sigma}(1,2)}{CSDU_2 \times CSDV_2} \tag{3.18}$$

where SU_1, SV_1, SU_2 and SV_2 are variances, SU_1V_1, SU_2V_2, SU_1U_2, SV_1V_2, SU_1V_2 and SV_1U_2 are the covariances. Covariance can be obtained by the product of standard deviations and correlation coefficient. For variables A and B, covariance can be obtained by the following equation.

$$SAB = R(A,B) \times SDA \times SDB \tag{3.19}$$

Five conditional statistical parameters given in Equations (3.6), (3.9), (3.16), (3.17) and (3.18) can be used to compute the conditional bivariate normal probability ellipse to approximate the largest shear. For the largest shear between the altitudes a new clocking angle 180° from the previous one should be selected.

Final step of the conditional wind vector derivation is to find (UC_2, VC_2) and adding the resulting vector to (U_1, V_1) to obtain (U_2, V_2) at new altitude H_2.

$$UC_2 = CMU_2 + RSC\cos(CC) \tag{3.20}$$

$$VC_2 = CMV_2 + RSC\sin(CC) \tag{3.21}$$

$$CC = CA + 180^{\circ} \tag{3.23}$$

$$RSC = \frac{1}{A_c / \sqrt{-2\ln(1-P)}} \tag{3.24}$$

$$A_c^{\,2} = \frac{1}{1 - CR(U_1,V_1)^2}\begin{bmatrix} \left(\dfrac{\cos(CC)}{CSDU_1}\right)^2 + \left(\dfrac{\sin(CC)}{CSDV_1}\right)^2 \ldots \\ \ldots - \dfrac{2CR(U_1,V_1)\cos(CC)\sin(CC)}{CSDU_1 \times CSDV_1} \end{bmatrix} \tag{3.25}$$

A schematic of the profile construction for a specific wind vector at 12 km and the required conditional wind vector is illustrated in Figure 31.

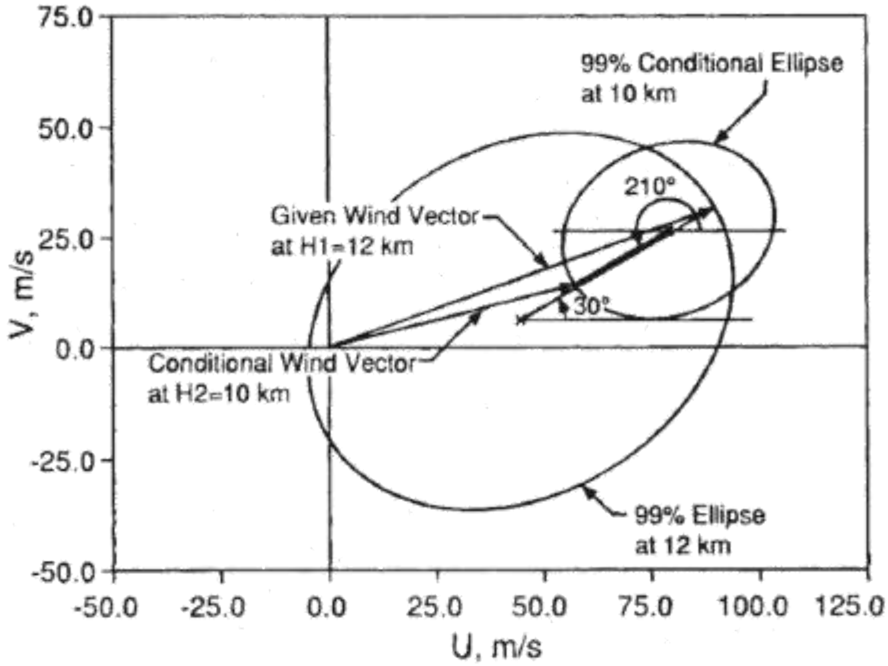

Figure 31 Schematic of Wind Profile Construction (NASA)

To complete the profile construction, this process should be repeated between all other altitude levels and the reference altitude. Profile construction using this methodology is appropriate for monthly reference periods and the designer may choose as few as two months to present the annual wind dispersions.

Although generation of a wind profile for maximum wind shear is useful for launch vehicle preliminary design studies, generating random profiles that satisfies the probability distributions or profiles at the maximum envelope is more appropriate. For this purpose, *CC* angle can be replaced by a random number or zero respectively.

Probability ellipses that represent the envelope for the wind profile at 36.25° N, 29.25° E coordinate are given in Figure 32 and Figure 33.

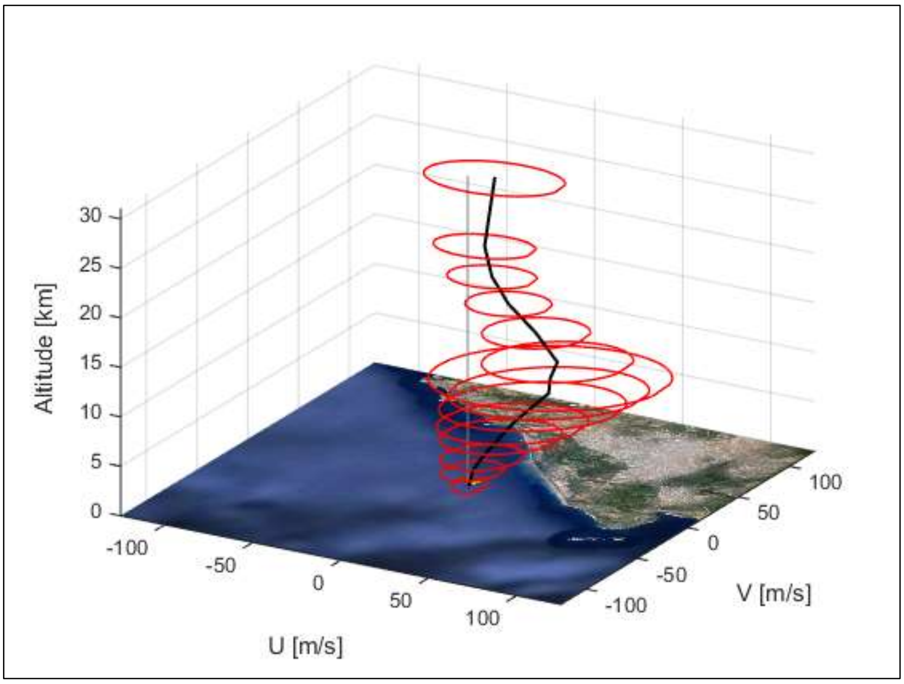

Figure 32 3D View of Wind Profile Envelope

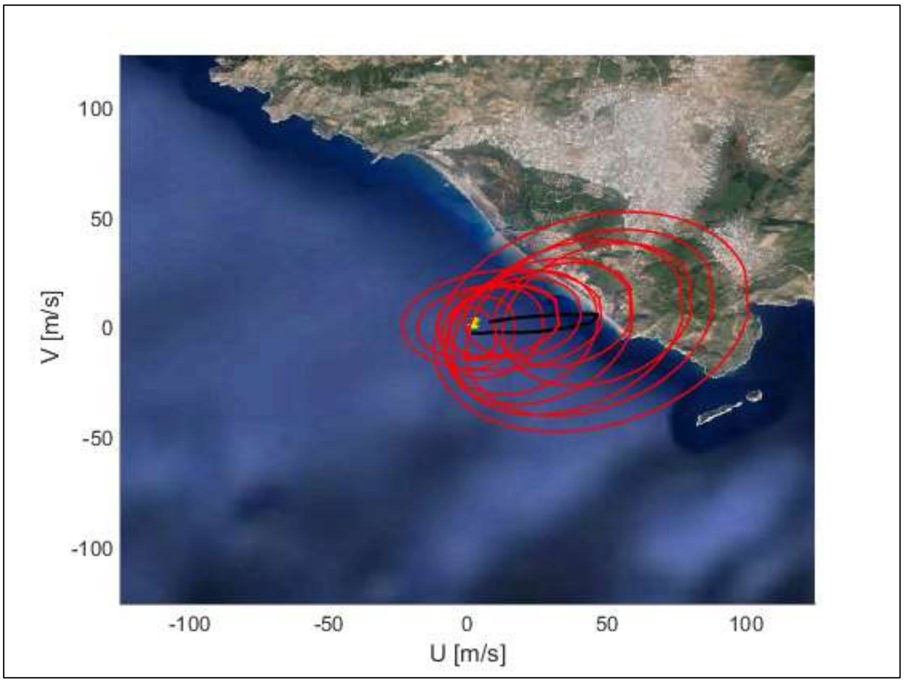

Figure 33 2D View of Wind Profile Envelope

CHAPTER 4

SPACE LAUNCH CENTER SAFETY

There are two types of possible main hazards at the launch vehicle flight phase. First type is the possible launch vehicle fragments in case of an inflight explosion or in case of an operation of perfectly functioning flight termination system. In this case, depending on the flight phase, fragments follow a ballistic trajectory under the influence of atmospheric conditions. Due to additional velocity imparted to the fragments, winds and aerodynamic properties of the fragments, launch vehicle parts can drift long distances. Other type of main hazards is the secondary effects. Overpressure in front of a shock wave which is formed as a result of launch vehicle explosion, high temperature thermal impact of explosion products and toxic pollution of atmosphere with the combustion and explosion products are the secondary effects.

4.1. LAUNCH VEHICLE FRAGMENTS

Defining debris characteristic is necessary for trajectory and impact modelling in safety analysis. In case of a launch vehicle break-up, ballistic trajectories of the defined fragments can be determined using the information on imparted velocities and debris definitions such as numbers, sizes and aerodynamic characteristics. Uncertainties in debris trajectories are due to following items.

- Ballistic Coefficient Uncertainty
- Imparted Velocity Dispersion
- Wind Uncertainty

Launch vehicle configuration is required for realistic determination of debris characteristics. Selection of LV configuration for safety analysis is explained in Appendix B.

4.1.1. Ballistic Coefficient

The ballistic coefficient is an important parameter that indicates the importance of aerodynamic and inertial forces on a fragment in a ballistic trajectory. It is often referred as beta (β) and defined as the ratio of drag coefficient (C_D) multiplied by the reference area (A_{ref}) to the weight of the fragment (W).

Drag coefficient is the ratio of the total drag force exerted on an object to the force due to dynamic pressure acting on the reference area. Although A_{ref} can be set to any value, it is often equal to the projected area to the direction of the flow. Drag coefficients of different fragment types are typically derived experimentally. It has a significant dependence on the objects shape, Mach number and Reynolds number.

Low ballistic coefficient debris tends to impact far downrange due to the fact that they are relatively uninfluenced by the wind. On the other hand, higher ballistic coefficient debris will slow down rapidly under the influence of dynamic pressure and will fall more directly below the break-up point being carried in the wind direction. Influence of ballistic coefficient and wind on debris trajectory is represented in Figure 34.

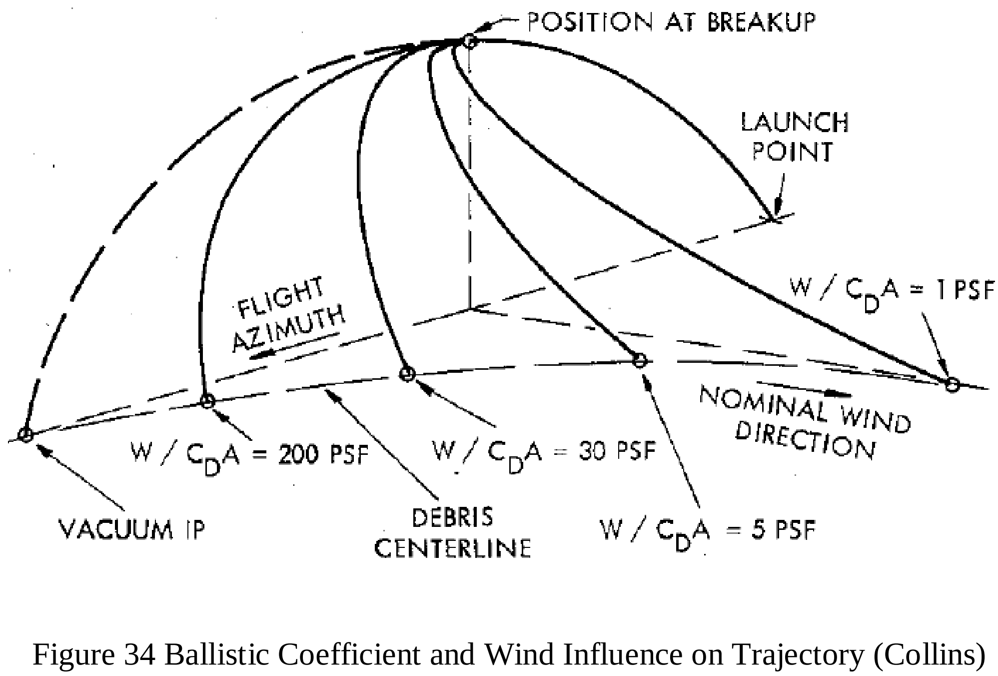

Figure 34 Ballistic Coefficient and Wind Influence on Trajectory (Collins)

It is valid to compute drag coefficient independent of altitude for typical launch vehicle debris for low altitudes. Although it is possible to conduct statistical analysis on event data, ballistic coefficient is generally determined experimentally. Several references include ballistic coefficient intervals of different launch vehicle parts for safety analysis. Ballistic coefficient intervals for launch vehicle parts and propellant debris are given in Table 7 and Table 8 respectively.

Table 7 Recommended Ballistic Coefficient Intervals

β [m^2/kgf] Interval	**Typical Launch Vehicle Parts**
0.114 – 0.068	Skin, doors, interstage structure, skirt, lighter bulkhead parts, straps, fairing sections
0.068 – 0.020	Ducts, heavier bulkhead parts, antennas, medium mass interstage parts, some fairing parts, struts, nozzle extension
0.020 – 0.012	Heavier antennas, interstage structure, telemetry box, small actuators, electronic packages, ACS jets, massive fairing parts
0.012 – 0.007	Small engines, batteries, receivers, helium tanks, nitrogen tanks, propellant lines
0.007 – 0.004	Batteries, actuators, large helium tanks
0.004 – 0.002	Main engines, heat exchangers, gas generators

Table 8 Propellant Debris Ballistic Coefficient Intervals

Burn Stage	**Area [m^2]**		**Mass [kg]**		**Ballistic Coefficient [m^2/kgf]**	
	Min	**Max**	**Min**	**Max**	**Min**	**Max**
0%	0.002	0.105	0.195	19.010	0.004	0.002
25%	0.002	0.104	0.186	14.914	0.005	0.003
50%	0.002	0.102	0.141	10.428	0.008	0.006
75%	0.002	0.092	0.132	5.262	0.015	0.012

Ballistic coefficient interval to be used in safety analysis is selected as 0.002 - 0.114 considering the selected configuration Alternative 1 (See Appendix B).

4.1.2. Imparted Velocity

In case of launch vehicle failure, some amount of energy is released with the potential to fracture the vehicle and disperse the resulting fragments with in-flight explosions or pressure vessel ruptures. The amount of energy of the fracture affects the number, size and shape of the vehicle fragments and the imparted velocity on the fragments. Velocities imparted to debris after the failure are typically modelled in an analytical manner. Models are then calibrated by comparison with empirical data from observed failures.

There are two main sources of imparted velocity. In the first type, pressurized gases can split open the cases of solid rocket motors or liquid propellant tanks. Amount of imparted velocity tends to increase later in the launch mission as a result of increasing chamber volume and decreasing web thickness. In the second type, remaining propellant on the launch vehicle may react chemically to explode. In this case amount of energy may decrease later in flight due to consumed amount of propellant.

Estimation of imparted velocity developed by various launch vehicle developers is given in Table 9. In addition, variation of additional velocity for different ballistic coefficients according to NASA break-up model is given in Figure 35. Amount of additional velocities for solid propellant fragments is given in Table 10.

Table 9 Ranges of Maximum Additional Velocities

Launch Vehicle Section	**Conditions**	**Max. Velocity [m/s]**	
		Lowest	**Highest**
Liquid Core	Large Fragments, All Flight Phases	1	125
	Small Fragments, All Fligt Phases	125	400
Large Multi-segment Solid Rocket Motor Fragments	Early in Flight	25	450
	Late in Flight	30	450
Solid Rocket Upper Stages, Medium Strap-on Boosters	Early in Flight	20	120
	Late in Flight	20	275
Payload Fairing	All Flight Phases	15	35

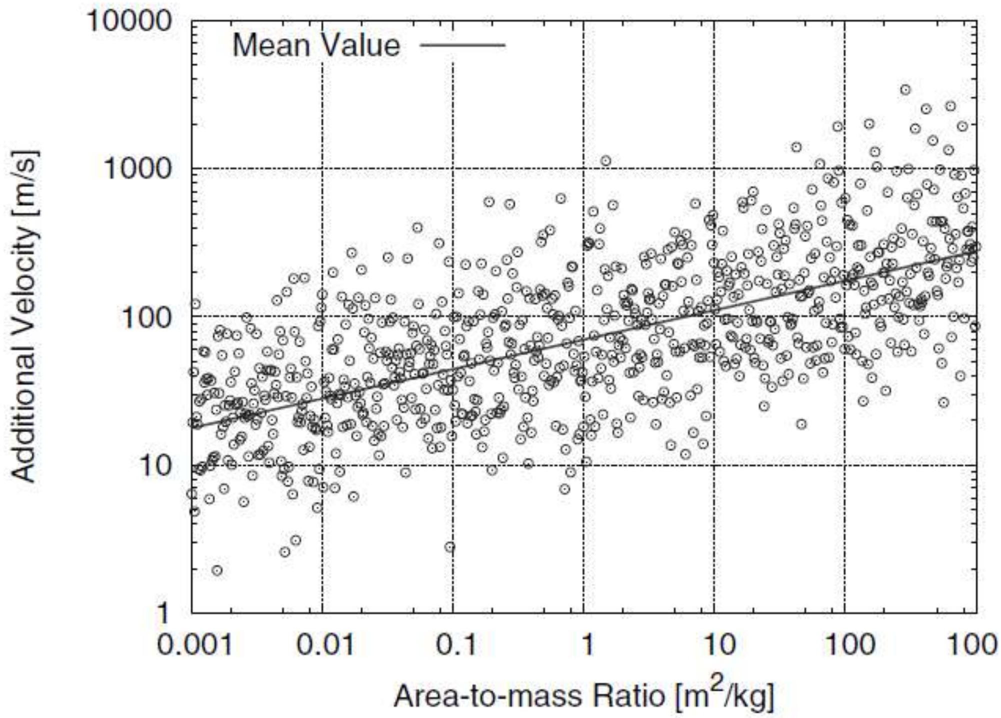

Figure 35 Additional Velocities for Different Ballistic Coefficients (NASA)

Considering typical fragments of turbopumps, nozzles and large tank sections for large fragments of liquid core in addition to the tunnel covers and skin sections for small fragments of liquid core, it can be stated that imparted velocity is related with the ballistic coefficient. NASA break-up model also supports this fact with a relationship. Imparted velocity in trajectory analysis of launch vehicle fragments will be related with ballistic coefficient accordingly.

Table 10 Additional Velocities for Solid Propellant Fragments

Burn Stage	Ballistic Coefficient [m^2/kgf]		Imparted Velocity [m/s]	
	Min	Max	Min	Max
0%	0.004	0.002	122.8	210.6
25%	0.005	0.003	147.2	222.5
50%	0.008	0.006	185.0	240.5
75%	0.015	0.012	258.2	260.9

4.2. SECONDARY EFFECTS

Overpressure effects, thermal effect and toxic pollution should also be considered during the safety analysis. Dispersion of these effects depends on several factors. Determination of safe distances depends on the endurance limits of people and defined in industrial standards. Highest of the distance requirement of secondary effects should be added to the maximum distance that launch vehicle fragments can reach.

4.2.1 Overpressure Effects

Launch vehicle mainly consists of unused propellants on board in the early phases of the flight. Since the relative velocity and the altitude of the vehicle is still low, unburned propellant may reach to earth surface intact which is not expected for the upper stages. In case of an explosion of unburned propellant, overpressure effect occurs. Equations to estimate overpressure due to blast have been developed by Kingery and Bulmash. These widely accepted equations which are based on data from explosive tests using charge weights of TNT from 1 kg to over 400 tons. For the calculation of safe distance for human, TNT equivalent of the explosive propellant and total allowable incident pressure are required. TNT equivalent of HTPB based solid propellants with certain amount of aluminum and AP varies between 0.70 and 0.85. Taking human tolerance to incident as 10 kPa and TNT equivalent of propellants as 0.85, variation of safety distance is given in Figure 36.

Figure 36 Variation Safety Distance with Propellant Weight

4.2.1 Thermal Effect

Fire ball formation occurs due to explosion of large propellant masses. Generated thermal radiation can be harmful for human depending on its intensity. According to fire hazard studies, thermal radiation intensity that is safe for human in canvas clothing depends on the intensity itself and the total duration of the exposure. Thermal radiation intensity for a fire ball can be calculated using the following formula:

$$Q(r) = Ef \times Fq(r) \times \tau(r) \tag{4.1}$$

Ef and $Fq(r)$ in this equation are the average and angular irradiance coefficients while $\tau(r)$ is the atmospheric transmission coefficient. Value of *Ef* is equal to 450 kW/m^2. In case of a fire ball formation at explosion, value of $Fq(r)$ depends on efficient diameter of fire ball *D*, altitude of fireball center from the surface *H* and the distance *r* of human to the projection of fire ball center to earth surface.

$$Fq(r) = \frac{H/D + 0.5}{4\left[\left(H/D + 0.5\right)^2 + \left(r/D\right)^2\right]^{1.5}} \tag{4.2}$$

The efficient diameter of fire ball *D* can be determined by the following relation with *M* being the mass of the propellants included in the explosion.

$$D = 5.33M^{0.327} \tag{4.3}$$

Atmospheric transmission coefficient $\tau(r)$ in Equation (4.1) can be calculated using the following equation.

$$\tau(r) = e^{\left[-0{,}0007\left(\sqrt{r^2+H^2} - \frac{D}{2}\right)\right]} \tag{4.4}$$

Value for altitude of fireball center is recommended as H=D/2. According to fire hazard studies, thermal radiation intensity of 4 kW/m^2 is safe for human with canvas clothing independent of the total exposure time. Safety distances of thermal effects for different propellant masses in case of an explosion and variation of thermal radiation intensity is given in Figure 37.

