OPTIMAL EXTERNAL CONFIGURATION DESIGN OF A MISSILE

ÇAĞATAY TANIL

JULY 2009

OPTIMAL EXTERNAL CONFIGURATION DESIGN OF MISSILES

A THESIS SUBMITTED TO
THE GRADUATE SCHOOL OF NATURAL AND APPLIED SCIENCE
OF
MIDDLE EAST TECHNICAL UNIVERSITY

BY

ÇAĞATAY TANIL

IN PARTIAL FULFILLMENT REQUIREMENTS
FOR
THE DEGREE OF MASTER OF SCIENCE
IN
MECHANICAL ENGINEERING

JULY 2009

Approval of the thesis:

OPTIMAL EXTERNAL CONFIGURATION DESIGN OF MISSILES

submitted by **ÇAĞATAY TANIL** in partial fulfillment of the requirements for the degree of **Master of Science in Mechanical Engineering Department, Middle East Technical University** by,

Prof. Dr. Canan Özgen
Dean, Graduate School of **Natural and Applied Sciences** _________________

Prof. Dr. Suha Oral
Head of Department, **Mechanical Engineering** _________________

Prof.Dr. Bülent E. Platin
Supervisor, **Mechanical Engineering Dept., METU** _________________

Dr. Gökmen Mahmutyazıcıoğlu
Co-supervisor, **TÜBİTAK-SAGE** _________________

Examining Committee Members

Prof. Dr. Kemal Özgören
Mechanical Engineering Dept., METU _________________

Prof. Dr. Bülent E. Platin
Mechanical Engineering Dept., METU _________________

Prof. Dr. Tuna Balkan
Mechanical Engineering Dept., METU _________________

Asst. Prof. Dr. Yiğit Yazıcıoğlu
Mechanical Engineering Dept., METU _________________

Prof. Dr. Kemal Leblebicioğlu
Electrical and Electronics Engineering Dept., METU _________________

Date: 22.07.2009

I hereby declare that all information in this document has been obtained and presented in accordance with academic rules and ethical conduct. I also declare that, as required by these rules and conduct, I have fully cited and referenced all material and results that are not original to this work.

Name, Last name : Çağatay Tanıl

Signature :

ABSTRACT

OPTIMAL EXTERNAL CONFIGURATION DESIGN OF MISSILES

Tanıl, Çağatay

M.Sc., Department of Mechanical Engineering

Supervisor : Prof. Dr. Bülent E. Platin

Co-Supervisor : Dr. Gökmen Mahmutyazıcıoğlu

July 2009, 136 pages

The main area of emphasis in this study is to investigate the methods and technology for aerodynamic configuration sizing of missiles and to develop a software platform in MATLAB® environment as a design tool which has an ability of optimizing the external configuration of missiles for a set of flight requirements specified by the user through a graphical user interface. A genetic algorithm based optimization tool is prepared by MATLAB is expected to help the designer to find out the best external geometry candidates in the conceptual design stage. Missile DATCOM software package is employed to predict the aerodynamic coefficients needed in finding the performance merits of a missile for each external geometry candidate by integrating its dynamic equations of motion.

Numerous external geometry candidates are rapidly eliminated according to objectives and constraints specified by designers, which provide necessary information in preliminary design. In this elimination, the external geometry candidates are graded according to their flight performances in order to discover an optimum solution.

In the conceptual design, the most important performance objectives related to the external geometry of a missile are range, speed, maneuverability, and control effectiveness. These objectives are directly related to the equations of motion of the missile, concluding that the speed and flight range are related to the total mass and the drag-to-lift ratio acting on missile. Also, maneuverability depends on the normal force acting on missile body and mass whereas the control effectiveness is affected by pitching moment and mass moment of inertia of missile. All of the flight performance data are obtained by running a two degree-of-freedom simulation.