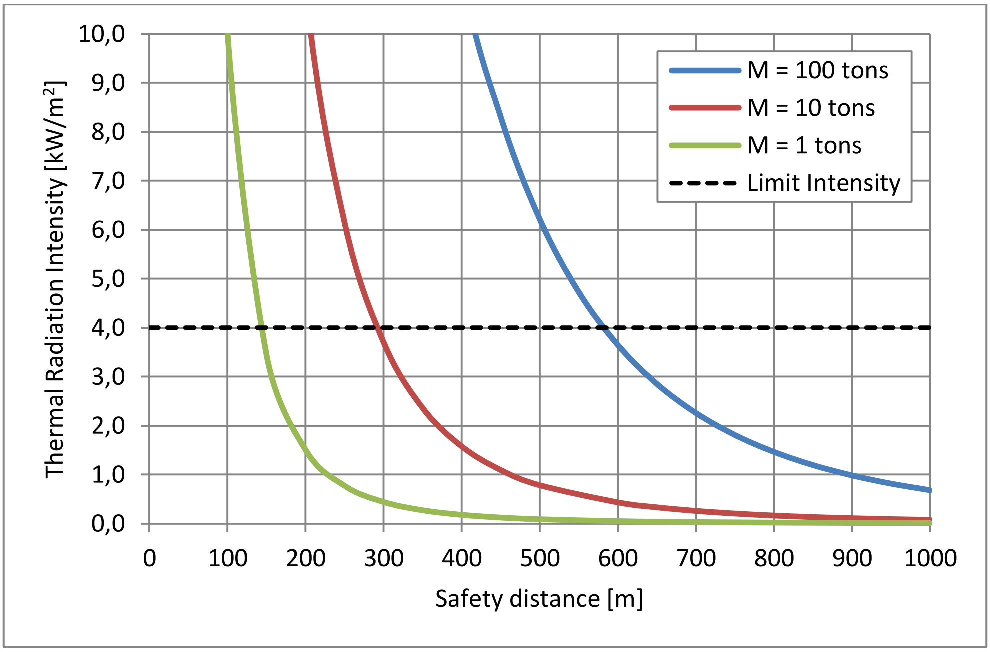

Figure 37 Variation Safety Distance with Propellant Weight (Thermal Effects)

4.2.1 Toxic Pollution

Most of the rocket propellants are composed of toxic materials which can be harmful at even low concentrations. Consequently, toxic pollution is generally the most significant secondary effect in safety analysis. Dispersion of toxic substances depends on atmospheric stability classes and various other conditions such as surface roughness. A classification scheme of the atmospheric stability class for use with Pasquill-Gifford Dispersion Model is given in Table 11. Nighttime in given table refers to the period from one hour before sunset to one hour after sunrise. Strong insolation corresponds to sunny midday in England; slight insolation to similar conditions in midwinter. The neutral category D should also be used, regardless of wind speed, for overcast conditions during day or night and for any sky conditions during the hour before or after sunset or sunrise, respectively. Proper wording for atmospheric stability categories is as follows;

A	Extremely Unstable Conditions
B	Moderately Unstable Conditions
C	Slightly Unstable Conditions
D	Neutral Conditions
E	Slightly Stable Conditions
F	Moderately Stable Conditions

Table 11 Atmospheric Stability Classes

	Daytime Insolation			Nighttime Conditions	
Surface Wind Speed [m/s]	Strong	Moderate	Slight	Thin Overcast or >4/8 Low Cloud	≤3/8 Cloudiness
<2	A	A-B	B	F	F
2-3	A-B	B	C	E	F
3-4	B	B-C	C	D	E
4-6	C	C-D	D	D	D
>6	C	D	D	D	D

Surface wind speed strongly depends on geographical location. Seasonal variations of mean wind speed should also be considered for safety analysis. Annual mean wind speed for Turkey varies between 0.5 and 6.1 m/s.

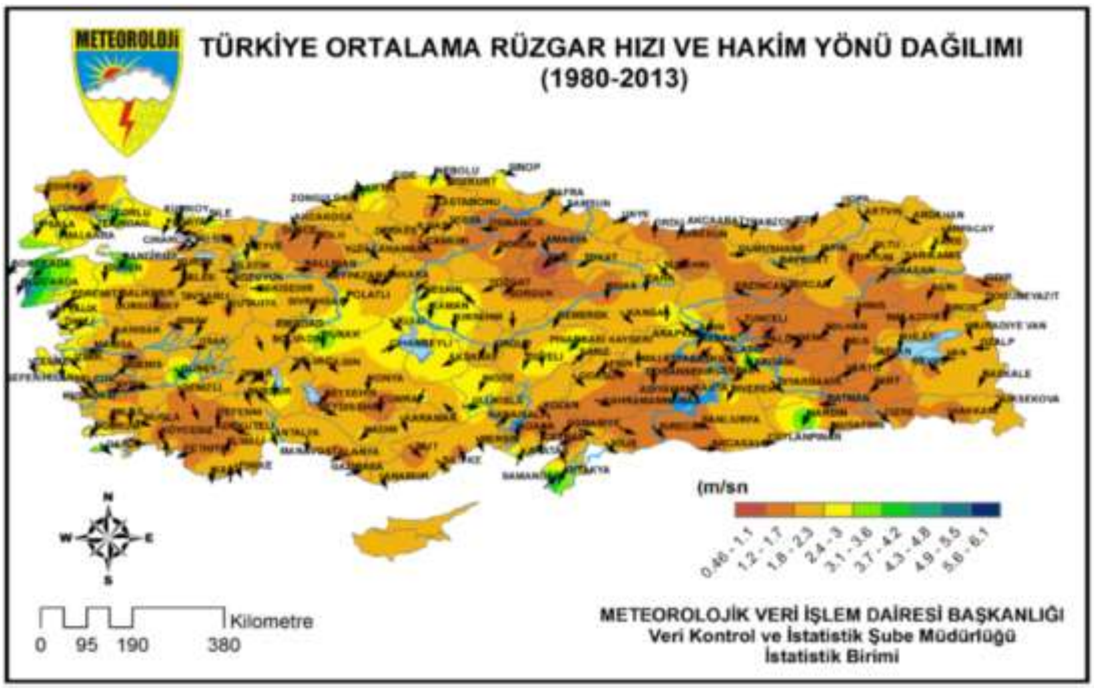

Figure 38 Mean Wind Speed and Prevailing Wind Directions for Turkey

Daytime insolation depends on the cloudiness of the space launch center. Although valuable statistics for cloudiness is available from General Directorate of State Meteorology Affairs of Turkey, Moderate Resolution Imaging Spectroradiometer data from Terra and Aqua satellites is also a key source of cloudiness information. Cloudiness information around selected location is given in the following figures.

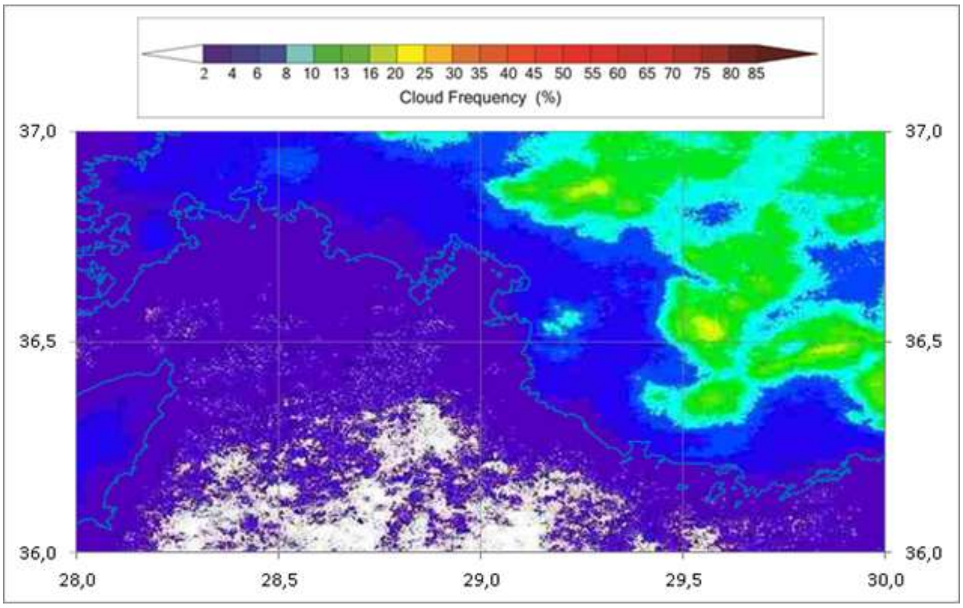

Figure 39 Cloud Frequency between May and October

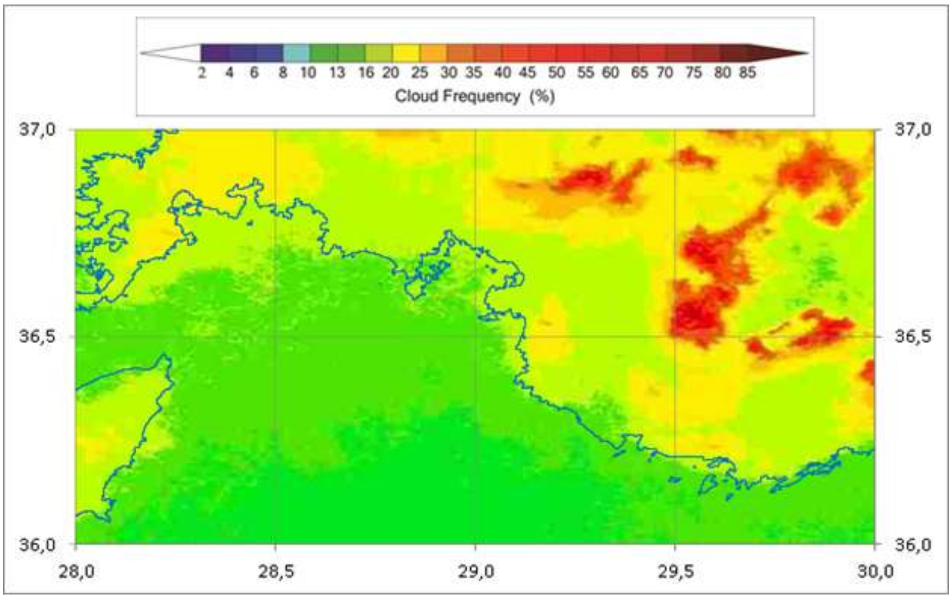

Figure 40 Cloud Frequency between November and April

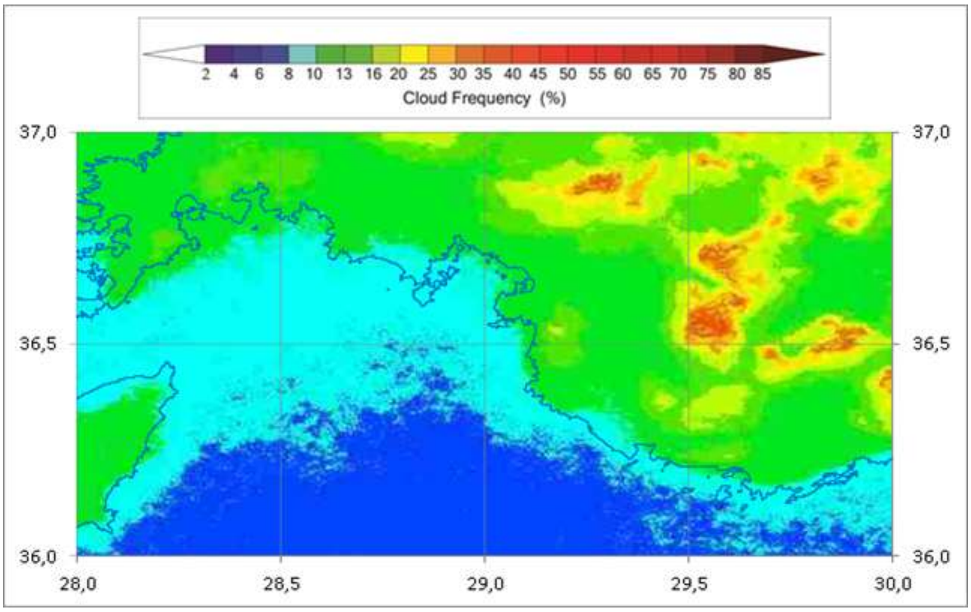

Figure 41 Cloud Frequency (Annual)

There are various dispersion models for toxic substances. Wind speed, atmospheric stability, ground conditions and height of the release above the surface are the main parameters. Atmospheric stability is related with the vertical mixing of the air. Ground conditions are related with the mixing at the surface and the variation of wind profile with increasing height. Increasing release height decreases ground concentrations. The puff model can be used for the analysis of instantaneous release of toxic materials in case of an explosion at ground level. For a puff with instantaneous point source, where wind is in direction *x* and constant, average concentration can be obtained by the following equation.

$$C(x,y,z,t) = \frac{Q_m^*}{\sqrt{2}\pi^{3/2}\sigma_x\sigma_y\sigma_z}\exp\left\{-\frac{1}{2}\left[\left(\frac{x-ut}{\sigma_x}\right)^2 + \frac{y^2}{\sigma_y^2} + \frac{z^2}{\sigma_z^2}\right]\right\} \qquad (4.5)$$

In the coordinate system used for dispersion model, *x* is the downwind direction, *y* is off-wind direction and *z* is the vertical direction Q_m^* is the mass of the released toxic material. At ground level z = 0, concentration can be computed as follows;

$$C(x,y,0,t)=\frac{Q_m^*}{\sqrt{2}\pi^{3/2}\sigma_x\sigma_y\sigma_z}\exp\left\{-\frac{1}{2}\left[\left(\frac{x-ut}{\sigma_x}\right)^2+\frac{y^2}{\sigma_y^2}\right]\right\} \tag{4.6}$$

Total dose received by an individual staying at fixed coordinates is equal to time integral of the concentration. Ground level concentration can be found as follows;

$$D(x,y,z)=\int_0^\infty C(x,y,z,t)dt \tag{4.7}$$

Total dose for an individual at the ground level is found by integrating Equation (5.6).

$$D(x,y,0)=\frac{Q_m^*}{\pi\sigma_y\sigma_z u}\exp\left(-\frac{1}{2}\frac{y^2}{\sigma_y^2}\right) \tag{4.8}$$

Pasquill-Gilford dispersion coefficients, σ_x, σ_y and σ_z, for the puff model depend on the surface roughness values. Reference surface roughness values of different terrains are given in Table 12.

Table 12 Surface Roughness Values for Various Types of Terrains

Terrain Description	**Surface Roughness z_0 [cm]**
Snow, lawn with 1 cm height	0.1
Sea, coastal area exposed to the open sea	0.3
Mowed and low grass up to 15 cm	0.6 - 2
High grass up to 60 cm	4 - 9
Non-uniform surface with grass, bushes etc.	10 - 20
Park, forest up to 10 m high	20 - 100
City with buildings of various heights	100

First step for the determination of dispersion coefficients is to check the value of σ_z^{max} according to atmospheric stability class.

Table 13 Value of σ_z^{max} for Atmospheric Stability Classes

Stability Category	**σ_z^{max} [m]**	**Stability Category**	**σ_z^{max} [m]**	**Stability Category**	**σ_z^{max} [m]**
A	1600	C	640	E	220
B	920	D	400	F	100

Value of $\sigma_z(x)$ depends on product of two functions $f(z_0, x)$ and $g(x)$ compared to σ_z^{max}.

$$\sigma_z(x) = \begin{cases} f(z_0,x)g(x) & \text{if} \quad f(z_0,x)g(x) \le \sigma_z^{\max} \\ \sigma_z^{\max} & \text{if} \quad f(z_0,x)g(x) > \sigma_z^{\max} \end{cases} \tag{4.9}$$

Functions $f(z_0, x)$ and $g(x)$ obtained using different coefficients of atmospheric stability classes.

$$g(x) = a_1 x^{b_1} / (1 + a_2 x^{b_2}) \tag{4.10}$$

$$f(z_0,x) = \begin{cases} \ln\left[c_1 x^{d_1} \times (1 + c_2 x^{d_2})\right] & \text{if} \quad z_0 > 10cm \\ \ln\left[c_1 x^{d_1} / (1 + c_2 x^{d_2})\right] & \text{if} \quad z_0 \le 10cm \end{cases} \tag{4.11}$$

Coefficients of functions $g(x)$ and $f(z_0, x)$ are given in Table 14 and Table 15, respectively.

Table 14 Coefficients for $g(x)$

Stability Category	**a_1**	**a_2**	**b_1**	**b_2**
A	0.1120	5.38E-04	1.060	0.815
B	0.1300	6.52E-04	0.950	0.750
C	0.1120	9.05E-04	0.920	0.718
D	0.0980	1.35E-03	0.889	0.688
E	0.0609	1.96E-03	0.895	0.684
F	0.0638	1.36E-03	0.783	0.672

Table 15 Coefficients for $f(z_0, x)$

Surface Roughness z_0 [cm]	c_1	c_2	d_1	d_2
1	1.56	6.25E-04	0.04800	0.450
4	2.02	7.76E-04	0.02690	0.370
10	2.73	0	0	0
40	5.16	5.38E-02	-0.09800	0.225
100	7.37	2.33E-04	-0.00957	0.600
400	11.70	2.18E-05	-0.12800	0.780

Taking the values of lateral dispersion coefficients σ_x and σ_y equal is a good estimate for the puff model and it can be calculated using the following formula.

$$\sigma_x(x) = \sigma_y(x) = \mathrm{c}_3\, x / \sqrt{1 + 0.0001x} \tag{4.12}$$

Value of the c_3 depends on atmospheric stability category as follows;

Table 16 Coefficients for Calculation of σ_x

Stability Category	c_3	**Stability Category**	c_3	**Stability Category**	c_3
A	0.22	C	0.11	E	0.06
B	0.16	D	0.08	F	0.04

4.3. TRAJECTORY SOLUTION

Aerodynamic forces, which are the primary concern in case of atmospheric flight, arise due to the motion relative to the atmosphere. Due to the fact that atmosphere rotates with Earth, an Earth-fixed rotating reference frame is required for the expression of the equations of motion. Such an Earth-fixed reference frame (*SXYZ*) is shown in Figure 42.

Spherical coordinates v, ϕ and A, can be used to express the velocity **v** relative to the rotating frame. In this case v, ϕ and A represents the magnitude of the velocity vector, flight path angle and velocity azimuth respectively which are measured in the local horizon frame (*oxyz*). Local horizon plane is also given in Figure 42. Derivation of the equations of motions for spherical coordinates is well known and can be found in books on flight dynamics.

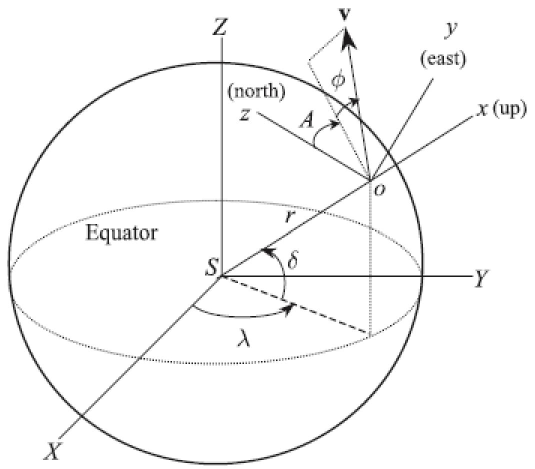

Figure 42 Earth-fixed Reference Frame for Atmospheric Flight (Tewari)

Kinematic equations of motion relative to rotating Earth are as follows;

$$\dot{r} = v \sin \phi \tag{4.13}$$

$$\dot{\delta} = \frac{v}{r} \cos \phi \cos A \tag{4.14}$$

$$\dot{\lambda} = \frac{v\cos\phi\sin A}{r\cos\delta} \tag{4.15}$$

In order to complete the solution for the trajectory, after the determination of position vector (r, δ, λ) from the kinematic equations, determination of relative velocity vector (v, ϕ, A) is required from the solution of the dynamic equations. Dynamic equations of motion are as follows;

$$\begin{aligned} m\dot{v} = {} & f_T\cos\varepsilon\cos\mu - D - mg_c\sin\phi + mg_\delta\cos\phi\cos A \\ & -m\omega^2 r\cos\delta(\cos\phi\cos A\sin\delta - \sin\phi\cos\delta) \end{aligned} \tag{4.16}$$

$$\begin{aligned} mv\cos\phi\dot{A} = {} & m\frac{v^2}{r}\cos^2\phi\sin A\tan\delta + f_T\sin\mu + f_Y - mg_\delta\sin A \\ & +m\omega^2 r\sin A\sin\delta\cos\delta \\ & -2m\omega v(\sin\phi\cos A\cos\delta - \cos\phi\sin\delta) \end{aligned} \tag{4.17}$$

$$\begin{aligned} mv\dot{\phi} = {} & m\frac{v^2}{r}\cos\phi + f_T\sin\varepsilon\cos\mu + L - mg_c\cos\phi - mg_\delta\sin\phi\cos A \\ & +m\omega^2 r\cos\delta(\sin\phi\cos A\sin\delta + \cos\phi\cos\delta) \\ & +2m\omega v\sin A\cos\delta \end{aligned} \tag{4.18}$$

For a flight dynamic model gravity model, atmospheric models, aerodynamic and propulsion models are also required. In case of ballistic trajectories of launch vehicle debris, propulsion model is out of scope. It is possible to neglect the terms with thrust force f_T and side force f_Y and lift L in dynamic equations of motion.

Although it is necessary to take into account the radial and latitudinal variations of gravity with a nonspherical planet model for long flight durations, Newton's law of universal gravitation and spherical Earth assumptions is sufficient for debris impact analysis for the early phases of the flight. In this case, latitudinal component of the gravity g_δ equals to zero.