In order to solve the resulting multi-objective optimization problem with a set of constraint of linear and nonlinear nature and in equality and inequality forms, genetic-algorithm-based methods are applied. Hybrid encoding methods in which the integer configuration variables (i.e., nose shape and control type) and real-valued geometrical dimension (i.e., diameter, length) parameters are encoded in the same individual chromosome.

An external configuration design tool (EXCON) is developed as a synthesis and external sizing tool for the subsonic cruise missiles. A graphical user interface (GUI), a flight simulator and optimization modules are embedded into the tool. A numerical example, the re-configuration problem of an anti-ship cruise missile Harpoon, is presented to demonstrate the accuracy and feasibility of the conceptual design tool. The optimum external geometries found for different penalty weights of penalty terms in the cost function are compared according to their constraint violations and launch mass values. By means of using EXCON, the launch mass original baseline Harpoon is reduced by approximately 30% without deteriorating the other flight performance characteristics of the original Harpoon.

Keywords: External Configuration, Conceptual Design, Flight Performance, Maneuverability, Control Effectiveness, Control Surface, Multi-objective Optimization, Cruise Missile, Penalty weight, Cost Function, Penalty Function

ÖZ

FÜZELERİN DIŞ GEOMETRİK KONFİGÜRASYONLARININ ENİYİLENMESİ

Tanıl, Çağatay
Yüksek Lisans, Makina Mühendisliği Bölümü
Tez Yöneticisi : Prof. Dr. Bülent E. Platin
Ortak Tez Yöneticisi : Dr. Gökmen Mahmutyazıcıoğlu

Temmuz 2009, 136 sayfa

Bu çalışmadaki temel amaç, füzelerin aerodinamik dış geometrik parametrelerini ölçülendiren yöntem ve teknolojilerin araştırılması ve, füze dış geometrisini kullanıcı arayüzüyle belirlenen uçuş gereksinimlerini karşılayacak şekilde eniyileyebilen bir tasarım aracının Matlab® ortamında geliştirilmesidir. Genetik algoritma tabanlı hazırlanan bu eniyileme aracının, kavramsal tasarım aşamasında, tasarımcının en iyi geometri adaylarını keşfetmesine yardımcı olması beklenmektedir. Geliştirilen yazılımda, her bir dış geometri adayına sahip füzenin performans başarım değerlerinin hareket denklemlerinin yardımıyla elde edilirken gereken aerodinamik katsayılar Missile DATCOM yazılımı aracılığıyla kestirilmektedir.

Kavramsal tasarım aşamasının başında, gereksinimleri sağlayacak füze konfigürasyon adaylarının sayısını azaltmak ve ön tasarım aşamasına yönelik bilgi edinmek amacıyla, yüksek sayıda konfigürasyon adayının hızlı bir şekilde irdelenip, elenmesi gerekmektedir. Bu elemede, en iyi geometriyi bulabilmek için, dış geometri adayları, başarım ölçütlerine göre puanlandırılmıştır.

Kavramsal tasarım aşamasında dış geometri oluşturulması açısından en temel başarım ölçütleri menzil, hız, manevra kabiliyeti ve kontrol etkinliğidir. Her bir ölçüt hareket denklemleriyle ilişkilendirildiğinde, menzil ve hızın, toplam kütle ve füze gövdesine etkiyen kaldırma kuvvetinin hava direncine oranıyla ilgili olduğu anlaşılmıştır. Ayrıca, manevra kabiliyetinin gövde üzerine etkiyen yanal kuvvet ve kütleyle, kontrol etkinliğinin ise yunuslama momenti ve eylemsizlikle bağıntılı olduğu görülmüştür. Tüm performans başarım verileri, iki serbestlik dereceli uçuş benzetiminin çalıştırılması ile elde edilmiştir.

Doğrusal ve doğrusal olmayan eşitsizlik ve eşitlik formlarındaki kısıtlara sahip bu çok amaçlı eniyileme problemini çözebilmek için genetik algoritma tabanlı yöntemler uygulanmıştır. Tek bir kromozom içerisinde, konfigurasyon parametreleri (örn., burun tipi ve kontrol tipi) için tamsayılar ile, geometrik uzunluk parametreleri (örn., çap ve boy) için gerçek sayılar ile kodlamaların uygulandığı karma şifreleme yöntemleri uygulanmıştır.

Ses hızı altında uçan füzeler için, dış geometri boyutlandırma ve sentez aracı olarak kullanılan bir yazılım (EXCON) geliştirilmiştir. Bu yazılımın içinde, kullanıcı grafik arayüzü, uçuş simülatörü ve eniyileme modülleri yer almaktadır. Geliştirilen kavramsal tasarım aracını doğrulamak amacıyla, sayısal bir örnek çalışma olarak, gemilere karşı atılabilen seyir füzesi Harpoon'un dış geometrisinin yeniden şekillendirilmesi problemi ele alınmıştır. Ceza fonksiyonunun başına farklı ceza ağırlıkları konularak oluşturulan maliyet fonksiyonlarına göre bulunan eniyi geometriler, kısıtları ihlal etmelerine ve fırlatma kütlesi değerlerine göre karşılaştırılmıştır. EXCON sayesinde, diğer uçuş başarım değerleri kötüleşmeksizin, özgün Harpoon füzesinin kütlesi yaklaşık olarak %30 düşürülebilmiştir.

Anahtar Kelimeler: Dış Geometri, Kavramsal Tasarım, Uçuş Başarımı, Manevra Kabiliyeti, Kontrol Etkinliği, Kontrol Yüzeyi, Çok Amaçlı Eniyileme, Seyir füzesi, Ceza Ağırlığı, Maliyet Fonksiyonu, Ceza fonksiyonu

To My Parents

ACKNOWLEDGMENTS

I would express my sincere thanks and appreciation to my supervisor Prof. Dr. Bülent E. Platin for his endless support, encouragement and insight during the preparation of this thesis and my co-supervisor Dr. Gökmen Mahmutyazıcıoğlu for his precious guidance and suggestions.

I would like to thank to my mother Lütfiye Tanıl and my father İsmail Tanıl for their endless love and understanding. Also my special thanks go to Gözde Doğan not only for encouraging me but also for her precious support and helpful suggestions.

My group coordinator Koray Dayanç is greatly acknowledged for his support and understanding. In addition, technical assistance and critical suggestions of Ms. Emel Mahmutyazıcıoğlu, Mr. Şamil Korkmaz, Mr. Emre Dede and Mr. Ali Karakoç are greatfully acknowledged.

This study is carried out at Defense Industries Research and Development Institute, Modelling and Simulation Division, TÜBİTAK - SAGE. The support provided by TÜBİTAK - SAGE is deeply acknowledged.

TABLE OF CONTENTS

LIST OF TABLES

TABLES

LIST OF FIGURES

FIGURES

LIST OF SYMBOLS

R	Range of the missile
C_n	Aerodynamic normal force coefficient
C_a	Aerodynamic axial force coefficient
$(C_n)_{\text{baseline}}$	Normal force coefficient of the baseline missile
$(C_a)_{\text{baseline}}$	Axial force coefficient of the baseline geometry
SM	Static margin
b_i	Regression coefficients
m	Instantaneous mass of the missile
m_0	Initial mass of the missile
t	Time elapsed during flight
$\dot{m}$	Mass flow rate of the propulsion system
C_G	Center of mass of the missile from nose tip
C_{G_0}	Initial center of mass location of the missile from nose tip
C_{G_f}	Final center of mass location of the missile from nose tip
u	Axial speed of the missile in the body fixed frame
F_x	Net force acting on missile in x-direction in the earth-fixed frame
w	Normal speed in the body axis of the missile in the body fixed frame
F_z	Net force acting on the missile in z-direction in the earth-fixed frame
$\overline{C}^{(b,e)}$	Transformation matrix from the body fixed to the earth fixed frame
θ	Flight path angle of the missile
$\dot{x}_e$	Missile speed in x-axis in the earth fixed frame
$\dot{z}_e$	Missile speed in z-axis in the earth fixed frame
T	Thrust force