Atmosphere is never at rest relative to the Earth. Changes in the wind direction and velocity in different altitudes affects the ballistic trajectory of the debris by changing the aerodynamic force and its direction. For velocity vector of debris relative to the earth defined as $\vec{v}$, velocity vector of wind which is assumed to be horizontal and resolved in the local horizon frame can be defined as $\vec{v}_w$.

$$\vec{v}_w = v_w(\sin A_w \vec{\mathrm{j}} + \cos A_w \vec{\mathrm{k}}) \tag{4.19}$$

Debris velocity relative to the atmosphere can be obtained as follows;

$$\begin{aligned}\vec{v}' = \vec{v} - \vec{v}_w = v\sin\phi\vec{\mathrm{i}} + (v\cos\phi\sin A - v_w\sin A_w)\vec{\mathrm{j}} \\ +(v\cos\phi\cos A - v_w\cos A_w)\vec{\mathrm{k}}\end{aligned} \tag{4.20}$$

Relative velocity in the wind axis can be expressed as follows;

$$\vec{v}' = v'\vec{\mathrm{i}}'_{\mathrm{v}} \tag{4.21}$$

Magnitude of the effective velocity for the calculation of the aerodynamic forces and the effective sideslip angle caused by the wind is as follows;

$$v' = \sqrt{v^2 + v_w^2 - 2vv_w\cos\phi\cos(A - A_w)} \tag{4.22}$$

$$\begin{aligned}\beta = \cos^{-1}((v\sin^2\phi + (v\cos\phi\sin A - v_w\sin A_w)\cos\phi\sin A \\ +(v\cos\phi\cos A - v_w\cos A_w)\cos\phi\cos A)/v')\end{aligned} \tag{4.23}$$

In this case, wind azimuth A_w is defined as the angle in which the wind is blowing and measured from the north. Wind from southeast corresponds to $A_w = 135^{\circ}$.

Integration of these ordinary differential equations which are coupled and nonlinear requires a numerical solution like iterative Runge-Kutta method. In addition simultaneous operation of functions for the generation of synthetic wind profile, determination of atmospheric density and addition of imparted velocities are also required for the solution. Computer codes used for the trajectory simulations were developed using MATLAB® (Version R2013b, 8.2.0.701) and given in APPENDIX A.

4.4. ATMOSPHERIC CONDITIONS

Distance requirements arising from the secondary effects, depending on atmospheric conditions, should be added to the drop point areas obtained with trajectory analysis. Since seasonal variations of the atmospheric conditions are the primary concern for the presented test case of the selected launch center location, it is required to determine the reference conditions.

Variation of safety distance depending on the propellant weight and due to overpressure effect is provided in Figure 36. In this case, only required input is the propellant consumption rate and total propellant of the launch vehicle. Taking into account that selected baseline configuration Alternative 1 contains more than 140 tons of solid propellant and more than 8 tons of liquid propellant, variation of safety distance can be calculated for human tolerance of 10 kPa.

Table 17 Safety Distance Due to Overpressure Effect

Time [s]	Used Propellant [kg]	Total Propellant [kg]	TNT Equivalent [kg]	Safety Distance [m]
0	0	152570	122515	632
10	13511	139059	111030	612
20	28540	124031	98256	587
30	40576	111994	88025	566
40	49812	102758	80174	549

Variation of total propellant weight of the launch vehicle can be used for the determination of the safety distance by considering the limit of thermal radiation intensity 4 kW/m^2 for human in canvas clothing. Safety distance due to heat flux in case of an explosion is given in the following table.

Table 18 Safety Distance Due to Thermal Effect

Time [s]	Used Propellant [kg]	Total Propellant [kg]	Safety Distance [m]
0	0	152570	657
10	13511	139059	640
20	28540	124031	619
30	40576	111994	601
40	49812	102758	586

It is required to determine seasonal atmospheric stability category for winter and summer conditions by considering the surface wind speed and average insolation. Average wind speeds for the selected launch center location are 0.8 m/s and 3.6 m/s for winter and summer conditions respectively. These values are also confirmative with the annual mean wind speed for Turkey that varies between 0.5 and 6.1 m/s. Average cloudiness can be determined using Figure 39 and Figure 40. For winter average cloudiness changes between 10-13% while for summer this range is in between 6-8%. Both of these values correspond to slight daytime insolation. As a result, atmospheric stability class for winter and summer is C and B respectively. In addition surface roughness for the selected location can be taken as 0.3 cm for both seasons according to the information given in Table 12 and due to the fact that it is located at open sea.

Table 19. Required Inputs for Toxic Pollution

Season	Average Surface Wind Speed [m/s]	Daytime Insolation [%]	Atmospheric Stability Class
Winter	0.8	10 - 13	C
Summer	3.6	6 - 8	B

Main toxic components of solid propellant combustion are Hydrogen chloride (HCl) and Carbon monoxide (CO). Safe to estimate percentages are 20% HCl and 35% CO of the unused solid propellant. Since liquid propellants contained in upper stage are hypergolic and assumed to be completely combusted in case of an accident, they are not included in toxic hazard calculations.

Table 20 Amount of Toxic Substances

Time [s]	Used Propellant [kg]	Solid Propellant [kg]	HCl (20%) [kg]	CO (35%) [kg]
0	0	143700	28740	10059
10	13511	130189	26038	9113
20	28540	115161	23032	8061
30	40576	103124	20625	7219
40	49812	93888	18778	6572

In order to determine safety distances of toxic substances, it is required to know the limit values that are harmful to humans. For this purpose, Immediately Dangerous to Life or Health (IDLH) limits at guide to chemical hazards which is published by The National Institute for Occupational Safety and Health (NIOSH) can be used. According to NIOSH, IDLH limits for HCl and CO are as follows;

- HCl: 50 ppm (75 mg/m^3)
- CO: 1200 ppm (1380 mg/m^3)

Total exposure time of 15 minutes is considered for the determination of the received doses and safety distances. Due to the fact that amount of released CO is less than HCl and IDLH limit of CO is higher, safety distances of HCL is much higher than CO. Variation of the total exposed doses for both substances with respect to the distance from the incident point is compared for summer conditions in the following figure.

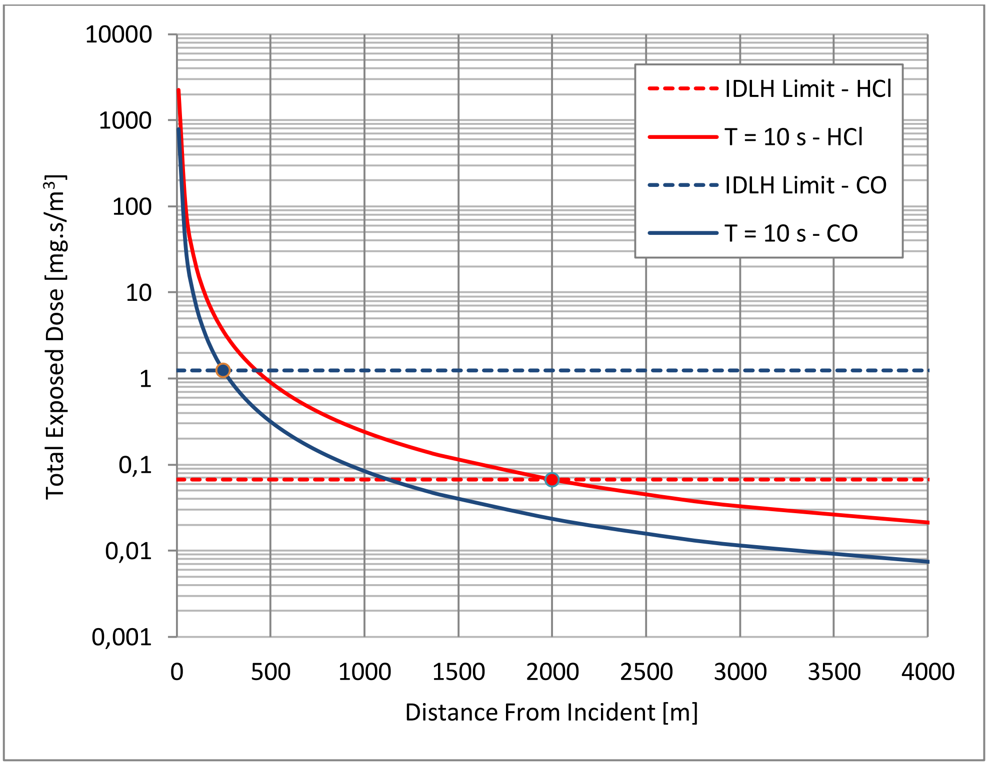

Figure 43 Comparison of Safety Distances: HCl & CO (Summer Conditions)

It is also important to note that safety distance depends on seasonal variations. Decreasing surface winds with winter causes an increase in the total exposed doses. Safety distances of winter conditions are about four times higher in winter conditions. Comparison of safety distances for HCl vapor at summer and winter conditions is given in the following figure.

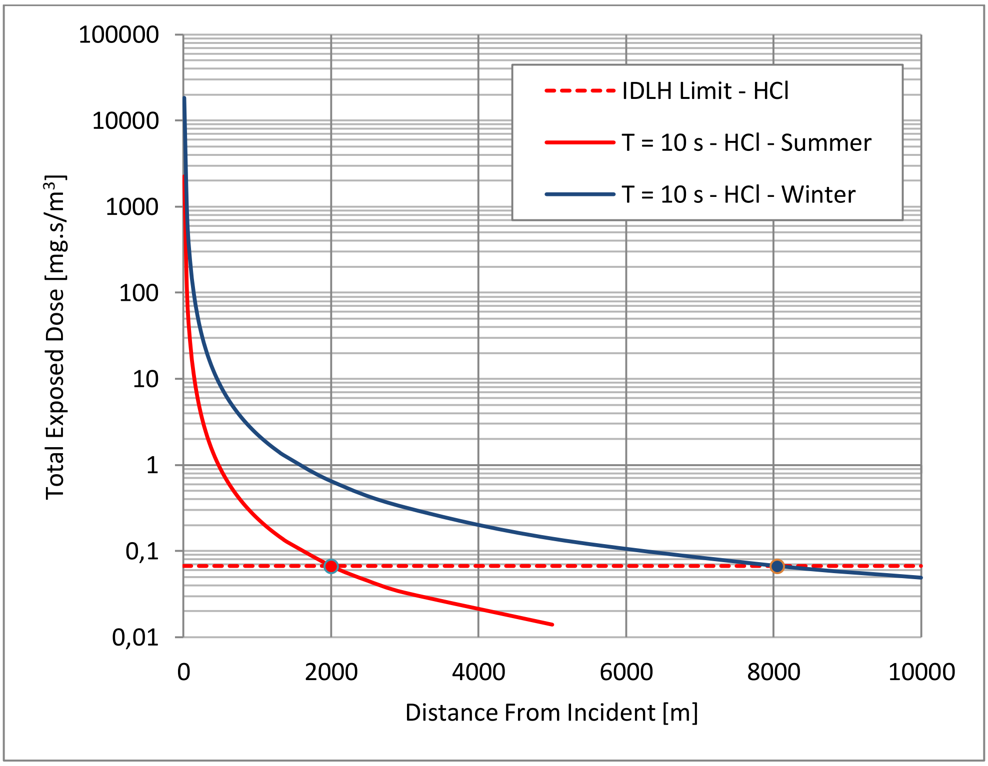

Figure 44 Comparison of Safety Distances: Summer & Winter (HCl)

Safety distances due to toxic substances are given in the following table.

Table 21 Safety Distances Due to Toxic Pollution

Time [s]	Safety Distances [m]			
	Summer		Winter	
	HCl	CO	HCl	CO
0	2112	261	8615	829
10	1999	248	8055	786
20	1867	233	7414	736
30	1757	221	6890	694
40	1668	211	6477	661

Considering the reference conditions of different seasons, safety distance requirements due to secondary effects are determined. According to the results, most important effect is the toxic pollution due to HCl in winter conditions. Obtained safety distances should be added to the results of drop point analysis.

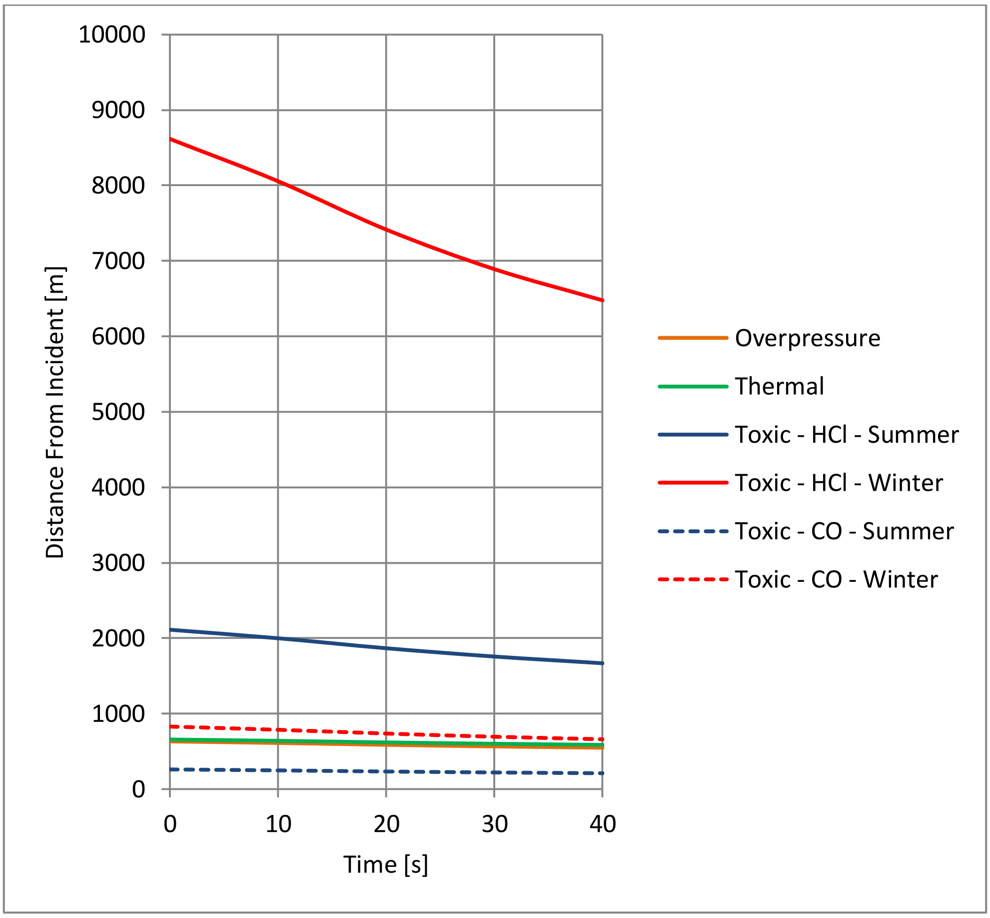

Figure 45 Comparison of Safety Distances of Different Effects

4.5. SAFETY RESULTS

Launch vehicles generally start their flight vertically. Vertical flight phase is followed by a pitching maneuver which decreases the flight path angle. For the initial phases of the flight, altitude and velocity increases and azimuth is constant. In the safety analysis Azimuth angle is equal to 191.6^o for all emergency instants. Initial conditions of the trajectory solution for the safety analysis are as follows:

Table 22 Initial Conditions at Emergency Instants

Emergency Instant [s]	Flight Path Angle [^{o}]	Relative Velocity [m/s]	Altitude [m]	Latitude [^{o}]	Longitude [^{o}]
10	89.8	101.7	445	36.2500	29.2500
20	76.5	245.0	2178	36.2484	29.2496
30	68.3	359.8	5064	36.2405	29.2475
40	62.3	437.9	8704	36.2266	29.2438

Variation of flight path angle, relative velocity and altitude in addition to the ground track of the launch vehicle is given in the following figures.

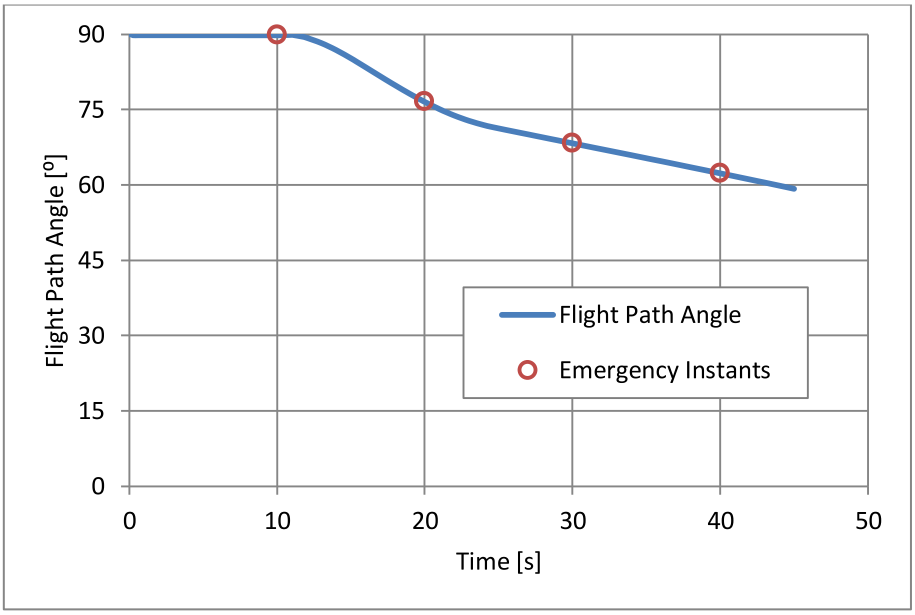

Figure 46 Variation of Flight Path Angle on Nominal Trajectory

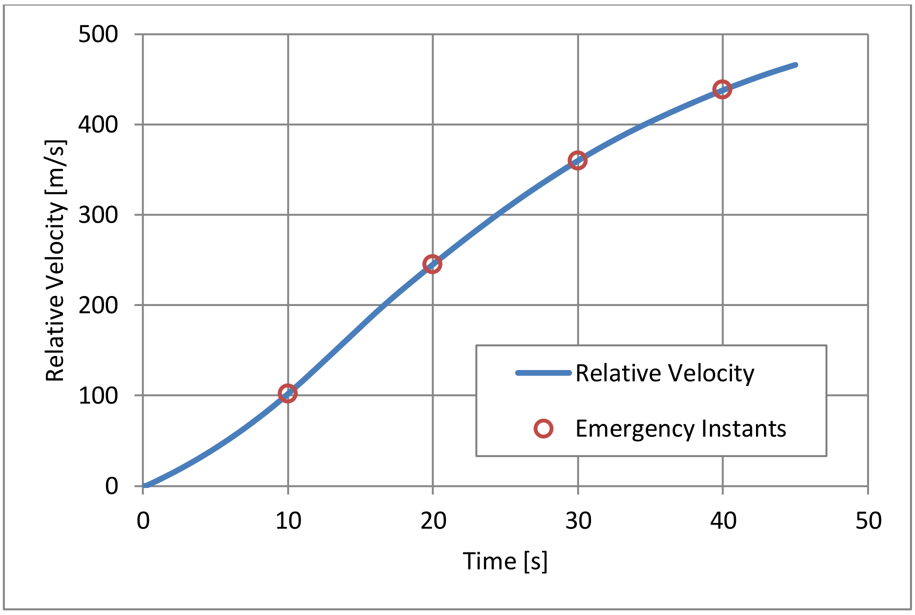

Figure 47 Variation of Relative Velocity on Nominal Trajectory

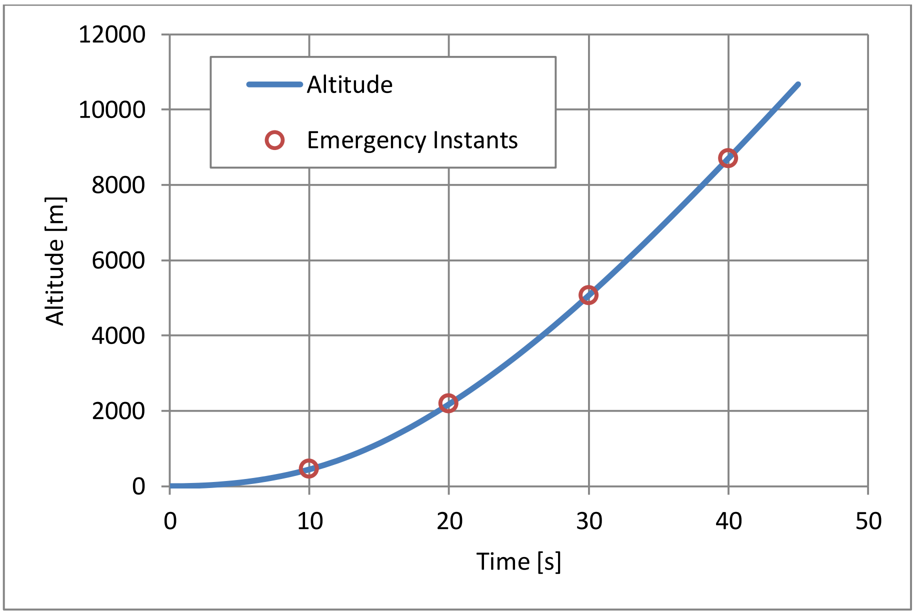

Figure 48 Variation of Altitude on Nominal Trajectory

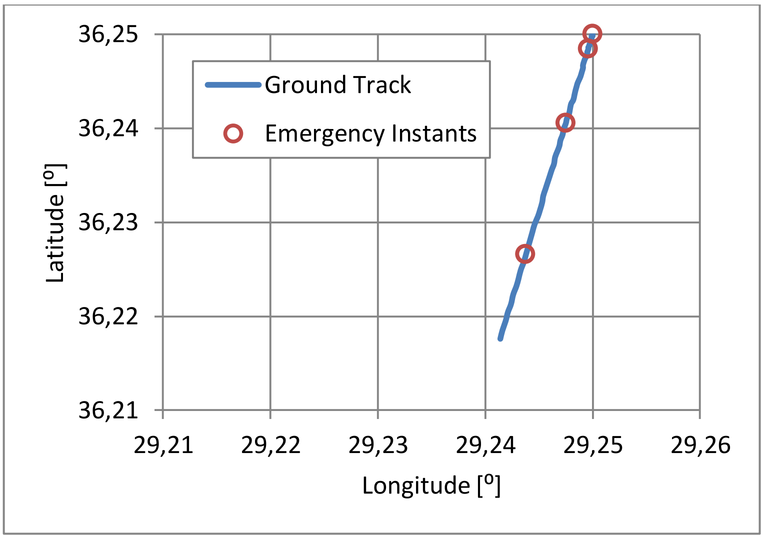

Figure 49 Ground Track of the Launch Vehicle

Results of drop point analysis is compared for each emergency instant. Drop points for summer conditions are given in blue while red is used for the winter conditions. Same scale for the latitude and longitude axes is used for ease of comparison. Obtained results are presented in Figure 50 to Figure 53.