M	Mach number
D	Drag force
ρ	Air density
V	Speed of the missile
C_D	Aerodynamic drag force coefficient
S	Reference area of the missile
L	Lift force
C_L	Aerodynamic lift force coefficient
I_{sp}	Specific impulse
g_0	Gravitational acceleration at the earth surface
m_F	Total fuel mass
η	Fuel to launch mass ratio
m_L	Launch mass of the missile
T^*	Nominal thrust value
α	Angle of attack
T_c	Control thrust command
K_P	Proportional gain of the controller
e	Error
u_r	Reference value
u	Actual value
α_T	Angle of attack at trim condition
W	Weight of the missile
α_C	Control angle of attack command
θ_r	Reference value of the flight path angle
n	Load factor
r	Radius of curvature
V_{AVG}	Average cruise speed of the missile
W_L	Launch weight of the missile
W_F	Fuel weight of the missile
$R_{\min}$	Minimum acceptable range of the missile

$R_{\max}$	Maximum acceptable range of the missile
ρ_M	Average density of the missile
V	Volume of the missile
l	Total length of the missile
d	Diameter of the missile
x_{CG}	Center of mass location of the missile
x_N	Center of mass location of the nose section
x_G	Center of mass location of the guidance section
x_W	Center of mass location of the warhead section
x_F	Center of mass location of the fuel tank section
x_M	Center of mass location of the motor section
m_N	Mass of the nose section
m_G	Mass of the guidance section
m_W	Mass of the warhead section
m_F	Mass of the fuel tank section
m_M	Mass of the motor section
m_{burnout}	Burnout mass of the missile
M^*	Pitch moment created by changing the center of mass of the missile
F_N	Normal force
C_m	Aerodynamic pitch moment force coefficient
X_{CG0}	Initial location of center of mass of missile
C_{m_α}	Derivative of pitch moment coefficient with respect to the angle of attack
δ	Fin deflection angle
n^u	Upper limit of load factor
n^l	Lower limit of load factor
${C_{m\alpha}}^l$	Lower limit of C_{m_α}
${C_{m\alpha}}^u$	Upper limit of C_{m_α}
BD	Body diameter of the missile

$(d_m)_{max}$	The largest diameter of the motor section
$S1$	Span of the first fin set of the missile
S^*	Maximum span of the missile from its tip to the tip of fins
LN	Nose length of the missile
BL	Main body length of the missile excluding nose section length
l^*	Maximum length of the missile
$\overline{x}$	Optimization parameter vector
$\overline{x}^u$	Upper bound vector of optimization parameters
$\overline{x}^l$	Lower bound vector of optimization parameters
$f(\overline{x})$	Cost function
$\overline{h}(\overline{x})$	Equality constraint function
$\overline{g}(\overline{x})$	Inequality constraint function
$f_i(\overline{x})$	Cost function of the i^{th} objective
$f_{i_{\min}}$	Minimum objective value of the i^{th} objective
$f_{i_{\max}}$	Maximum objective value of the i^{th} objective
f_i^*	i^{th} objective value of the baseline missile
w_i	Weighting of the i^{th} objective
G_i	Grade of the i^{th} objective
$f_{N,W}(\overline{x})$	Weighted and normalized objective function
S	Set of inequality constraints
$g_i(\overline{x})$	Inequality function of i^{th} constraint
p	Number of inequality constraints
$P(\overline{x})$	Penalty function
$B(\overline{x})$	Barrier function
$f_{N,W,P}(\overline{x})$	Weighted and normalized objective function including penalty term
λ	Penalty weight
$h_i(x)$	Equality function of i^{th} constraint
υ	Random weighting number for arithmetic cross over method
$\overline{x}_{C1}$	First children produced by the crossover operator

$\overline{x}_{C2}$	Second children produced by the crossover operator
CE	Control effectiveness
$n_{baseline}$	Load factor of the baseline missile
$SSM_{baseline}$	Static stability margin of the baseline missile
$R_{baseline}$	Range of the baseline missile
$V_{baseline}$	Speed of the baseline missile
S_{λ}	Set of Penalty Weights

CHAPTER 1

INTRODUCTION

1.1 Conceptual Design of the Missiles

Conceptual design is an explicit construction of ideas or concepts that a user needs to learn about what a product is, what it can do, and how it is intended to be used [14]. It also describes how a new product will work and meet the requirements at the beginning of a design process before starting its preliminary design. Therefore, the requirements of the system to be designed should be clearly stated and the methods for satisfying the design objectives need to be expressed in the conceptual design stage. To accomplish the design constraints and the requirements, the conceptual design involves some iterative processes, requiring a number of design iterations [8].

The conceptual design of missiles usually is a rapid analysis during which there are some constraints due to the pressure of time scheduling in completing this design stage, small number of staff, and uncertainties on the system requirements at the beginning. At the end of a conceptual design, its outputs are then served as a baseline to be used as the starting point for more detailed design stages.

The design of large and complex systems such as missiles requires making some appropriate compromises to achieve a balance among several, coupled objectives defined in terms of range, speed, weight, and production cost. The difficulty associated with conceptual design is that the relationships among design objectives and conceptual design parameters are often not well modeled or understood. This difficulty often results in an inefficient final design. In order to improve the outcomes of the conceptual design stage, a multidisciplinary design should be applied in this stage. The multidisciplinary design involves both analysis and synthesis in several disciplines concurrently to realize more effective solutions during design. Since a set of complex interrelations exist between mission requirements and constraints such as trajectory shaping, propulsion, weights, and aerodynamics, an appropriate optimization strategy should be applied in order to match conflicting goals [3].

The mission requirement synthesis in the conceptual design of missiles is an iterative process that requires the evaluation of alternative external geometry configurations and resizing the missile. Initial steps of the conceptual design stage are the mission definition and weapon requirements [8]. In the mission or scenario definition, an approximate trajectory should be specified first and then weapon requirements are to be determined in order the missile to follow this trajectory in a certain time.

The aerodynamic configuration sizing is defined as the process of searching the optimum geometric dimensions and configurations of a missile that satisfies the aerodynamic performance constraints and makes the objectives the best. It is carried out to improve the missile configuration and dimensions, which include its diameter, length, nose geometry, stabilizer as well as control surface size and geometry [8]. There are some analytical and simulation based methods which investigate the impact of sizing parameters on flight performance of missile.

1.2 Literature Survey

Eugene L. Fleeman stated that there are several major tasks of conceptual design which are mission definition, weapon requirements, sensitivity analyses, physical integration of the missile with the launch platform, weapon concept design synthesis, and technology assessment or development road map [8].

Since the initial process begins with a very general mission definition with some uncertainty, there could be several updates during the design process. The initial inputs are the requirements determined by the customer. These requirements should be evaluated with respect to the potential technology available. The requirements such as range, speed, maneuverability, or time-to-target are refined through computer modeling and simulation. The launch platform integration is also another task to handle, which determines the length, span, and mass of missile. The most iterative part of design is the synthesis of a missile in which its flight performance is determined and its geometry is resized in order to improve this flight performance. The iteration cycle of the synthesis of a missile is illustrated in Figure 1.1.