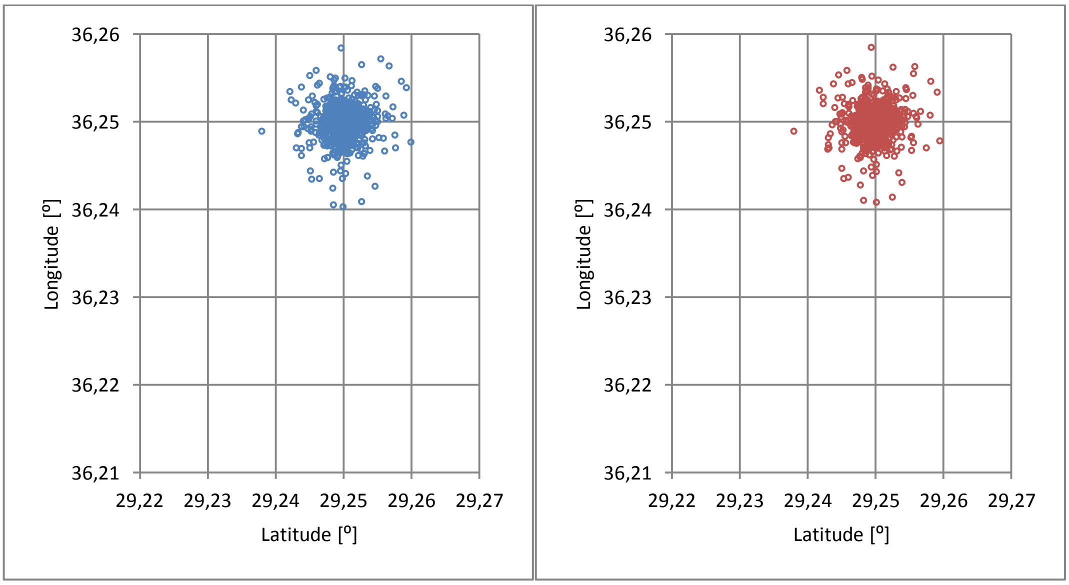

Figure 50 Comparison of Drop Points for Summer & Winter (t = 10 s)

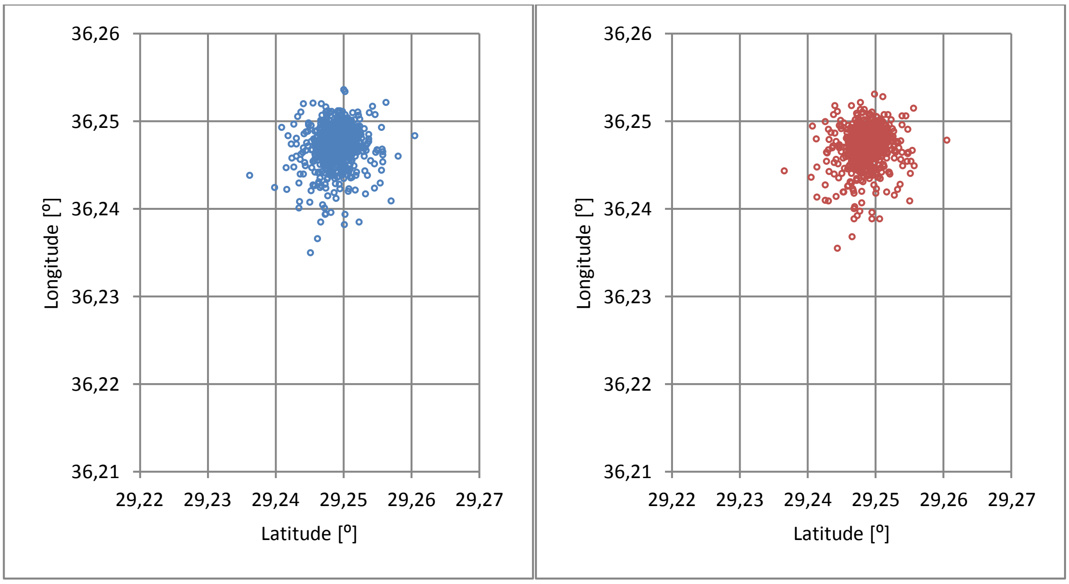

Figure 51 Comparison of Drop Points for Summer & Winter (t = 20 s)

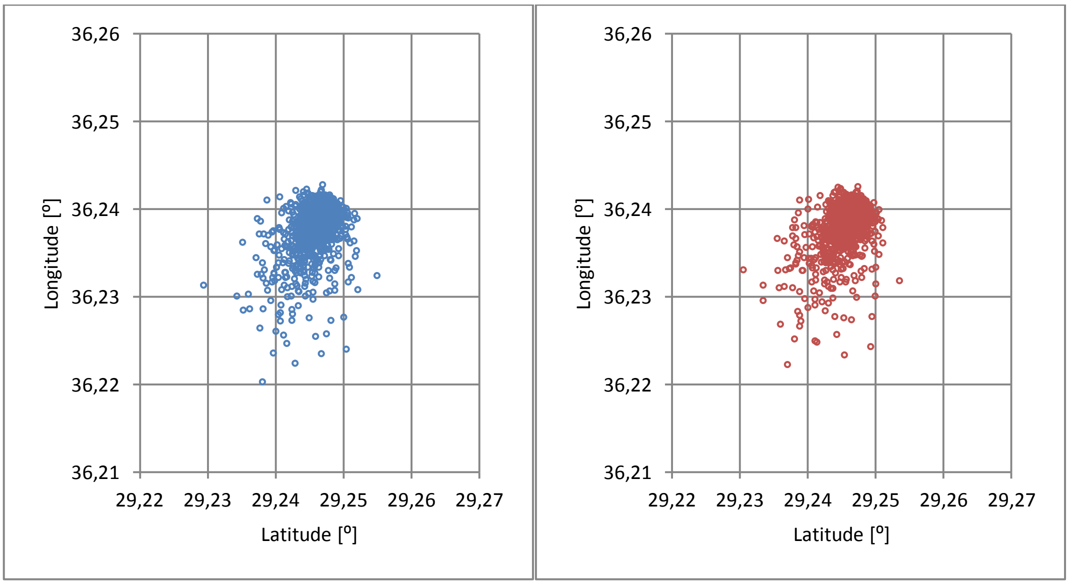

Figure 52 Comparison of Drop Points for Summer & Winter (t = 30 s)

Although drop point comparisons up to the emergency instance of t = 30 s does not contain a significant difference in the covered total area, drop points' tendency to east can be seen in summer conditions of the emergency instance of t = 40 s. This is due to the longer duration of ballistic flight under the influence of high altitude winds which are higher in scalar magnitude.

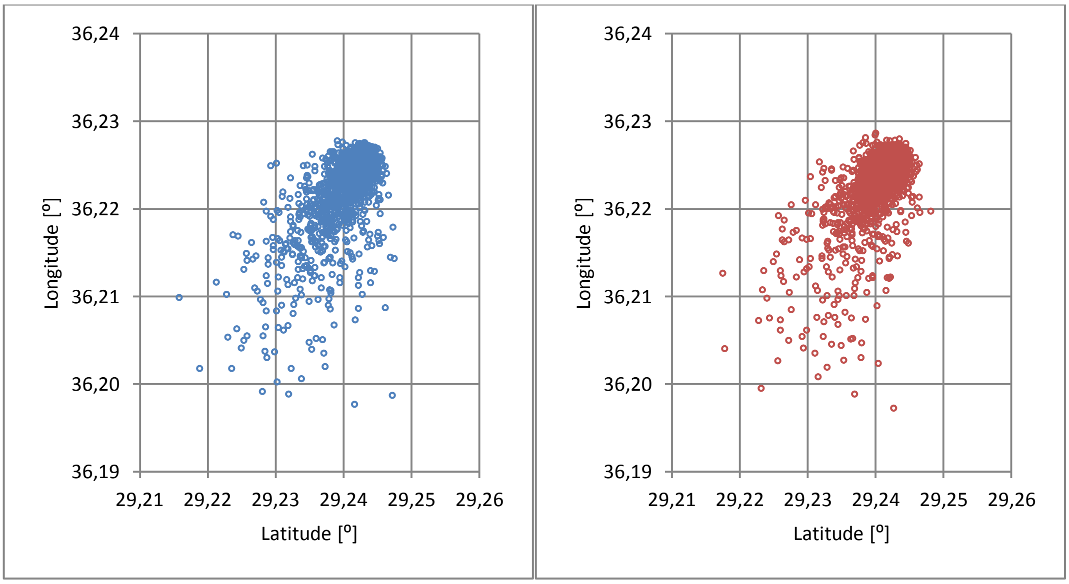

Figure 53 Comparison of Drop Points for Summer & Winter (t = 40 s)

Comparison of calm (no wind) condition of emergency instant t = 40 s reveals the effect of wind on the drop points' pattern. For the calm condition, center line of the drop points' area is along the azimuth of the launch vehicle and dispersion is symmetric with respect to the center line.

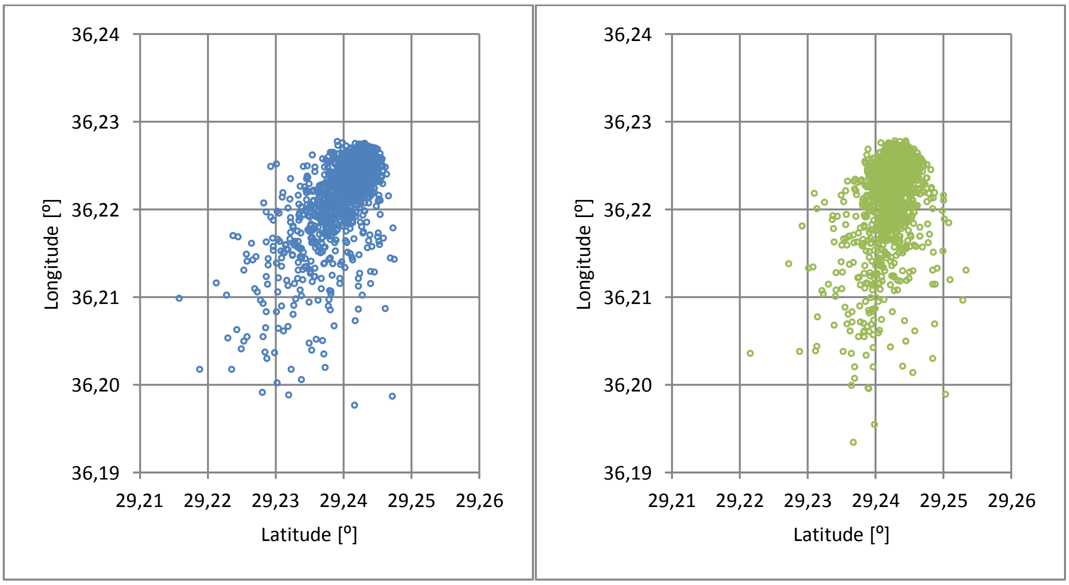

Figure 54 Comparison of Drop Points for Summer & Calm (t = 40 s)

Combined comparison of summer and winter conditions in addition to the combined comparison of summer and calm conditions is given in the following figures. Due high altitude winds of the selected launch center location, dispersed trajectories of launch vehicle fragments have a tendency of sliding to west.

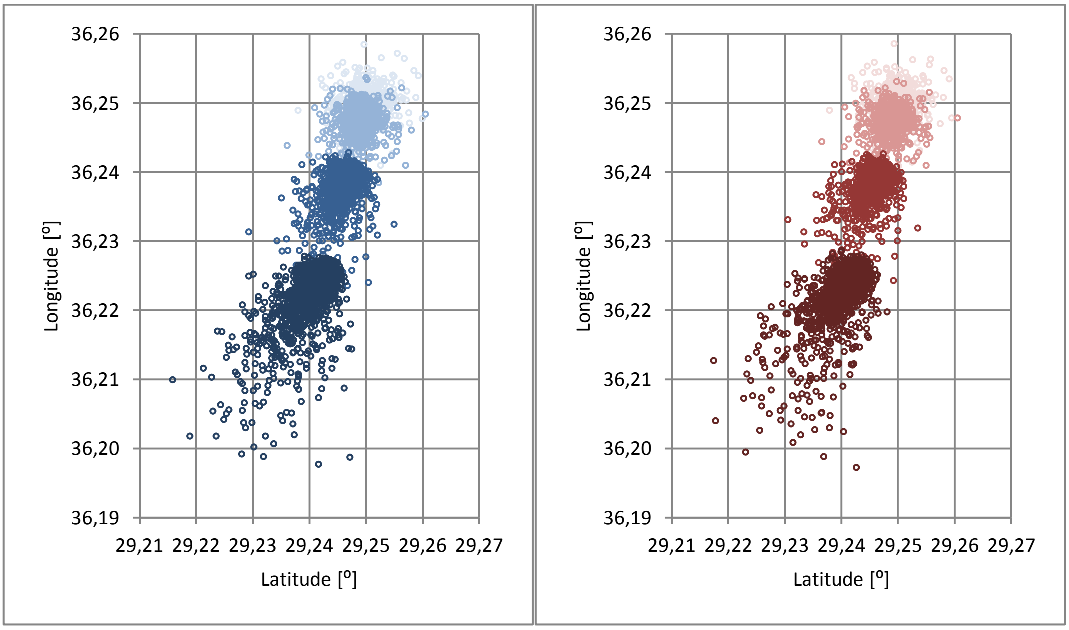

Figure 55 Comparison of Drop Points for Summer & Winter

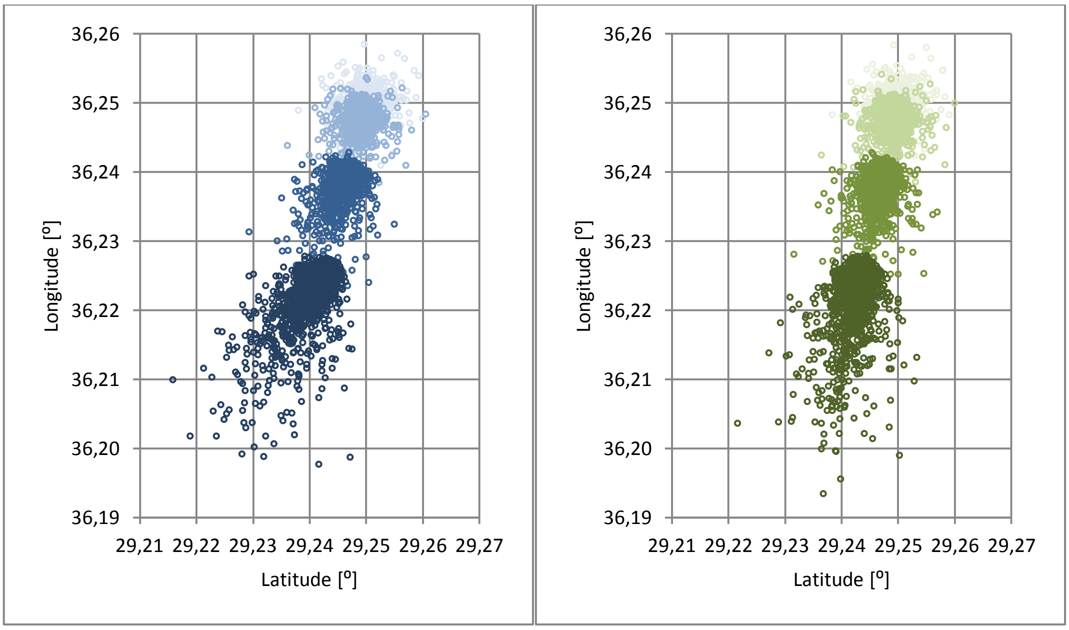

Figure 56 Combined Comparison of Drop Points for Summer & Calm

In order to understand the lack of difference between drop points' area of summer and winter conditions it is required to compare the wind speeds at high altitudes. Although surface mean wind speeds of different seasons are different, difference between the mean values decrease with increasing altitude. This reduces the total displacement due to wind. In addition, parts that are far from the incident point are the parts with low ballistic coefficient. They are affected from the wind less when compared to the parts with high ballistic coefficients.

Table 23 Wind Information for Summer Conditions

Alt. [km]	U [m/s]	SDU	V [m/s]	SDV	Mean [m/s]
0	2.73	2.00	-2.33	2.23	3.58
1	1.47	3.08	-3.73	3.41	4.01
2	0.97	3.50	-3.18	3.74	3.33
3	1.64	4.03	-2.27	4.30	2.80
4	3.73	4.71	-1.84	5.07	4.16
5	6.04	5.41	-1.48	5.86	6.22
6	8.44	6.12	-0.97	6.70	8.49
7	11.29	6.87	0.29	7.77	11.29
8	14.45	7.56	2.00	8.83	14.59
9	17.98	8.18	4.22	9.87	18.47
10	21.17	8.70	6.52	10.78	22.15

Table 24 Wind Information for Winter Conditions

Alt. [km]	U [m/s]	SDU	V [m/s]	SDV	Mean [m/s]
0	0.83	2.98	-0.08	3.95	0.83
1	1.92	4.44	0.33	5.83	1.94
2	3.57	5.39	0.31	6.64	3.59
3	5.67	6.44	-0.11	7.59	5.67
4	7.72	7.88	-0.40	8.76	7.73
5	9.77	9.35	-0.68	9.94	9.79
6	11.85	10.95	-0.90	11.11	11.89
7	13.99	12.75	-1.02	12.26	14.02
8	16.67	14.41	-0.99	13.20	16.70
9	19.49	16.04	-0.91	14.08	19.51
10	22.58	16.76	-0.71	14.26	22.59

Variation of mean wind speed and density for summer and winter conditions is compared in the following figure.

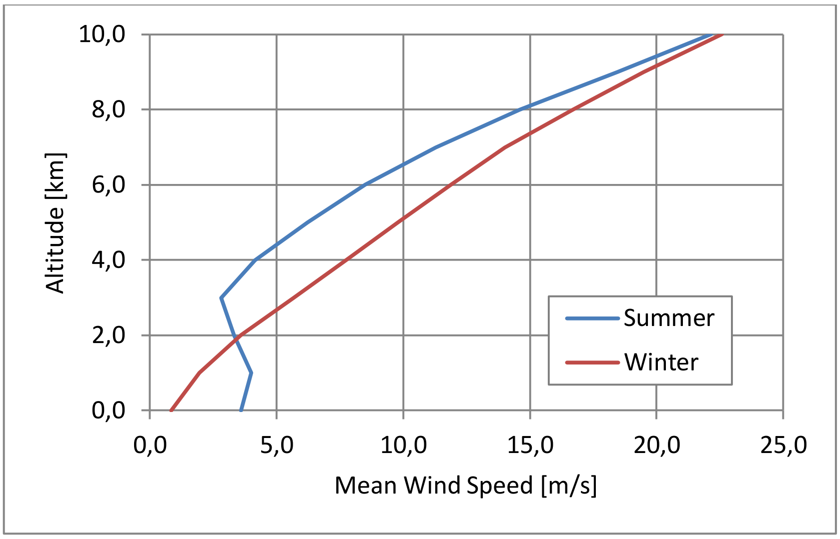

Figure 57 Variation of Mean Wind Speed Comparison

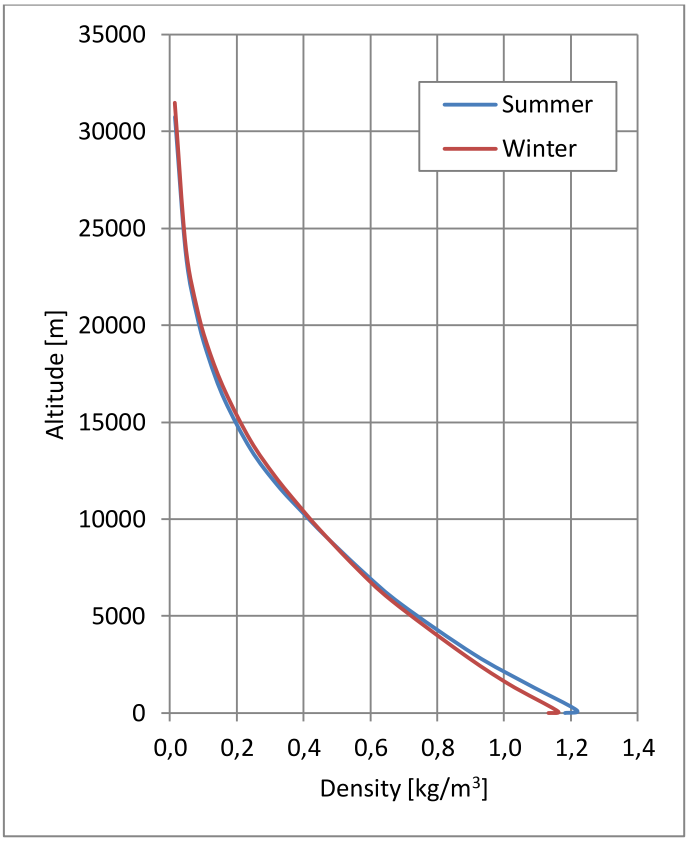

Figure 58 Variation of Density Comparison

In order to complete the safety analysis, it is required to add safety distances arising from the secondary effects to the safety distances of drop point analysis. Toxic effect of HCl is the most important factor for both seasons (Figure 45) and it is sufficient to add the HCl safety distances.