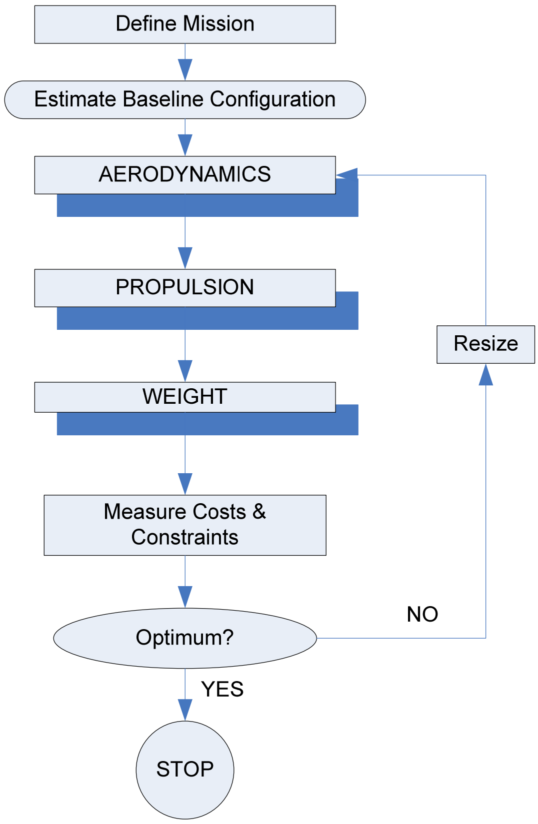
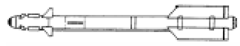
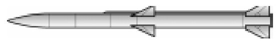
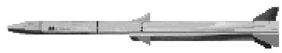
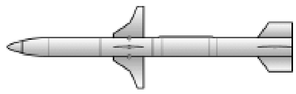
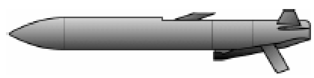
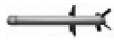
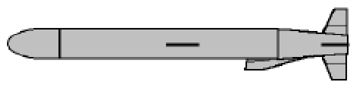
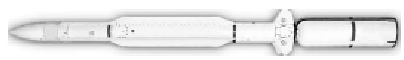


Figure 1.1 Iteration of Synthesis of a Missile [8]

In order to reduce the number of alternative solutions, the range of possibilities is reduced to a smaller set from a broad range in the conceptual design. Finally, a development map is determined for designing subsystems of the missile by using the available technologies.

Missiles can be categorized into four groups according to their launch type. In Table 1.1, some examples of typical air to air, air to surface, surface to surface and surface to air launched missiles are shown:

Table 1.1 Examples of Missiles According to Launch Type

Launch Type	Missile Name	Geometry
Air to Air	Short range ATA. AA-11.	
	Medium range ATA. AIM-120	
	Long Range ATA. Meteor	
Air to Surface	Short range ATS. AGM-114.	
	Medium range ATS. AGM-88.	
	Long range ATS. Storm Shadow	
Surface to Surface	Short range STS. Javelin.	
	Medium range STS. MGM-140.	
	Long range STS. BGM-109.	
Surface to Air	Short range STA. FIM-92.	
	Medium range STA. PAC-3.	
	Long range STA. SM-3.	

For air to air missiles, the most important objectives are maneuverability, range, and light weight. Also, the main drivers of air to surface missiles are accuracy, speed, modularity, versatility, and range. However, for the missiles shown in surface to surface part of Table 1.1, the aim is to obtain only range and modularity. In addition to them, altitude and accuracy are the factors which change the geometry in surface to air launch types [8].

Eugene L. Fleeman also criticized the impacts of configuration sizing variables on measures of performance of a missile. Their effects are graded according to their intensities in Table 1.2:

Table 1.2 Aerodynamic Configuration Sizing Effects on Weapon Requirements [8]

Aero Configuration Sizing Parameter	Impact on Weapon Requirement									
	Aero Measures of Merit			Other Measures of Merit						Constraint
	Weight	Range / Maneuver	Time to Target	Robust-ness	Lethality	Miss Distance	Observ-ables	Survivability	Cost	Launch Platform
Nose Fineness	●	●	●	○	●	●	●	●	◒	○
Diameter	●	●	●	○	●	◒	○	○	●	●
Length	●	●	◒	○	◒	◒	○	○	●	●
Wing Geometry / Size	●	●	◒	○	●	●	◒	◒	◒	●
Stabilizer Geometry / Size	●	●	○	○	●	●	◒	◒	◒	●
Flight Control Geometry / Size	●	●	○	○	●	●	◒	◒	◒	●
Propellant / Fuel	●	●	●	○	◒	◒	◒	●	◒	◒
Thrust Profile	●	●	●	○	◒	◒	●	●	◒	–
Flight Conditions (α, M, h)	●	●	●	●	●	●	●	●	●	○

● Very Strong ◒ Strong ○ Moderate – Relatively Low

As seen in Table 1.2, the flight conditions have a great impact on almost every performance measure implying that the configuration sizing design is strictly dependent on flight conditions. Besides, the aerodynamic sizing parameters also have a strong impact in the areas of lethality, miss distance, and cost.

The design of aircraft external configurations that have an acceptable flight performance is a complex multidisciplinary process. Therefore, some conceptual design computer tools were developed for rapid synthesis. One of the examples was developed for arbitrary body shaped missile configurations by B. K. Bennett [2]. The tool named LODST (Low Observables Design Synthesis Tool) uses analytical and semi-empirical methods to predict missile aerodynamics, mass properties and propulsion system performance characteristics. It has also a graphical user interface and other analysis modules around a common wireframe model of the configuration. Figure 1.2 shows the interactions between inputs, modeling, analysis and output modules of the LODST:

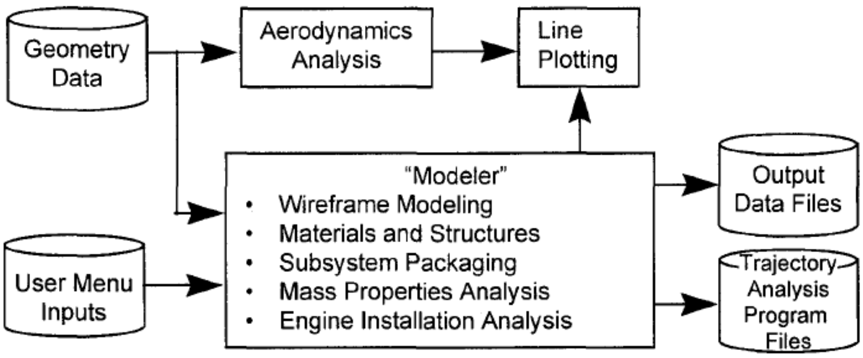


Figure 1.2 Low Observables Design Synthesis Tool Interconnectivity [2]

The LODST approach to vehicle design synthesis is based on a digital computer model that estimates the actual vehicle outer mold line and the major subsystems such as guidance unit, warhead, engine, and fuel. The model is combined with user inputs to compute mass properties, aerodynamic and installed engine performances of the missile. In addition, the LODST also provides output files for performance analysis by Trajectory Analysis Program (TRAP).

The GUI (Graphical User Interface) incorporates a variety of menus, buttons, and dialog boxes available from the X Windows System which executes on a UNIX workstation. The geometry model includes series of grid points connected by straight line segments representing missile outer mold line. The user is given an option to create infinite variety of completely arbitrary shapes. Subsystem inputs such as their weights and volumes are given by user. Internal package shapes can be modified into different shapes conforming to the outer mold lines shown in Figure 1.3. Each package is defined by shape, size, position, and mass including variable mass model for fuel tank.