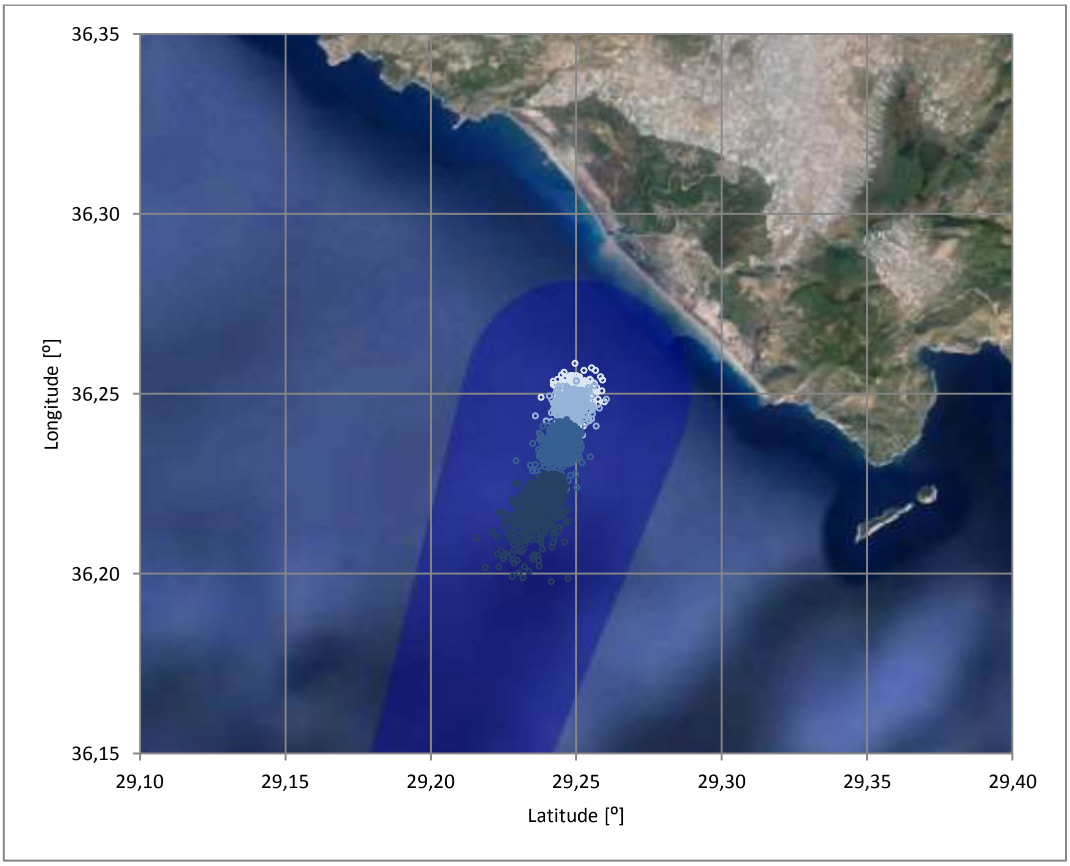

Figure 59 Total Safety Distance for Summer Conditions

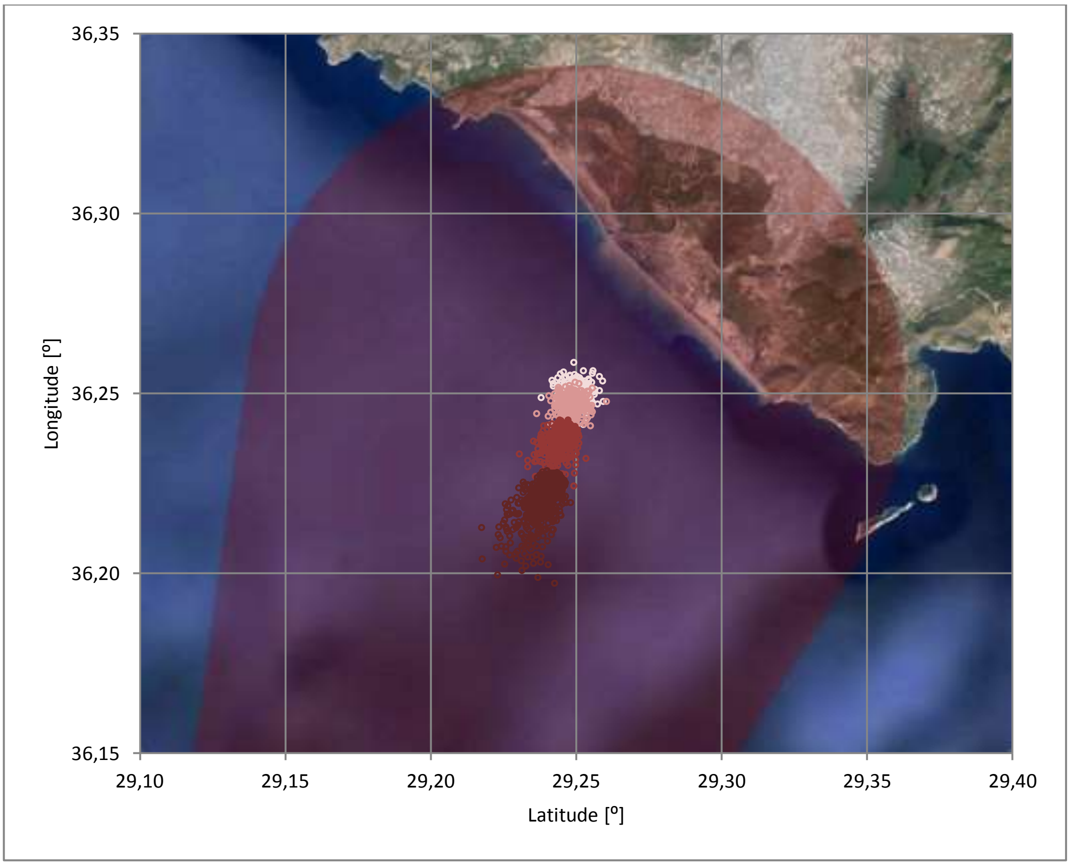

Figure 60 Total Safety Distance for Winter Conditions

It can be seen from Figure 59 and Figure 60 that safety distance changes significantly for different seasons. Although debris drop points are similar for summer and winter conditions, variation of surface wind speed changes the safety zone required for toxic pollutants. Safety zone requirement due to secondary effects is higher than the requirement due to drop points in both seasons.

CHAPTER 5

CONCLUSION

5.1. SUMMARY AND CONCLUSIONS

Throughout this thesis work, effect of atmospheric parameters on space launch center safety is studied. Within this scope, first, conceptual design of a small launch vehicle is studied. Launch vehicles consist of single or multiple stages and there are many operational launch vehicles in the market with variety of shapes and sizes. Investigations on recent developments and launch activities in addition to the near term manifest of payload mass distribution show that there will be an increase in the small launch vehicle demand in the future. Moreover, Turkey also has a goal of producing a small launch vehicle and inserting small satellites to orbit for different missions. Developing a launch vehicle that is dedicated to small satellites and LEO missions can provide advantage in terms of unit launch cost and time for operational readiness.

Different parameters can be selected for launch vehicles as a merit of performance. It is shown in the study that velocity as a performance parameter is independent of the mass of the launch vehicle or the payload in addition to the amount of thrust or the total duration of burns. This makes velocity preferable in the conceptual design studies without trajectory analyses. In addition, staging a vehicle more than four does not provide feasible percent increase to the performance as the fewer stages provide. Complexity of the launch vehicle increases and reliability decreases with increasing number of stages.

Total required velocity of a launch vehicle consists of several components. It is not possible to determine amount of velocity losses to be included in the total velocity requirement exactly without the detailed trajectory analyses. Although suggestions on velocity losses exist in the literature, they are not accurate enough for realistic conceptual studies. Small launch vehicles and their payload capacities are examined for the determination of losses. Polynomials for different orbits independent of launch site location and dependent to target orbits altitude and inclination are obtained. These polynomials, which provide better estimations, can be used on conceptual design phase without detailed trajectory analyses or additional effort for separate calculations on each component.

Although it is easy to obtain optimum mass distribution for a simple rocket, staging and existence of boosters with motors having different structural ratios and performance makes direct solutions inapplicable. Requirement for numerical methods to find related Lagrange multipliers arises for complex launch vehicles. It is also not possible to obtain ideal mass distributions without assumptions on fairing or booster jettisoning time. Although general solution for the optimum launch vehicle size in terms of gross lift-off weight is presented, method of Lagrange multipliers is not used for the determination of mass distribution.

History of space programs shows that heritage of the past launch vehicles belongs to ballistic missiles developed for martial purposes. However, it can be seen from recent developments that design process of new vehicles become cost effective commercial operations with increasing launch demand and companies carrying out their own space programs independent of their governments. Cost effective solutions for the launch market must be provided due to the fact that even small launch vehicles which are only capable of carrying light payloads to orbit cost millions of dollars. A parametric cost model, TRANSCOST, is also implemented to the solution in a component based manner for the simultaneous consideration of performance and cost.

Mass distributions of selected launch vehicle concepts are derived and alternative configurations are compared with help of the performance and cost models for small launch vehicles. Comparisons among alternatives show that launch vehicle solutions with minimum gross lift-off weight or optimum mass distribution do not provide the best solution in terms of total project or lifetime costs. This study shows that project goals and targeted market also plays a role in the concept selection due to the fact that lifetime cost of a launch vehicle depends on the total number of annual launches and planned operational life. In addition, development costs can be reduced by hardware commonality and lifetime cost of a vehicle can be lower than project cost which includes production of only one item for a certain number of launches. After the comparison of alternative configurations, best alternative is selected for the safety analysis.

There are two types of possible main hazards for the flight phase of launch vehicle. Possible launch vehicle fragments in case of an inflight explosion can drift long distance due to high altitude winds and secondary effects can occur such as overpressure, thermal effects and toxic pollution. In order to determine the safety distances, an accurate knowledge of atmospheric parameters is required. For this purpose difference between global and reference atmosphere models were presented. For determination of debris trajectories under the influence of winds, a seasonal random wind model to be implemented in trajectory analysis is presented.

Trajectory analysis of possible launch vehicle fragments are conducted using the statistical values of atmospheric parameters. Three degrees of freedom flight path equations of motions is modified to consider winds and drop points' areas for the fragments are obtained. Atmospheric conditions of the selected space launch center location are considered for the determination of safety distances due to secondary effects. Drop point areas of debris differs seasonally due to changing wind speed. Safety distances of secondary effects also change due to surface winds.

Obtained results of the safety analysis shows that most dominant factor in space launch safety is the toxic pollution for the investigated case. This is mainly due to the usage of solid propellant on the first and second stages of the launch vehicle. Selecting a launch vehicle configuration with non-toxic propellants such as liquid oxygen and kerosene can be a solution to the high safety distances. In addition, trajectory of the launch vehicle can be modified in such a way that launch vehicle diverges from the populated areas faster. However, it is not an easy task due to the fact that launch vehicles starts their flight vertically and rate of change of flight path is limited to reduce the aerodynamic loads on the vehicle.

Presented methodology for the determination of launch vehicle mass distribution, which considers the lifetime costs of the vehicle, can be used for conceptual design studies such as evaluation of alternatives in different aspects. Velocity requirement approach for performance evaluation provides accurate enough results. It can be applied on launch vehicles with different size classes or different operational types such as reusable ones.

Modified three degrees of freedom equations of motions, which are considering the effects of wind on the trajectory, can also be used for the analysis of launch vehicle trajectory. In addition, easy to implement methodology explained for the space launch center safety can be used for rapid evaluation of launch center alternatives. Safety distance calculations of secondary effects can also be applied to the storage facilities of the space launch center which may contain many ready to flight stages for upcoming launches.

5.2. FUTURE WORK

Although this thesis work covers many different topics and couples them together, suggestions on possible improvements and additions are listed in the following.

- Expected casualty is another aspect of space launch center safety. Probability of launch vehicle fragments falling over populated areas and harming inhabitants also should be considered for the space launch center safety. A common value accepted for expected casualty is 30×10^{-6}. This value depends on probability of impact of each possible fragment and launch vehicle reliability. Expected casualty analysis can be conducted as an additional study for short term.

- Since safety depends on the mission (target orbit, carried payload etc.) and the trajectory, it should be repeated for different cases. Comparison of different atmospheric conditions can be conducted such as day and night or monthly evaluations. Extended study for a selected launch center location can reveal the available launch windows intervals throughout the year.

- Safety analyses are conducted for each mission repeatedly. In some cases, real time debris patterns are used for the decision of launch or flight abort. In this case initial conditions used for the analyses are the real time trajectory data of the launch vehicle which is always different than the pre-flight predictions. Codes presented in this thesis can be supported by additional capabilities and integrated to the decision making mechanisms.

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APPENDIX A

MATLAB CODE OF TRAJECTORY SOLUTION

Computer codes used for the trajectory simulations were developed using MATLAB® (Version R2013b, 8.2.0.701).

A.1 MAIN FUNCTION

```
function point = mainfunction()
% Last Modified @ 13.12.2015 @YAMAN

close all
clc

global meanU;   global meanV;
global sigmaU;  global sigmaV;
global num;     global num2;
global HGHT;    global DENS;    global num3;

% GET Wind NFO
[num]  = xlsread('CorrCoef.xlsx','Inter_Intra');
[num2] = xlsread('CorrCoef.xlsx','Cross_Intra');
% altitude = num(:,1);    % [km ]
meanU = num(:,2);         % [m/s]
meanV = num(2,:);         % [m/s]
sigmaU = num(:,3);        % [m/s]
sigmaV = num(3,:);        % [m/s]
% GET Atmos NFO
[num3] = xlsread('AtmosData.xlsx','DensDATA');
HGHT = num3(:,1);
DENS = num3(:,2);

% Global Variables for Trajectory
global dtr;     dtr = 0.01745329252;
global Re;      Re = 6378137;
global OMEG;    OMEG = 7.29212E-5;
global MASS;    MASS = 1;
global WIND;    altlvl = 10;
global BCOEF;   BMIN = 0.002; BMAX = 0.114;

%%%%%%%%%%%%%%%
nmax = 5000; %% Number of Runs
%%%%%%%%%%%%%%%

point = zeros(6,nmax);
color = zeros(1,nmax);
```

```
for n = 1:1:nmax

WIND = sentetic(altlvl);
BCOEF = BMIN+rand*(BMAX-BMIN);

% Initial Conditions for Trajectory
POST = 8704+Re;
LONG = 29.2438*dtr;
DECL = 36.2266*dtr;
VELO = 437.88;
FPAA = 62.31*dtr;
AZMI = 192.2*dtr;

VECP = imparted(BCOEF,VELO,FPAA,AZMI);
VELO = VECP(1);
FPAA = VECP(2);
AZMI = VECP(3);

% Print Results at each step
fprintf(' BCOEF  NUMBER    VELO  FPATH   AZIMT \n')
fprintf('%6.4f %7.0f %7.2f %6.2f %6.2f \n', BCOEF, n, VELO,
FPAA/dtr, AZMI/dtr)

difinit = [POST LONG DECL VELO FPAA AZMI]';
tspan = 0:1:2000;

options = odeset('RelTol',1e-1,'MaxStep',0.1,'events',@event);
[~,~,~,YE,~] = ode23(@fpeomswind, tspan, difinit, options);
% keyboard
point(:,n) = YE;
color(:,n) = BCOEF;

end

scatter(point(2,:)/dtr,point(3,:)/dtr,[],color)
grid on
xlabel('Longitude [deg]')
ylabel('Lattitude [deg]')
ylim([36.2 36.3])
xlim([29.2 29.3])
axis equal
hold on
scatter(29.25,36.25,'r')
% scatter(29.25,36.23,'r')
hold off

xlswrite('RESULTS.xlsx',point,'Sheet1');
xlswrite('RESULTS.xlsx',color,'Sheet2');
end
```

A.2 RANDOM WIND PROFILE

```
function array = sentetic(altlvl)

global meanU;   global meanV;
global sigmaU;  global sigmaV;
global num;     global num2;

% PROBABILITY
sigmalvl = 2.57585;
P = erf(sigmalvl/sqrt(2));

% STORE
U = zeros(altlvl+2,1);
V = zeros(altlvl+2,1);
Alt = zeros(altlvl+2,1);

% % % % % % % % % % % % % % % % % %
% % GET Wind NFO
% [num]  = xlsread('CorrCoef.xlsx','Inter_Intra');
% [num2] = xlsread('CorrCoef.xlsx','Cross_Intra');
% % altitude = num(:,1);    % [km ]
% meanU = num(:,2);        % [m/s]
% meanV = num(2,:);        % [m/s]
% sigmaU = num(:,3);       % [m/s]
% sigmaV = num(3,:);       % [m/s]
% % % % % % % % % % % % % % % % % %
% This Section is embedded in Mainfunction for computational time

% REQUIRED VARIABLES @ STARTING ALTITUDE
MU1 = meanU(altlvl+4);
SDU1 = sigmaU(altlvl+4);
MV1 = meanV(altlvl+4);
SDV1 = sigmaV(altlvl+4);
RU1V1 = num(altlvl+4,altlvl+4);

% RANDOM WIND @ STARTING ALTITUDE
CA = 360*rand;
% fprintf('\n')
Akare = (1/(1-RU1V1^2))*((cosd(CA)/SDU1)^2-...
    2*RU1V1*cosd(CA)*sind(CA)/(SDU1*SDV1)+...
    (sind(CA)/SDV1)^2);
A = sqrt(Akare);
le = sqrt(2)*sqrt(-log(1-P));
RS = 1/(A/le)*rand;        %!!!

U1p = MU1 + RS*cosd(CA);
V1p = MV1 + RS*sind(CA);

U(1) = U1p;
V(1) = V1p;
Alt(1) = altlvl;
U(altlvl+1) = U1p;
V(altlvl+1) = V1p;
Alt(altlvl+1) = altlvl;
```

```
for n = 1:1:altlvl
% REQUIRED VARIABLES @ NEW ALTITUDE
MU2 = meanU(altlvl+4-n);
SDU2 = sigmaU(altlvl+4-n);
MV2 = meanV(altlvl+4-n);
SDV2 = sigmaV(altlvl+4-n);
RU2V2 = num(altlvl+4-n,altlvl+4-n);
% INTERLEVEL CORRELATIONS
RU1U2 = num(altlvl+4,altlvl+4-n);
RV1V2 = num(altlvl+4-n,altlvl+4);
% CROSSLEVEL CORRELATIONS
RU1V2 = num2(altlvl+4,altlvl+4-n);
RV1U2 = num2(altlvl+4-n,altlvl+4);
RV2U1 = RU1V2;
RU2V1 = RV1U2;
% CONDITIONAL MEAN VECTOR
T1 = (RU1U2-RU1V2*RU1V1)*(U1p-MU1)*(SDU2/SDU1);
T2 = (RU1V2-RU1U2*RU1V1)*(V1p-MV1)*(SDU2/SDV1);
T3 = (RV1U2-RV1V2*RU1V1)*(U1p-MU1)*(SDV2/SDU1);
T4 = (RV1V2-RV1U2*RU1V1)*(V1p-MV1)*(SDV2/SDV1);
CMU2 = MU2+(T1+T2)/(1-RU1V1*RU1V1);
CMV2 = MV2+(T3+T4)/(1-RU1V1*RU1V1);

% COVARIANCES
% Defn.: SUiVj = RUiVj*SDUi*SDVj
SU1U2 = RU1U2*SDU1*SDU2;
SU1V1 = RU1V1*SDU1*SDV1;
SU1V2 = RU1V2*SDU1*SDV2;
SU2V1 = RU2V1*SDU2*SDV1;
SU2V2 = RU2V2*SDU2*SDV2;
SV1V2 = RV1V2*SDV1*SDV2;

D = (SDU1^2)*(SDV1^2)-SU1V1^2;

% COVARIANCE MATRIX SIGMA COMPONENTS
sigma11 = (SDU2^2)-SU1U2*(SU1U2*(SDV1^2)-SU2V1*SU1V1)/D-...
    SU2V1*(-SU1U2*SU1V1+SU2V1*(SDU1^2))/D;

sigma22 = (SDV2^2)-SU1V2*(SU1V2*(SDV1^2)-SV1V2*SU1V1)/D-...
    SV1V2*(-SU1V2*SU1V1+SV1V2*(SDU1^2))/D;

sigma12 = SU2V2-SU1V2*(SU1U2*(SDV1^2)-SU2V1*SU1V1)/D-...
    SV1V2*(-SU1U2*SU1V1+SU2V1*(SDU1^2))/D;

% CONDITIONAL STANDART DEVIATIONS
CSDU2 = sqrt(sigma11);
CSDV2 = sqrt(sigma22);

% CONDITIONAL CORRELATION COEFFICIENT
CRU2V2 = sigma12/(CSDU2*CSDV2);

CC = 360*rand; %CA; %
Ackare = (1/(1-CRU2V2^2))*((cosd(CC)/CSDU2)^2-...
    2*CRU2V2*cosd(CC)*sind(CC)/(CSDU2*CSDV2)+...
    (sind(CC)/CSDV2)^2);
```

```
Ac = sqrt(Ackare);

le = sqrt(2)*sqrt(-log(1-P));
RSC = 1/(Ac/le)*rand;                    %!!!
U2 = CMU2 + RSC*cosd(CC);
V2 = CMV2 + RSC*sind(CC);

U(altlvl+1-n) = U2;
V(altlvl+1-n) = V2;
Alt(altlvl+1-n) = altlvl-n;
% U(n+1) = U2;
% V(n+1) = V2;
% Alt(n+1) = altlvl-n;
end

U(altlvl+2) = U(altlvl+1);
V(altlvl+2) = V(altlvl+1);
Alt(altlvl+2) = altlvl*9;

array = [Alt U V]

end
```

A.3 TRAJECTORY SOLUTION

```
function [xdot] = fpeomswind(t,x)

global BCOEF; global MASS; global OMEG; global Re;
global dtr; global WIND;

gr = 3.986004e14/x(1)^2;

ALTI = (x(1)-Re)/1000;
if ALTI < 0
    ALTI = 0;
end

if x(5) == 0
    x(5) = 0.0001;
end

if x(5) == pi/2
    x(5) = pi/2-0.0001;
end

WINDu = interp1(WIND(:,1),WIND(:,2),ALTI);
WINDv = interp1(WIND(:,1),WIND(:,3),ALTI);
WINDs = sqrt(WINDu^2+WINDv^2);
WINDa = 180*dtr+atan2(WINDv,WINDu);

VELOp = sqrt(x(4)^2+WINDs^2-...
    2*x(4)*WINDs*cos(x(5))*cos(x(6)-WINDa));
```

```
beta = acos((x(4)*sin(x(5))*sin(x(5))+...
     (x(4)*cos(x(5))*sin(x(6))-WINDs*sin(WINDa))*...
     cos(x(5))*sin(x(6))+...
     (x(4)*cos(x(5))*cos(x(6))-WINDs*cos(WINDa))*...
     cos(x(5))*cos(x(6)))/VELOp);

if x(1) < Re
     x(1) = Re;
end

RHO = AtmosDens(x(1)-Re);

DRAGF = 0.5*RHO*(VELOp^2)*BCOEF*MASS;

XF = -DRAGF*cos(beta);
YF = -DRAGF*sin(beta);
ZF = 0;

LONGd = x(4)*cos(x(5))*sin(x(6))/(x(1)*cos(x(3)));

DECLd = x(4)*cos(x(5))*cos(x(6))/x(1);

POSTd = x(4)*sin(x(5));

VELOd = XF/MASS-gr*sin(x(5))+...
     (OMEG^2)*x(1)*cos(x(3))*(sin(x(5))*cos(x(3))-...
     cos(x(5))*cos(x(6))*sin(x(3)));

FPAAd = ZF/(MASS*x(4))+x(4)*cos(x(5))/x(1)+...
     2*OMEG*sin(x(6))*cos(x(3))-gr*cos(x(5))/x(4)+...
     (OMEG^2)*(x(1)*cos(x(3))/x(4))*(cos(x(5))*cos(x(3))+...
     sin(x(5))*cos(x(6))*sin(x(3)));

AZMId = -YF/(MASS*x(4)*cos(x(5)))+...
     x(4)*cos(x(5))*sin(x(6))*tan(x(3))/x(1)+...
     (OMEG^2)*x(1)*sin(x(6))*sin(x(3))*cos(x(3))/(x(4)*cos(x(5)))-
...
     2*OMEG*(tan(x(5))*cos(x(6))*cos(x(3))-sin(x(3)));

xdot = [POSTd LONGd DECLd VELOd FPAAd AZMId]';
end
```

A.4 STOP CONDITION

```
function [stopvalue, isterminal, direction] = event(t,x)

stopvalue = x(1)-6378137+1;
isterminal = 1;
direction = 0;

end
```

A.5 IMPARTED VELOCITY

```
function velocity = imparted(bc,vel,fpa,azm)

% imparted([-],[m/s],[rad],[rad])

% Here is the magic numbers :)
ivmin  =  51.920*(bc^0.5718);
ivmax  = 830.900*(bc^0.2801);

% In case we use the mean directly:
% mean =  70.993*(bc^0.1986);

ivel = ivmin+rand*(ivmax-ivmin);
ifpa = rand*pi-pi/2;
iazm = 2*rand*pi;

iz = ivel*sin(ifpa);
ix = ivel*cos(ifpa)*cos(iazm);
iy = ivel*cos(ifpa)*sin(iazm);

iz1 = vel*sin(fpa);
ix1 = vel*cos(fpa)*cos(azm);
iy1 = vel*cos(fpa)*sin(azm);

izp = iz + iz1;
ixp = ix + ix1;
iyp = iy + iy1;

velp = sqrt(ixp^2 + iyp^2 + izp^2);
azmp = atan2(iyp,ixp);
fpap = atan2(izp, sqrt(ixp^2+iyp^2));

% New Velocity Vector as initial conditions
velocity = [velp fpap azmp]';

end
```

APPENDIX B

COST MODEL

The only prevailing principle for the launch vehicle design was the maximum performance and the minimum weight. However, decreasing the weight of the launch vehicle components leads to increased costs. This phenomenon is due to the requirement of using highly technological and expensive materials. The new principle is to satisfy the mission requirements with the minimum cost. Following the successful design and operation of many launch vehicles, design process of new launch vehicles becomes cost effective commercial operations instead of national reputation. Using off the shelf and highly reliable components can decrease development and production cost in addition to the reduction of the risks in the design process.

It is possible to define three different design types related with launch vehicle costs.

- Design without Specific Cost Requirements
- Design with a Limited Budget
- Application of Cost Engineering

Designing a launch vehicle without any constraint has the goal of optimizing the LV performance for the defined mission. On the contrary, design with a limited budget aims to achieve the required performance without exceeding a predefined maximum development budget. In cost engineering applications, main goal is to attain minimum development cost and/or production and operation costs. In such a study, it is important for the design team to have knowledge and decision capability about the cost factors.

It is important to start cost analyses in the early phases of the design instead of the detailed design period. Bottom-up methods which are applied during the detailed design phase may result in unacceptable total costs and risk the realization of project. Application of cost engineering methods for the analyses is helpful for selection of the launch vehicle configuration and estimation of the costs from the beginning of the projects.

B.1. MAJOR PHASES

In a launch vehicle project, the common method is to define distinct phases for the program. Phases for space transportation systems of a project can be defined as follows;

Pre-Phase A: Investigating the possible launch vehicle concept options and selection of the launch vehicle concept with the minimum cost

Phase A: Launch vehicle conceptual design, determination of optimum vehicle size in terms of performance and overall cost

Phase B: Definition of the system design and determination of specifications and development projects in addition to the verification of cost estimations with bottom-up methods

Phase C: Development of launch vehicle systems and subsystems, integration and verification of launch vehicle and flight tests for the qualification

Phase D: Continuous production of the vehicle parts and integration of launch vehicle for the launch operations

Phase E: Operations to fulfill the mission requirements, realization of the project with successful launches

Performing of cost engineering studies is most important in **Pre-Phase A** due to the fact that all decisions such as number and type of stages, which have a major impact on cost and economics, are made in this phase. After completing the concept selection, conceptual design of the launch vehicle and determination of its specifications is carried out during the phases A and B.

Since fast methods for conceptual design studies are preferred especially in LV design, method for cost estimation should also be easy to apply. This is only possible by using parametric models derived with help of previous data.

B.2. COST MODELS

Although reliable cost data are not accessible in literature, cost estimation for a launch vehicle project should be based on previous projects and past experience. For accurate cost estimation, it is necessary to define different cost areas which depend on different technical criteria. Key cost areas for a launch vehicle project are as follows;

- Development Costs
- Production Costs
- Direct and Indirect Operation Costs

There are few cost models which are using different methodologies for launch vehicle projects. PRICE-H, TRANSIM and TRANSCOST are the most common examples to cost models. PRICE-H is a component level model and not really dedicated to launch vehicles. It has a confidential database and does not provide information about operation costs of launch vehicles. TRANSIM is a model that is specialized for space transportation. Although it provides cost estimations for every key cost area, source for its database is unknown. TRANSCOST model on the contrary, has a visible database, provides cost estimations clearly using the defined cost estimation relationships and it is a launch vehicle dedicated model. TRANSCOST which has been established by D. E. Koelle is based on a continuously updated engine and vehicle database. It is used by space agencies, aerospace companies and institution at Europe, USA, Russia, China, Japan and India.

B.2.1. TRANSCOST Model

The TRANSCOST model for space transportation systems has been established in 1971 and updated until today for the conceptual design phase of launch vehicle project by Koelle. Since it is not feasible to consider all of the subsystems of a LV during the conceptual design phase, model only provides system level cost estimation by analytical cost models. Instead of classified database, model also provides information about reference projects.

TRANSCOST can be used for

- selection among launch vehicle concepts in the conceptual design phase,
- determination of optimum vehicle size in terms of the total project cost and
- evaluation of using existing components instead of new development.

Model structure consists of three submodels. It is possible to make estimations in all key cost areas separately and combine them for different applications for the determination of overall project cost.

- The Development Cost Submodel
- The Vehicle Cost Submodel (Production, Integration and Verification)
- The Ground and Flight Operations Submodel

Different system groups are defined for each submodel. Technical Development System Groups, which are included in the model and related only with the development of expandable launch vehicles, are given in Table 25.

Table 25 Technical Development System Groups

Code:	**Explanation:**
ES	Solid Propellant Rocket Motors
EL	Liquid Propellant Rocket Engines with Turbopumps
EP	Pressure-fed Rocket Engines
VR	Solid Propellant Rocket Boosters
VP	Propulsion Systems / Modules
VE	Expendable Ballistic Rocket Vehicles

For Vehicle Cost Submodel, number of defined cost estimation relationships (CER) which are related to expendable launch vehicles and to be used for production, integration and verification costs, are reduced to the items given in Table 26.

Table 26 Technical Production System Groups for Vehicle Cost Submodel

Code:	**Explanation:**
ES	Solid Propellant Rocket Motors
EL	Liquid Propellant Rocket Engines with Turbopumps
VP	Propulsion Systems / Modules
VE	Expendable Ballistic Rocket Vehicles

CERs associated with the technical system groups are generally mass related with a basic form $C=aM^x$. Where C represents the cost of the item, a and x are system specific constant values and M is mass in kilograms.

For the Ground and Flight Operations Submodel, cost estimation relationships are based on the type of the activities implemented instead of reference mass values of systems. Except the specific cases given below, some assumptions are required for uncovered direct and indirect operation costs.

- Prelaunch Ground Operations
- Launch and Mission Operations
- Ground Transportation (and Recovery if applicable)
- Propellants, Gases and Material
- Program Administration and System Management
- Technical System Support
- Launch Site and Range Cost

Most of the CERs given in the TRANSCOST model are related with mass. However, this does not mean that costs are directly related to system mass. Realistic mass estimations rely on accurate relationships. In order to achieve accurate enough relations, there is degression factors defined for different cases.

Another specific property of TRANSCOST is the implementation of the Work-Year effort as costing value instead of Man-Year (MYr) which is not applicable for different time periods because of the inflation rates. Work-Year (WYr) is total annual budget divided by the number of productive full-time employees by definition. Secondary costs such as travel, office, taxes and profit in addition to a certain share for the administrative and support stuff costs are included in Work-Year value.

B.2.1.1. Development Costs

Estimation of development costs for a launch vehicle project is the most difficult task when compared to other key cost areas due to the fact that there are many technical criteria which have an impact on development process itself. Major development cost drivers are as follows;

- Vehicle launch mass and size
- Number and types of stages
- Technology readiness / scope of existing subsystems, components
- Type and number of engines
- Reliability and safety requirements
- Verification and test strategy
- Number of flight units and flight tests
- Company and team experience
- Program organization and management procedures
- Program budget planning and schedule / delays
- Technical changes required or ordered by the customer
- Contract conditions, etc.

Type of the stages on a LV has impact not only on the development cost but also production and operational costs. Launch vehicles with storable liquid propellants require 2 to 3 stages while launch vehicles with cryogenic or solid propellants require 1 to 2 or 3 to 4 stages to fulfill the mission requirements. Minimizing the number of stages can decrease the development costs by reducing the number of systems that are required to be developed. However, sizes of the stages are also required to be considered for the costs.

Development cost includes all of the activities from the detailed design to verification of the launch vehicle. All of the ground stations such as launch pad, assembly halls or test facilities in addition to the production of first flight unit are also included in development cost. Development Cost Submodel is made up of three elements; launch vehicle CERs (H_V), engine CERs (H_E) and booster CERs (H_B) if applicable. Total development cost for a launch vehicle (C_D) can be represented by the sum of all the related cost items.

$$C_D = f_0\left(\sum H_B + \sum H_V + \sum H_E\right) f_6 f_7 f_8 \tag{B.1}$$

Considering the previous launch vehicle projects, it is necessary to add some mass margin to launch vehicle or its subsystems in order to take some possible additional requirements that may arise during the detailed design phase into consideration for overall development costs. Depending on the phase of the project, this mass margin should be in between 5 – 20 %.

Development Costs Degression Factors

Degression factors included in the total development cost are represented by main script f in the equation. f_6, f_7 and f_8 are programmatic cost factors and not considered in the development costs due to the fact that they are not related with technical properties of the launch vehicle.

f_0 - *System Engineering / Integration Factor*

In order to be able to consider the impact of number of stages involved in a launch vehicle f_0 can be used with a value of 1.04^N where N is the number of the stages.

f_1 – *Technical Development Status Factor*

Effort required for development can be compared with similar projects. It is possible to use completely new technologies or existing components in a LV or use existing components with minor changes. Selection of this factor is subjective and the recommended intervals for different cases are given in Table 27.

Table 27 Recommended Intervals for Technical Development Status Factor

Factor:	Explanation:	Interval:
f_1	First of its kind, use of new technologies	1.3 - 1.4
	Design with some new features but existing technologies	1.1 - 1.2
	State of the art standard projects	0.9 - 1.1
	Application of design modifications on existing systems	0.7 - 0.9
	Minor changes on existing projects	0.4 - 0.6

f_2 – *Technical Quality Factor*

This is a factor representing the technical characteristics of a project. Value of this factor is defined in a different way for each technical system depending on relative net mass fraction, performance or another cost impact factor.

f_3 – Team Experience Factor

It is clear that an inexperienced team will need a higher development effort when compared to a team worked on a similar task for the completion of a new project. Recommended intervals for different cases are given in Table 28.

Table 28 Recommended Intervals for Technical Development Status Factor

Factor:	Explanation:	Interval:
f_3	New team and no company experience	1.3 - 1.4
	Partially new activities for the team	1.1 - 1.2
	Team with some related experience	1.0
	Experience on development of similar projects	0.8 - 0.9
	Superior experience on the project	0.7 - 0.8

Solid Propellant Rocket Motors Development Costs

This group of propulsion system covers solid propellant rocket motors with fixed nozzle such as simple strap on boosters or apogee kick motors. Included CER for this group with *M* being the burnout mass of the solid propellant rocket engine is as follows;

$$H_{ES} = 16.3 M^{0.54} f_1 f_3 \tag{B.2}$$

Liquid Propellant Rocket Engines with Turbopumps

This group covers liquid propellant engines with turbopumps working with both storable and cryogenic propellants. Reference value M is again the dry mass of the engine. There are references in literature for mass values of certain thrust levels if the engine specifications are unknown.

$$H_{ES} = 277 M^{0.48} f_1 f_2 f_3 \tag{B.3}$$

f_2 *– Technical Quality Factor*

Specific impulse or the type of the propellant is not the major cost factor in development of liquid propellant rocket engines. Required operational reliability level determines the number of qualification tests and has a direct impact on development cost. Technical Quality Factor f_2 can be determined by using the Equation (B.4). N_Q is the number of qualification test firings. Reference point for f_2 is 1.0 with 500 tests.

$$f_2 = 0.026 \left(\ln N_Q \right)^2 \tag{B.4}$$

Liquid Propellant Rocket Engines (Pressure-fed)

Thrust level for pressure-fed liquid propellant engines is limited. They are widely used on upper stage propulsion modules or secondary systems for attitude control. In such systems propellant and / or oxidizer tank pressures kept low in order to decrease the structural weights. Since operating pressures of such engines are low, development costs are lower than the systems with turbopumps.

$$H_{EP} = 167 M^{0.35} f_1 f_3 \tag{B.5}$$

Solid Propellant Rocket Boosters

This group covers large solid rocket boosters with steerable nozzles or secondary fluid injection systems for thrust vectoring as in the case of Ariane 5. CER given in the model for this group is given in Equation (B.6).

$$H_{VR} = 10.4M^{0.60} f_1 f_3 \tag{B.6}$$

Liquid Propellant Propulsion Systems / Modules

In this group, propulsion systems that are both integrated on spacecraft and upper stages are covered. These types of systems have their own basic structure but no power supply or control system.

$$H_{VP} = 14.2M^{0.577} f_1 f_3 \tag{B.7}$$

Liquid Propellant Expandable Stages

Launch vehicle stages that are using liquid propellants are covered in this section. Reference value for the CER is the vehicle or stage dry mass excluding the weight of the engine(s). Weights of secondary structures such as payload fairing or inter-stage structure between the first and second stages should be included in the net mass fraction of the first stage for a two staged LV.

$$H_{VE} = 100M^{0.555} f_1 f_2 f_3 \tag{B.8}$$

f_2 – Technical Quality Factor

In order to determine the technical quality factor, it is required to calculate the net mass fraction (NMF). NMF is defined as the dry mass plus the residuals and gases at the burnout excluding the weight of the engine(s). Technical quality factor f_2 is determined by the ratio of reference constant k^* to effective constant k_{eff}. M_n is the net mass of the stage and M_e is the engine mass.

$$k_{eff} = \left(M_n - M_e\right) / M_p \tag{B.9}$$

$$f_2 = k^* / k_{eff} \tag{B.10}$$

k^* can be obtained for stages with storable propellants from Figure 61 depending on the propellant mass of the stage.

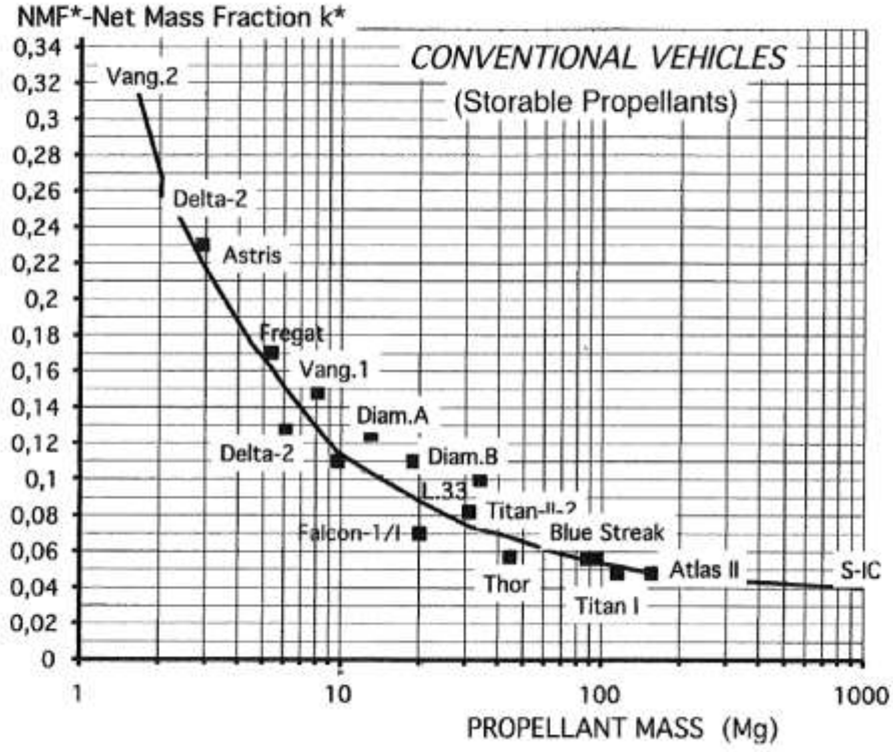

Figure 61 *k** Reference Curve (Koelle, 2010)

B.2.1.2. Production Costs

Major part of production costs are made up of secondary costs such as engineering support. Even in the case of using highly expensive materials, material costs are relatively lower when compared to other costs. Production costs include material, processing and manufacturing, assembly, verification and test costs in addition to the secondary costs such as engineering support and quality assurance. Production of special tools required for the assembly is also a part of production costs. Total production cost of a LV is generally about 1.5 – 2.0 % of the developments costs of the project.

f_4 – Production Quantity p and Learning Factor

Learning factor, f_4 in production costs represents the impact of production quantity. It is one of the most important factors affecting the production cost. Reduction in the production effort based on the number of the units that are going to be produced with an annual production rate is considered in unit production costs with help of learning factor. p indicates the new percentage of the production cost for the next production item with respect to old one. Applicability of learning factor depends on items being completely identical without any technical changes.

For launch vehicles, value of p is in between 0.8 and 1.0. For larger annual production rate, learning factor f_4 decreases. f_4 for a batch of n units can be determined from learning factor chart or using the Equation (B.11) where n is the number of units built.

$$f_4 = \frac{1}{n}\sum_{1}^{n} n^{\frac{\ln p}{\ln 2}} \qquad \text{(B.11)}$$

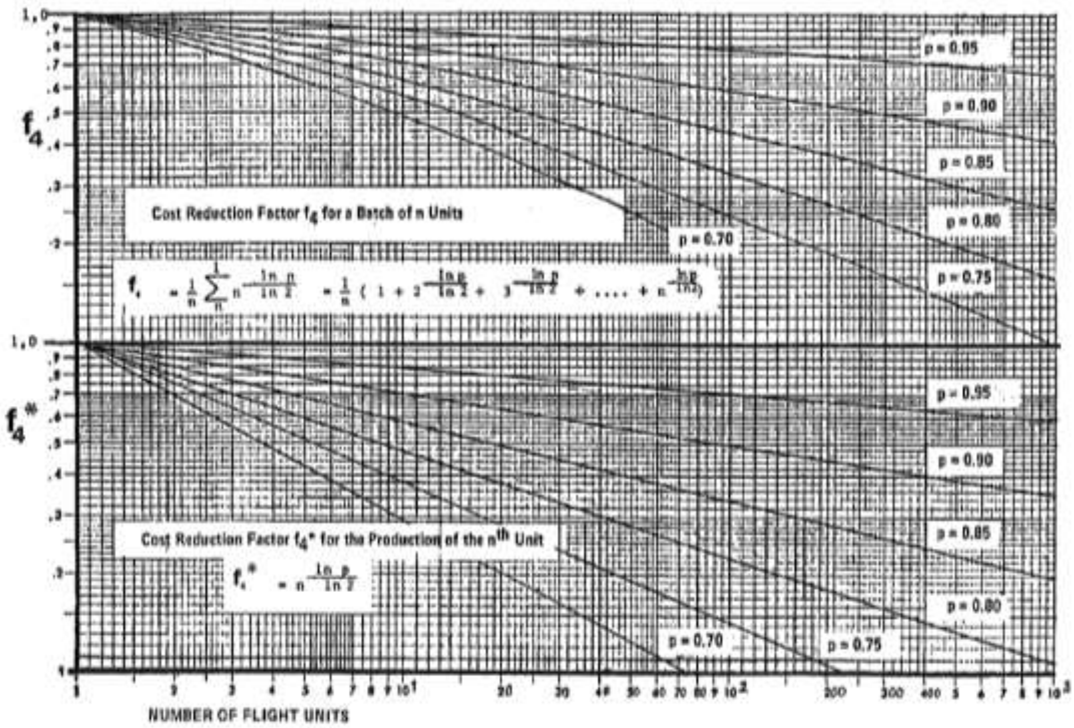

Figure 62 k* Learning Factor Chart (Koelle, 2010)

Except the engines of liquid propellant stages, production cost submodel of TRANSCOST does not provide cost information of subsystems. Basic form of the cost estimation relationships is as follows;

$$F = naM^{x} f_4 \tag{B.12}$$

where

F is the required total effort for the production,

n is the number of units,

x is the specific sensitivity value for the unit,

a is the specific constant for the unit,

M is reference mass value in kg and

f_4 is the learning factor.

CERs for the following items are given in model;

- Solid Propellant Motors and Boosters
- Cryogenic Rocket Engines
- Rocket Engines with Storable Propellants
- Monopropellant Rocket Engines
- Propulsion Modules
- Ballistic Vehicles

Production costs for launch vehicle is equal to sum of all elements multiplied by a system management factor f_0. N in the equation represents the number of stages and n is the number of identical units such as boosters or engines. Value of f_0 is in between 1.02 and 1.04 for modular systems and is equal to 1.4 for launch vehicles with boosters due to the vehicle complexity.

$$C_F = {f_0}^{N} \left(\sum_{1}^{n} F_S + \sum_{1}^{n} F_E \right) \tag{B.13}$$

Engine Production Costs

CERs for production costs of different items are given in the following section. Degression factors f_{10} and f_{11} are technical cost reduction factor and commercial development factor, respectively. By using the knowledge gained from previous projects and due to the fact that launch vehicle related electronics become cheaper, it becomes possible to reduce launch vehicle production costs for future projects.

$$f_{10} = 0.85 – 0.75$$

In addition, for commercial projects completely independent from governmental organizations, it is necessary to use commercial development factor for cost degression. For commercial projects, it is no more necessary to achieve many revision meetings. Since this will result in less effort as well as reduced requirement for excessive documentation, production costs can be decreased.

$$f_{11} = 0.45 – 0.55$$

Solid Propellant Rocket Motors and Boosters

This CER covers variety of solid rocket motors including small kick engines on upper stages, boosters and large motors with TVC capability. Cost of the solid propellant is also included in the relation.

$$F_{ES} = 2.3M^{0.40} f_4 f_8 f_{10} f_{11} \tag{B.14}$$

Liquid Propellant Engines

Previous studies show reducing the chamber pressure of engines does not reduce the engine costs. However chamber pressures higher than 120 bar result in cost growth due to the use of expensive materials in production. There are three different engine groups given in TRANSCOST for liquid propellants.

- Cryogenic Propellant (Pump-fed Engines)
- Storable Propellants (Pump-fed and Pressure-fed Engines)
- Monopropellant Rocket Engines

For small monopropellant and bipropellant (storable) engines, there are two different CERs which are given in Equations (B.15) and (B.16).

$$F_{EP(m)} = 1.2M^{0.535} f_4 f_8 f_{10} f_{11} \tag{B.15}$$

$$F_{EP(b)} = 1.85M^{0.535} f_4 f_8 f_{10} f_{11} \tag{B.16}$$

For pump-fed engines using cryogenic propellants;

$$F_{EL(c)} = 3.11M^{0.535} f_4 f_8 f_{10} f_{11} \tag{B.17}$$

Finally, for engines working with storable propellants, independent of propellant fed type, CER included in the model is as follows;

$$F_{EL(s)} = 1.9M^{0.535} f_4 f_8 f_{10} f_{11} \tag{B.18}$$

Due to the fact that production costs are reduced with the help of using advanced production techniques recently, it is also possible to use a generic CER which is given in Equation (B.19) for all modern rocket engines.

$$F_{EP} = 1.2M^{0.535} f_4 f_8 f_{11} \tag{B.19}$$

System Production Costs

Propulsion Modules

Automated Transfer Vehicle, ATV on top of Ariane 5 launch vehicle is an example for propulsion modules. Propulsion modules have their own structure but no electronic equipment for attitude control, telemetry or power. They consist of a multi tank assembly with several thrusters. Production costs of propulsion modules are generally higher then expendable stages

$$F_{VP} = 3.04M^{0.581} f_4 f_8 f_{10} f_{11} \tag{B.20}$$

Ballistic Vehicles / Stages

This group covers the launch vehicle stages. Reference mass *M* in the given CER does not include the engine mass. However, adapters and fairing as well as the electronics located on the stage are included in the CER. Since there is a distinct difference between the production costs of stages with storable propellants and cryogenic propellants, two different CER is given in TRANSCOST. CERS for storable and Cryogenic Stages are given in Equations (B.21) and (B.22), respectively.

$$F_{VP} = 0.90M^{0.63} f_4 f_8 f_{10} f_{11} \tag{B.21}$$

$$F_{VP} = 1.48M^{0.63} f_4 f_8 f_{10} f_{11} \tag{B.22}$$

B.2.1.3. Flight and Ground Operations Costs

Determination of flight and ground operations costs are complex due to

- Relationships between many operational criteria,
- Lack of reliable reference data.

Operation costs consist of two major sections;

- Direct Operations Costs (DOC)
- Indirect Operations Costs (IOC)

All activities directly related to ground preparations for the vehicle in addition to the launch and mission operations are covered by direct operations costs. Costs independent from the launch operations such as administration or management of the launch and the technical support for the launch site and remote ground stations are subject of indirect operations costs. Payload related activities are not included either in DOC or IOC. Flight and ground operations costs are about 20 – 35 % of unit production cost of the launch vehicle.

Direct Operations Costs (DOC)

Direct operations costs consist of five major areas;

- Size and complexity of launch vehicle (number of stages and boosters)
- Type of the launch vehicle (i.e. crewed, automated etc.)
- Assembly and transportation type (i.e. vertical, horizontal etc.)

- Launch type (i.e. vertical, horizontal etc.)
- Number of annual launches

Main elements of direct operations costs related with expendable launch vehicles are given in Table 29.

Table 29 Main Elements of DOC

DIRECT OPERATIONS COST	
Ground Operations	*Engineering, Site Management & Support* *Assembly, Integration and Checkout* *Launch Preparations (Erection, Prop. Loading etc.)* *Equipment Maintenance* *Pad Refurbishment*
Materials & Propellants	*Fuel (incl. Evtl. Boil off Loss)* *Oxidizer (incl. Evtl. Boil off Loss)* *Gases and other Consumables*
Flight and Mission Ops.	*Mission Plans, Evaluation & Management* *Launch and Flight Operations* *Crewed Mission Operations* *Tracking and Data Relay Operations*
Transport & Recovery	*Transportation to the Launch Site* *(ELVs and Sea Launch Ops.)* *Launch Assist Operations (i.e. Sled Launch)*
Fees and Insurance	*Launch Site User Fee per Launch* *Public Damage Insurance* *Vehicle Loss Charge* *Other Charges*

In order to prepare the launch vehicle for the mission, it is required to prepare the ground facilities, complete the functional checkout of vehicle elements, update and load the flight software to the launch vehicle, encapsulate the payload and integrate it to the vehicle, load the propellants if necessary and complete the final checkout for the interfaces.

Three main options are available for the launch vehicle final assembly. First option is the vertical assembly of the launch vehicle on the launch pad. In this option, a service tower is used for the vehicle assembly and it is moved away just before the launch. In the second option, launch vehicle assembly is completed in a remote location and carried to the launch pad in a vertical state. This option reduces the required time on the launch pad. Final option is the assembly of the launch vehicle in a horizontal way and erection of the vehicle on the launch pad with help of an erector mechanism. Loading of the propellants are completed on the launch pad. With this option, it becomes possible to conduct many missions frequently.

Ground Operations:

CER for ground operations is given in Equation (B.23).

$$C_{PLO} = 8{M_0}^{0.67} L^{-0.9} N^{0.7} f_v f_c f_4 f_8 f_{11} \tag{B.23}$$

M_0 in the above equation is the gross liftoff weight of the launch vehicle in metric tons. L is the annual launch rate of the vehicle and N is number of the stages. Small kick stages and boosters are required to be counted as half stages in the equation.

f_v in the equation is used to consider the type of the launch vehicle as follows;

- Liquid Propellant (Cryogenic) Expendable Launch Vehicles $f_v = 1.0$
- Liquid Propellant (Storable) Expendable Launch Vehicles $f_v = 0.8$
- Solid Propellant Launch Vehicles $f_v = 0.3$

It is proper to use an average f_v value for launch vehicles with a combined type such as lower stages are solid and the upper stage is using storable liquid propellants.

f_c in the equation is used to consider the type of assembly and integration operations as follows;

- Vertical assembly on the launch pad $f_c = 1.00$
- Transfer to launch pad after vertical assembly $f_c = 0.85$
- Horizontal assembly and transfer to launch pad for erection $f_c = 0.70$

Propellant Costs:

Propellant costs are only a small part of unit launch cost. Prices highly depend on the production capacity of the manufacturer. MMH is the most expensive propellant with a 340 $/kg unit cost while it is 6 – 8 $/kg for liquid hydrogen and 0.2 $/kg for liquid oxygen. Solid propellant costs are included in development and production CERS in TRANSCOST. Unit costs for solid propellants are higher for small motors.

Flight and Mission Operations Costs:

Preparations for the mission including the software update of the launch vehicle, flight control and safety until the end of the mission and tracking are the main components of flight and mission operations costs. Given CER for such operations is as follows;

$$C_m = 20\left(\sum Q_N\right) L^{-0.65} f_4 f_8 \tag{B.24}$$

L in the above equation is again the annual launch rate of the launch vehicle. Q_N is the specific value depending on the complexity of the launch vehicle. For the calculation of Q_N, following values can be added for each component on a vehicle.

- Stages with Solid Rocket Motors $Q_N = 0.15$
- Liquid Propellant Stages and Boosters $Q_N = 0.40$

Other Costs:

Remaining costs such as transport of launch vehicle elements to the launch site for the assembly or user fees and insurance costs are case specific and not easy to determine. These types of additional costs will not be considered during this study.

Indirect Operations Costs (IOC)

Indirect costs are the organizational costs of the launch provider company. This item depends on the number of employees working at the launch provider company for the service. Total cost of the company related with launch campaigns must be distributed to the annual launches. Indirect operations cost can be high for small number of annual launches. Since these types of costs depend on the organizational structure of the launch provider, they will not be considered for the comparison of different launcher options.

B.3. APLLICATION OF TRANSCOST MODEL

In order to be able to compare launch vehicle configurations in terms of overall costs, application of TRANSCOST Model in a limited level is sufficient assuming that many cost items such as launch facilities user fee or insurance fees under the main item DOC and all items under the IOC will be similar for different concepts. In addition to the development costs, which are required to be distributed over the total number of expected launches, following items should be considered;

- Vehicle Costs
  - $A_n a$ nth stage Vehicle Recurring Cost
  - $A_n b$ nth stage Engine (or motor) Recurring Cost
  - Af Other Items
- Direct Operations Costs
  - B_1 Ground Operations
  - B_2 Mission and Flight Operations
  - B_3 Propellants and Other Consumables

B.4. TEST CASE FOR COST

Achieving a successful mission for a launch vehicle depends on satisfying the total velocity requirement which consist of several components explained previously. Current optimization methods used in launch vehicle conceptual design are not applicable to all cases and hard to be implemented when additional requirements such as launch vehicle development cost is under consideration.

In this chapter, solution to the concept selection problem is introduced first. Obtained solutions have enough accuracy for conceptual design studies.

Application of parametric cost model makes it possible to compare launch vehicle concepts in terms of cost. Following the first part, example launch vehicles that can be used for a SSO mission with their weight distribution are presented and best concept in terms of total cost is selected.

Solution is generated using the Fortran 90 programming language. Prepared code has a modular structure. This provides ability of adding different subroutines for more detailed analysis when necessary. Output of each module used as an input for the following.

B.4.1. Solution Structure

Architecture of the prepared solution is given in Figure 63. There are six main modules in the solution. In addition, inputs are divided into two main sections as stage masses and orbital parameters. Orbital parameters are used only by orbital module to determine required orbital velocity considering the all loss terms depending on the launch site location and the target orbit.

There are two different options that can be used as inputs for orbital parameters. Depending on the type of the orbit, only the mean altitude or apogee and perigee altitudes in addition to the inclination of the orbit are required. Details of the orbital parameters are given in Figure 64. In case of target orbit being sun-synchronous, inclination of the target orbit and corresponding velocity requirement of the launch vehicle is calculated by Orbit Module using the Equation (2.28) and derived polynomials for loss terms considering the launch site location.

Stage Masses part includes number of stages and boosters if there is any, amount of propellant proposed for the each stage as an interval, specific impulses of the motors, weight of fairing and its expected time of separation, as inputs.

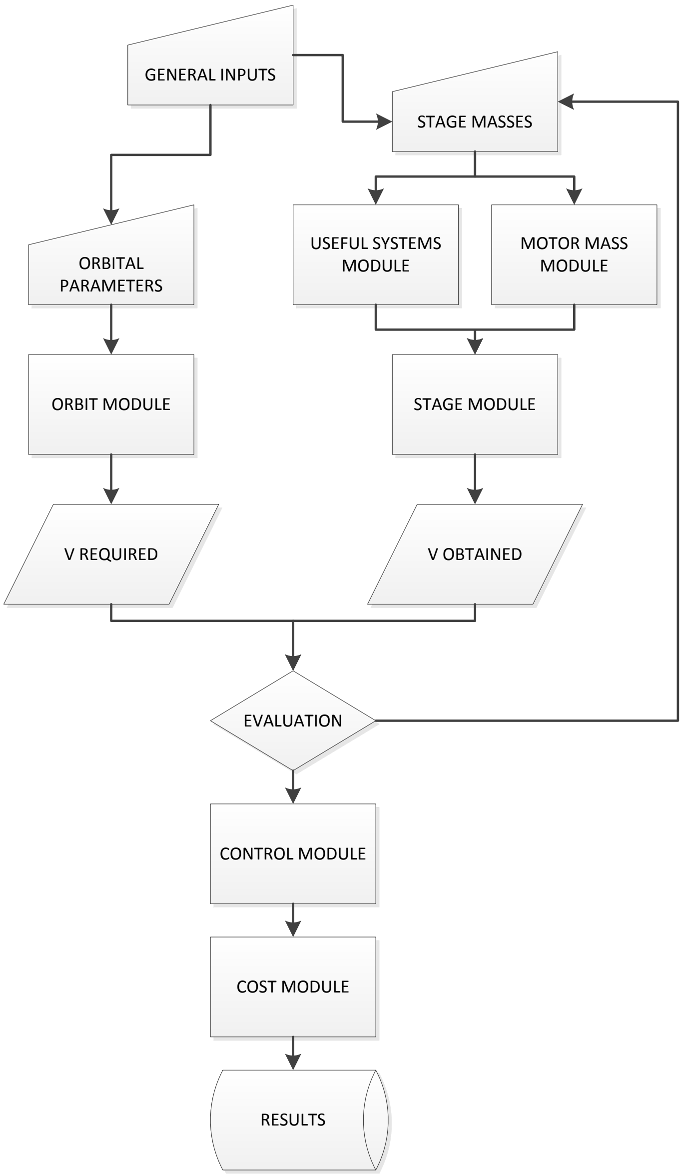

Figure 63 Architecture of the Prepared Solution

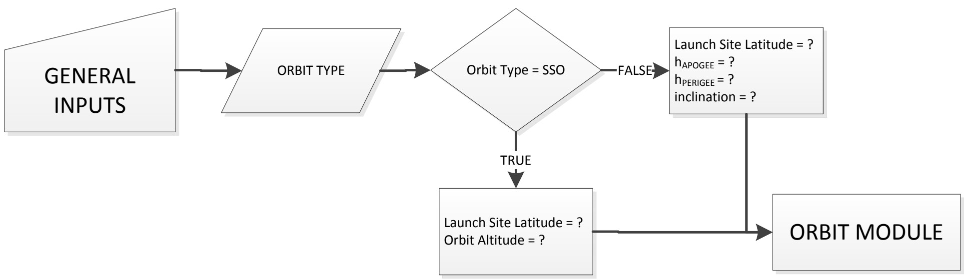

Figure 64 Orbital Parameters for the Operation of Orbit Module

Orbit Module

This module is used to calculate the velocity requirement of the launch vehicle depending on the type of the orbit (i.e. circular, elliptic and SSO), altitude of the orbit (mean altitude if circular and SSO, apogee and perigee heights if elliptic) and launch site latitude. It is important to determine $\Delta V_{required}$ as accurate as possible for realistic mass and corresponding cost estimations. Orbit velocity is the highest term in the velocity requirement. It is about 70% of the total while gravity losses and other losses explained previously are in the order of 15%. Only output of the orbit module, $\Delta V_{required}$, is used to evaluate the performance of the launch vehicle by comparing it to the obtained velocity, $\Delta V_{obtained}$.

Useful Systems Module

Useful systems in a launch vehicle includes secondary propulsion systems such as cold gas thrusters for attitude control, interstage structures, booster attachment units and avionics that are required for the operation of the launch vehicle. Although these systems are required for the launch vehicle operation, they do not contribute to the defined performance other than increasing the total weight of the vehicle. Weights calculated by the module are important for the determination of payload ratios and obtained velocity $\Delta V_{obtained}$. Inputs for this module are the number of stages and boosters, the propellant masses of the stages. After the first evaluation iteration for the determination of the total launch vehicle weight, useful systems module updates the weights of the related items. Outputs of this module are used to determine the stage masses on the launch vehicle.

Motor Mass Module

Depending on the propellant mass inputs and diameters of the stages and by considering the type of the engine case (i.e. metal or composite), this module is used to determine the structural masses of the motors. Determination of the motor structural mass relies on the previously developed stages. Structural mass fraction of solid rocket motors is strongly influenced by the type of the case material. Structural ratio of solid rocket motors with composite case is lower than the motors with metal cases. Propellant mass fractions of the existing solid propellant motors vary in between 0.70 and 0.94. Diameter of the motor also affects the mass fraction. Wall surface area to ratio of chamber volume decreases in small solid rocket motors. Outputs of this module is then used to determine the weight of the stages by adding the useful systems' weights and used to determine the calculation of launch vehicle performance $\Delta V_{obtained}$ by the stage module.

Stage Module

This module is used to calculate the launch vehicle performance. It is capable of determining the obtained velocity in a generic way by considering the number of stages and boosters in addition to their specific impulses and total burn times. Outputs of useful systems module and motor mass module are combined and used for stage dry masses. Separation time of the fairing is also considered during the calculations. Consideration of booster separation is included in the code as a separate stage. For the case of boosters and main stage are operating at the same time, average specific impulse is calculated using the average propellant flow rates and an average specific impulse. Each velocity increment calculated by the ideal rocket equation for the specific case is then added to acquire $\Delta V_{obtained}$. Performance of the launch vehicle is evaluated at the next step by comparing the ideal velocity increment that can be obtained by the launch vehicle in consideration and the required velocity for the target orbit.

Control Module

Control module checks the feasibility of the launch vehicle in consideration by interpreting some technical and geometric values such as total length of the launch vehicle, diameters of the stages and total lift-off weight. Number of boosters and their diameters are used to calculate the minimum main stage diameter to fit the boosters around it. If the required main stage diameter exceeds the maximum allowed first stage diameter, launch vehicle solution is not accepted. Similarly, launch vehicle solutions with upper stage diameters higher than a predefined value depending on lower stage diameters are also not accepted for the next step of the calculations. Only admitted solutions with suitable geometric properties are determined for the cost calculations in the next step.

Cost Module

Cost estimation for a launch vehicle project should be based on previous projects and past experience as explained previously. Since the new principle in launch vehicle design is to satisfy the mission requirements with the minimum cost, it is important to apply a cost model and determine the launch vehicle costs during the conceptual studies. TRANSCOST, which is launch vehicle dedicated cost model, is implemented in the cost module of the solution.

Only key cost items such as stages and boosters are considered in the module. Development cost of each item included in the launch vehicle is calculated in the module and added together to obtain the total development cost of the vehicle. For the determination of the development costs, dry masses of the stages are used. In addition, production costs for the launch vehicle is determined with the assumption of 100 identical units produced with an annual production rate of 10. Reference mass values used in the CERs are taken from the previous steps. Only DOC is included for the flight and operations costs due to the fact that IOC for a launch vehicle is only a small portion of the total cost, nearly the same for different concepts and so negligible.

B.4.2. Test Case: Three Stage Launch Vehicle with Serial Staging

Test case that will be explained is the solution for the three staged launch vehicle with serial alignment. Most of the launch vehicles in the market consist of three stages. Selected launch site location and target orbit in addition to the payload carrying capability will be given. Mass distributions of the obtained solutions in addition to the cost estimations will be explained and configurations will be compared for three staged launch vehicles.

Ideal Velocity Requirement

Target orbit is selected to be sun-synchronous with a mean altitude of 700 km for the test case. Inclination of this orbit can be calculated as 98.2^{o} using the Equation (2.8). Variation of inclination with respect to the increasing altitude is also given in Figure 17. Sun-synchronous orbits are common orbits in earth observation and reconnaissance purposes. Many of the satellites operating in LEO are on sun-synchronous inclinations.

Launch site for the test case is selected to be the same for all configurations as located at 36.25^{o} north latitude near south coast of Turkey. Launch site longitude does not have any effect on ideal velocity requirement due to the fact that drop zone restrictions are not considered in the study. Advantage of the selected launch site is that it is near the Mediterranean Sea and provides sufficient ground range of drop zones for separated lower stages. With a suitable selection of latitude, available drop zones on inhabited desert sections of Africa can be provided for upper stages in case of south directed launches for the proposed target orbit.

Payload capacity for the studied configurations is taken to be 1500 km which is the same as VEGA. Obtained solutions are small lift launch vehicles that are capable of carrying payload weights lower than 2000 kg. Launch mass of about 550 of the total of around 700 operational satellites in LEO at the end of 2015, December are less than 2000 kg and suitable to be launched by small lift launch vehicles. All of the configurations can be used to carry payloads which are micro to medium sized satellites as standalone missions to the target orbit.

Ideal velocity requirement of the target orbit considering the launch site location is calculated to be 10155.4 m/s by the prepared code under the previously explained assumptions. Items given in the velocity requirement is given in Equation (B.25). Corresponding values of the included items in $\Delta V_{required}$ is given in Table 30.

$$\Delta V_{required} = \Delta V_{orbit} + \Delta V_{rotation} + \Delta V_{gravity} + \Delta V_{other} \tag{B.25}$$

Table 30 Velocity Requirement of Launch Vehicle

Loss Term:	Explanation:	Value:
ΔV_{orbit}	Target Orbit Velocity	7504.3 [m/s]
$\Delta V_{gravity}$	Gravity Loss Assumption	1500.0 [m/s]
$\Delta V_{rotation}$	Earth's Rotation Effect	75.2 [m/s]
ΔV_{other}	Other Losses	1075.9 [m/s]
$\Delta V_{required}$	**Ideal Velocity Requirement**	**10155.4 [m/s]**

Mass distributions of the examined configurations are determined according to the velocity requirement. Three staged configurations will be explained in the following section.

Examined Configurations

In order to obtain a significant comparison between the alternatives, type of the motors used in the stages is fixed. First two stages of the alternatives assumed to be using HTPB based solid propellant motors while the last stages use a liquid propellant engine. The RD861K, which is using UDMH/NTO as propellant/oxidizer pair (Unsymmetrical Dimethyl Hydrazine ($(CH_3)_2NNH_2$) / Nitrogen Tetroxide (N_2O_4)), is selected for the upper stage engine. The RD861K engine is intended for thrust creation and flight control of LV third stage along the active leg of trajectory in pitch and yaw channels. Recently developed high performance upper stage of Cyclone-4 launch vehicle uses RD861K as main engine. Properties of the engine are given in Table 31. Average propellant plus oxidizer flow rate of the engine is around 24 kg/s. For a total operation of 370 s, 6270 kg of UDMH and 2600 kg of NTO are required which give a total of 8870 kg of propellant and oxidizer.

Table 31 Properties of RD861K (Yuzhnoye SDO)

Property:	Value:
Vacuum Thrust	77.63 [kN]
Vacuum Specific Impulse	330 [s]
Engine Mass	194 [kg]
Absolute Chamber Pressure	88.75 [bar]
Absolute Exit Pressure	29.4 [mbar]
Total Burn Time	370 [s]
Mixture Ratio	2.41 [-]
Number of Available Restarts	3 [-]

Four different alternatives are selected among the examined solutions for 3 staged launch vehicle configuration. In the first alternative, motors used in the first two stages are identical only with different specific impulse levels. Second alternative consists of a lower stage that is larger than the second stage while in the third alternative, second stage is larger. For the fourth and the last alternative, first stage includes three identical motors used as a cluster. Second stage of this alternative also uses the same motor with the first stage.

Considering the possibility of using higher expansion ratios on upper stages due to the decreasing atmospheric pressure, specific impulses of first and second stages are taken to be 274 s and 280 s respectively. When the existing solid propellant motors used in launch vehicles are examined, it can be seen that propellant mass ratios of upper stage motors can be as high as 91.0 to 93.0 % while this value is lower in first stages. Propellant mass ratio of used solid motors of first stages of the alternatives is taken to be 0.89 and mass ratio for second stage motors is taken to be 0.91. Although detailed analyses are required to determine the actual propellant mass ratios, assumed values can be considered valid for conceptual design studies.

Representative image of the alternatives is given in Figure 65 in order.

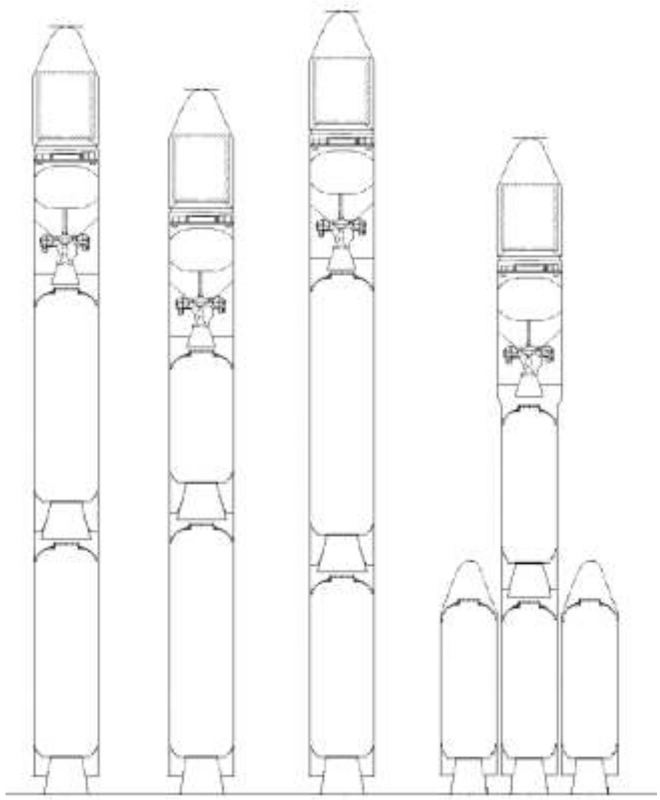

Figure 65 Three Stage Launch Vehicle Alternatives

System Properties

Obtained mass distributions for the alternatives by the prepared code are given in Table 32. Alternative 2, which includes two different solid rocket motors, has the minimum gross lift-off weight, while Alternative 3 has the maximum. Smaller stages obtained in the solution have small propellant mass ratios as expected due to the additional items such as thrust vectoring or possible telemetry equipment on the stages.

In the last row of Table 32, m_{PL}/GLOW ratio for the alternatives is given. This ratio can be considered as a performance variable as it shows the required total weight of the launch vehicle to fulfill its mission for unit payload. Since payload capacities of alternatives are taken to be the same as 1500 kg to 700 km SSO, maximum m_{PL}/GLOW ratio belongs to the Alternative 2 which has the minimum GLOW.

Table 32 Mass Distributions of Obtained Solutions

		Alternative 1	**Alternative 2**	**Alternative 3**	**Alternative 4**	
Stage 1	m_p	71850.0	81600.0	65500.0	103650.0	[kg]
	m_s	10264.3	11469.4	9479.5	16894.7	[kg]
	ε_1	0.875	0.877	0.874	0.860	[-]
	λ_1	1.118	0.786	1.414	0.421	[-]
	Isp	274				[s]
Stage 2	m_p	71850.0	54900.0	84800.0	34550.0	[kg]
	m_s	7519.0	5842.7	8799.8	3830.0	[kg]
	ε_2	0.905	0.904	0.906	0.900	[-]
	λ_2	0.157	0.205	0.133	0.324	[-]
	Isp	280				[s]
Stage 3	m_p	8870.0				[kg]
	m_s	1232.4				[kg]
	ε_3	0.878				[-]
	λ_3	0.165				[-]
	Isp	330				[s]
GLOW		**173905.7**	**166234.5**	**181001.7**	**171347.1**	[kg]
m_{PL}/GLOW		0.863	0.902	0.829	0.875	[%]

Weight of payload fairing is taken to be 650 kg in all alternatives. In addition, 170 kg of extra weight for payload adapter is also taken into account. These two items affect the vehicle performance directly and considered in the calculation of ideal velocity increment. Payload adapter, which is responsible of the interface between the satellite and the launch vehicle and separation of the satellite at the end of the mission, is carried by the launch vehicle until the end of the mission. On the contrary, payload fairing, which is responsible for protecting the satellite from environmental effects such as dynamic pressure of the flight and free molecular heating rate, is separated from the launch vehicle after a certain altitude. This provides improved payload capacities to the launch vehicle by decreasing the total weight to be accelerated to the orbital velocity.

Example ideal velocity increment $\Delta V_{OBTAINED}$ solution for Alternative 2 with corresponding stage masses is given in Table 33. Although obtained ΔV is 3.9 m/s higher than the required value, mission requirements in terms of velocity for all alternatives are ensured. This difference occurs due to the selected propellant step used in the calculations.

Table 33 Ideal ΔV Solution for Alternative 2

1500 kg - 700 km / 98.2°		
1st Stage	m_s [kg]	11469.4
	m_p [kg]	81600.0
	m_t [kg]	93069.4
	Isp [s]	274
2nd Stage	m_s [kg]	5842.7
	m_p [kg]	54900.0
	m_t [kg]	60742.7
	Isp [s]	280
3rd Stage	m_s [kg]	1232.4
	m_p [kg]	8870.0
	m_t [kg]	10102.4
	Isp [s]	330
	m_{pl} [kg]	1500
	m_{plf} [kg]	650
	$m_{ADAPTOR}$ [kg]	170

Ideal ΔV	
m_{i1} [kg]	166234.5
m_{f1} [kg]	84634.5
Isp_1 [s]	274
ΔV_1 [m/s]	1814.5
m_{i2} [kg]	73165.1
m_{f2} [kg]	18265.1
Isp_2 [s]	280
ΔV_2 [m/s]	3811.8
m_{i3} [kg]	11772.4
m_{f3} [kg]	2902.4
Isp_3 [s]	330.0
ΔV_3 [m/s]	4532.9
$\mathbf{\Delta V_{TOTAL}}$	**10159.3**

Distribution of provided velocity by stages to the payload is different for all of the alternatives due to the fact that proposed stages have different structural ratios and propellant mass distributions. Difference between the alternatives in terms of provided velocity increment of stages is given in Table 34. $\Delta V_{obtained}$ for alternatives which is given in the last row of the table is higher than the $\Delta V_{required}$ obtained by the Equation (5.1). Cost estimations of the alternatives are given in the following section.

Table 34 ΔV Distribution of Alternatives among Stages

	Alternative 1	**Alternative 2**	**Alternative 3**	**Alternative 4**
ΔV_1 [m/s]	1432.7	1814.5	1207.5	2496.2
ΔV_2 [m/s]	4193.6	3811.8	4418.5	3130.5
ΔV_3 [m/s]	4532.9			
ΔV_{TOTAL} [m/s]	10159.2	10159.3	10158.9	10159.6

Cost Estimations

Since cost estimations for development, production and operation costs of a LV is most important in Pre-Phase A, investigating the possible launch vehicle concept options and selection of the launch vehicle concept with the minimum cost, parametric cost model TRANSCOST is also included in the solution. Application of cost model is in a limited level and considers only the development and production of main items and operation costs for flight and mission operations and ground operations in addition to the cost of liquid propellants. Total costs for alternatives are given in Table 35 to Table 38.

Table 35 Total Cost for Alternative 1

	Alternative 1	Factors														
		f1	f2	f3	p	f4	f8	f10	f11	fv	fc	L	N	QN	M	Csost [Wyr]
Development Costs	Solid Propellant Rocket Motor (Single CER for Identical Stages)	1,00	-	0,90	-	-	-	-	-	-	-	-	-	-	8880	2189,4
	Liquid Propellant Engine (with Turbopump)	0,50	1,00	0,80	-	-	-	-	-	-	-	-	-	-	194	1388,9
	Liquid Propellant Stage (Additional Weights of All Items)	1,00	-	1,00	-	-	-	-	-	-	-	-	-	-	1038	781,0
Unit Production Costs	Production of Solid Propellant Stages (For both 1st and 2nd Stages)	-	-	-	0,95	0,96	0,97	0,90	1,00	-	-	-	-	-	8880	146,4
	Liquid Propellant Rocket Engine (Engine Only)	-	-	-	0,95	1,00	0,97	-	1,00	-	-	-	-	-	194	19,5
	Liquid Propellant Stage (Additional Weights of All Items)	-	-	-	0,95	1,00	0,97	0,90	1,00	-	-	-	-	-	1038	68,1
Operation Costs	Ground Operations	-	-	-	0,95	1,00	0,97	-	1,00	0,50	1,00	2,00	3,00	-	173,9	142,2
	Flight and Mission Operations	-	-	-	0,95	1,00	0,97	-	-	-	-	2,00	-	0,70	-	8,7
	Liquid Propellant Cost (UDMH)	3015800 $													3,3E-06	10,1
															Total Cost (In Wyr):	**4754,2**

Table 36 Total Cost for Alternative 2

	Alternative 2	Factors														
		f1	f2	f3	p	f4	f8	f10	f11	fv	fc	L	N	QN	M	Csost [Wyr]
Development Costs	Solid Propellant Rocket Motor (CER for 1st Stage Only)	1,00	-	0,90	-	-	-	-	-	-	-	-	-	-	10085	2363,1
	Solid Propellant Rocket Motor (Single CER for Identical Stages)	1,00		0,90											5430	1629,9
	Liquid Propellant Engine (with Turbopump)	0,50	1,00	0,80	-	-	-	-	-	-	-	-	-	-	194	1388,9
	Liquid Propellant Stage (Additional Weights of All Items)	1,00	-	1,00	-	-	-	-	-	-	-	-	-	-	1038	781,0
Unit Production Costs	Production of Solid Propellant Stage (Only for 1st Stage)	-	-	-	0,95	1,00	0,97	0,90	1,00	-	-	-	-	-	10085	80,2
	Production of Solid Propellant Stage (Only for 2nd Stage)				0,95	1,00	0,97	0,90	1,00						5430	62,6
	Liquid Propellant Rocket Engine (Engine Only)	-	-	-	0,95	1,00	0,97	-	1,00	-	-	-	-	-	194	19,5
	Liquid Propellant Stage (Additional Weights of All Items)	-	-	-	0,95	1,00	0,97	0,90	1,00	-	-	-	-	-	1038	68,1
Operation Costs	Ground Operations	-	-	-	0,95	1,00	0,97	-	1,00	0,50	1,00	2,00	3,00	-	166,2	137,9
	Flight and Mission Operations	-	-	-	0,95	1,00	0,97	-	-	-	-	2,00	-	0,70	-	8,7
	Liquid Propellant Cost (UDMH)	3015800 $													3,3E-06	10,1
															Total Cost (In Wyr):	**6550,0**

Table 37 Total Cost for Alternative 3

	Alternative 3	Factors														
		f1	f2	f3	p	f4	f8	f10	f11	fv	fc	L	N	QN	M	Csost [Wyr]
Development Costs	Solid Propellant Rocket Motor (CER for 1st Stage Only)	1,00	-	0,90	-	-	-	-	-	-	-	-	-	-	8095	2071,1
	Solid Propellant Rocket Motor (Single CER for Identical Stages)	1,00		0,90											8390	2116,1
	Liquid Propellant Engine (with Turbopump)	0,50	1,00	0,80	-	-	-	-	-	-	-	-	-	-	194	1388,9
	Liquid Propellant Stage (Additional Weights of All Items)	1,00	-	1,00	-	-	-	-	-	-	-	-	-	-	1038	781,0
Unit Production Costs	Production of Solid Propellant Stage (Only for 1st Stage)	-	-	-	0,95	1,00	0,97	0,90	1,00	-	-	-	-	-	8095	73,5
	Production of Solid Propellant Stage (Only for 2nd Stage)				0,95	1,00	0,97	0,90	1,00						8390	74,5
	Liquid Propellant Rocket Engine (Engine Only)	-	-	-	0,95	1,00	0,97	-	1,00	-	-	-	-	-	194	19,5
	Liquid Propellant Stage (Additional Weights of All Items)	-	-	-	0,95	1,00	0,97	0,90	1,00	-	-	-	-	-	1038	68,1
Operation Costs	Ground Operations	-	-	-	0,95	1,00	0,97	-	1,00	0,50	1,00	2,00	3,00	-	181	146,1
	Flight and Mission Operations	-	-	-	0,95	1,00	0,97	-	-	-	-	2,00	-	0,70	-	8,7
	Liquid Propellant Cost (UDMH)	3015800 $													3,3E-06	10,1
															Total Cost (In Wyr):	**6757,5**

Table 38 Total Cost for Alternative 4

	Alternative 4	Factors														
		f1	f2	f3	p	f4	f8	f10	f11	fv	fc	L	N	QN	M	Csost [Wyr]
Development Costs	Solid Propellant Rocket Motor (Single CER for Identical Stages)	1,00	-	0,90	-	-	-	-	-	-	-	-	-	-	4270	1411,0
	Liquid Propellant Engine (with Turbopump)	0,50	1,00	0,80	-	-	-	-	-	-	-	-	-	-	194	1388,9
	Liquid Propellant Stage (Additional Weights of All Items)	1,00	-	1,00	-	-	-	-	-	-	-	-	-	-	1038	781,0
Unit Production Costs	Production of Solid Propellant Stages (For both 1st and 2nd Stages)	-	-	-	0,95	0,92	0,97	0,90	1,00	-	-	-	-	-	4270	209,3
	Liquid Propellant Rocket Engine (Engine Only)	-	-	-	0,95	1,00	0,97	-	1,00	-	-	-	-	-	194	19,5
	Liquid Propellant Stage (Additional Weights of All Items)	-	-	-	0,95	1,00	0,97	0,90	1,00	-	-	-	-	-	1038	68,1
Operation Costs	Ground Operations	-	-	-	0,95	1,00	0,97	-	1,00	0,50	1,00	2,00	3,00	-	171,3	156,8
	Flight and Mission Operations	-	-	-	0,95	1,00	0,97	-	-	-	-	2,00	-	1,00	-	12,4
	Liquid Propellant Cost (UDMH)	3015800 $													3,3E-06	10,1
															Total Cost (In Wyr):	**4057,1**

Total costs of alternatives are determined under the assumption of two annual launches and total operation duration of 10 years which gives a total number of 20 launches. Values of the factors which are inputs for the cost estimation relationships are determined according to the current situation in Turkey.

Since the price of RD861K engine is not known, related development cost of the item is also included in the estimations. In addition, value of Wyr is estimated to be 300k $ to be able to include the propellant costs in the comparison. Comparison of the alternatives is given in Table 39.

Table 39 Comparison of Costs for Alternatives

	Alternative 1	Alternative 2	Alternative 3	Alternative 4
Development Costs	42.1 %	59.5 %	61.4 %	34.6 %
Unit Mission Cost	3.8 %	3.7 %	3.9 %	4.6 %
Unit Production Costs	*2.3 %*	*2.2 %*	*2.3 %*	*2.9 %*
Operation Costs	*1.6 %*	*1.5 %*	*1.6 %*	*1.7 %*
Project Completion Cost	49.7 %	67.0 %	69.1 %	43.8 %
Lifetime Cost	**80.2 %**	**96.9 %**	**100.0 %**	**80.5 %**

B.4.3. Discussions on Test Case

In this test case, four alternatives are selected among the solution for three staged launch vehicle and compared in terms of performance and cost. Mass distribution of alternatives is presented as system properties. Cost estimations which are obtained using the parametric cost model TRANSCOST are also given.

Solution with minimum gross lift-off weight belongs to Alternative 2 which also provides the maximum m_{PL}/GLOW ratio. Required solid propellant is 136.5 tons for this configuration while the worst configuration Alternative 3 requires 150.3 tons of solid propellant. Although Alternative 1 and Alternative 4 are similar in terms of GLOW, number of solid propellant engines and the complexity of the launch vehicle are higher in Alternative 4. Alternative 4 shows that smaller GLOW does not always require smaller structural masses even smaller solid propellant motors have smaller structural ratios.

Many of the factors are taken to be same among alternatives for the determination of the cost items and comparison of the alternatives. Cost reduction factor f_4 determined according to p = 0.95. f_4 is equal to 1.00 for solid propellant stages of Alternative 2 and 3 while 0.96 and 0.92 for Alternative 1 and 4, respectively. Producing similar units with higher production rate decreases the production cost of the item as expected. All of the alternative configurations have liquid propellant stages with storable propellants in addition to solid propellant stages. Considering the type of the vehicle f_v is selected to be equal to 0.5.

Minimum development cost belongs to the Alternative 4 independent of the GLOW of the launch vehicle due to the fact that similar solid propellant motors are used on 1st and 2nd stages. Unit production costs of first three alternatives are similar because of similar structural weights. Alternative 2 has the minimum GLOW and unit production cost at the same time which is also advantageous in terms of operations. Ground operations costs are directly related to the initial weight of the launch vehicle. Even unit mission costs for first three alternatives are similar and less than the last alternative, project completion cost of Alternative 4 is lower. Using higher number of motors only increases the complexity of the launch vehicle and so the ground operation cost.

It is also important to compare lifetime cost of launch vehicle projects. Lifetime cost includes the effect of total number of launches of the vehicle until the end of its use. Considering 10 launches for the alternatives, lifetime cost of Alternative 4 becomes higher than the Alternative 1 due to its higher unit mission cost. It is also important to note that launch vehicle with optimum weight, which provides the minimum gross lift-off weight, does not guarantee either the minimum development cost or the lifetime cost.

Best three staged launch vehicle configuration is Alternative 1 when related costs are also included in the determination of weight distribution for the concept selection.

Although optimum trajectories of the alternative configurations will be different, they will trace similar flight paths which have an important influence on space launch safety. Launch vehicle flight starts with a vertical phase. This phase is followed by a pitch over maneuver and a gravity turn to minimize atmospheric effects as soon as possible and to minimize aerodynamic loads on launch vehicle. In this case most important factor on space launch center safety remains as the total weight and the amount of propellant on board the launch vehicle.

When the alternative configurations are evaluated in terms of safety, lightest configuration Alternative 2 is the best candidate due to the advantage of its mass distribution. Alternative 3 is the least attractive configuration with its high weight and nontraditional weight distribution. Alternative 1 and Alternative 4 has similar amount of propellant and total weight. However, reliability is another important issue in safety and Alternative 4 should not be selected consequently.

Considering the more than 15 % lifetime cost difference between the best candidates Alternative 1 and Alternative 2, Alternative 1 is selected as the baseline for the numerical analyses of the effects of atmospheric parameters on space launch center safety.