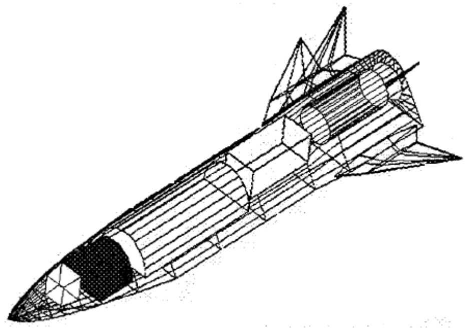


Figure 1.3 Typical Wireframe Geometry Model with Packages

The installed engine performance calculations are handled by user-supplied tables of engine properties and applying performance sensitivities. Also, the user selects materials of either the parts or sections via the wireframe modeler graphical user interface. The analysis is divided into two phases; namely, vehicle synthesis and aerodynamic predictions. The vehicle synthesis consists of structural sizing mass properties calculations and propulsion system computations whereas the aerodynamic analysis is performed by executing a program that receives wireframe model and outputs aerodynamic properties.

An aerodynamic analysis module called LODAA calculates all forces and moments for each individual part of missiles separately and then sum up their contributions. The total body normal force and side force are expressed analytically by using few methods which are SBT (Slender Body Theory) for linear terms and VFT (Viscous Cross Flow Theory) for nonlinear terms. The techniques used to predict the normal and side force also estimates the force distribution along the body. That helps expressing pitching and yawing moments. Note that SBT and VFT have some limitations due to the fact that they are assumed to be valid for missiles bodies having high fineness ratio noses, small boat tail angles, and low aspect ratio cross sections. The body axial force is assumed to be composed of four components which are potential force, pressure force, base and skin friction drag forces. These are calculated with the same algorithms implemented in Missile DATCOM which is an aerodynamic coefficient estimation tool developed by USAF (United States Air Force). The task of configuration synthesis is combining contributions of aerodynamics of each individual part for possible flight conditions and fin deflections.

Currently, a multidisciplinary sizing and synthesis program for the design of missiles is not commercially available as reported in the open literature [6]. However, there are several options for aircraft designers such as FLOPS. This tool performs the disciplines traditionally studied in aircraft design which includes aerodynamics and propulsion with trajectory analysis to size a vehicle. However, these programs are not adapted for missile design.

Another research on the conceptual design of high speed standoff missiles was performed in Georgia Tech's Aerospace System Design Laboratory (ASDL) by Tommer R. Ender, Erin K. McClure, and Dr. Dimitri N. Mavris [6]. Their objective was to create an environment that integrates disciplinary codes for conceptual sizing of a hypersonic missile.

At the beginning of design, they prepared a request of proposal (RFP) which served as the basis for customer requirements for the design. In RFP, a hierarchy of requirements including capability of striking a target between 500 and 1,500 km far within 5 to 15 minutes was clearly stated. Since the time to target and range are high priority requirements, a mission planner is only concerned with how long it would take a weapon system to reach its target destination. In addition, they defined the concept space for which several air-breathing propulsion, ducted rocket; liquid and solid fuel rockets baselines are examined. As a result, they concluded with the liquid fuel ramjet as the best fit propulsion model.

In ASDL, a modeling and simulation environment was also developed, whose inputs are the mission parameters and outputs are vehicle characteristics. In this phase, a series of multidisciplinary analyses were performed. Some of the disciplinary analyses were conducted using commercially available tools whereas others were performed in sizing and synthesis environment developed. The lists of platforms that are used in Georgia Tech's ASDL are shown in Table 1.3:

Table 1.3 Lists of Disciplinary Analysis Platforms

Analysis	**Platform(s)**
Inlet Analysis	MATLAB (Windows)
Propulsion	RAMSCRAM (UNIX)
Geometry Modeling	RAM (UNIX)
Aerodynamics	BDAP/AWAWE/SHABP (UNIX)
Trajectory and Sizing	MATLAB (Windows)
Structural Analysis	MATLAB (Windows)
Stability Analysis	MATLAB (Windows)

The interactions between various disciplinary analyses in sizing and synthesis environment are shown in Figure 1.4. The iterations in entire process are carried on until a convergence in dimensions is obtained.

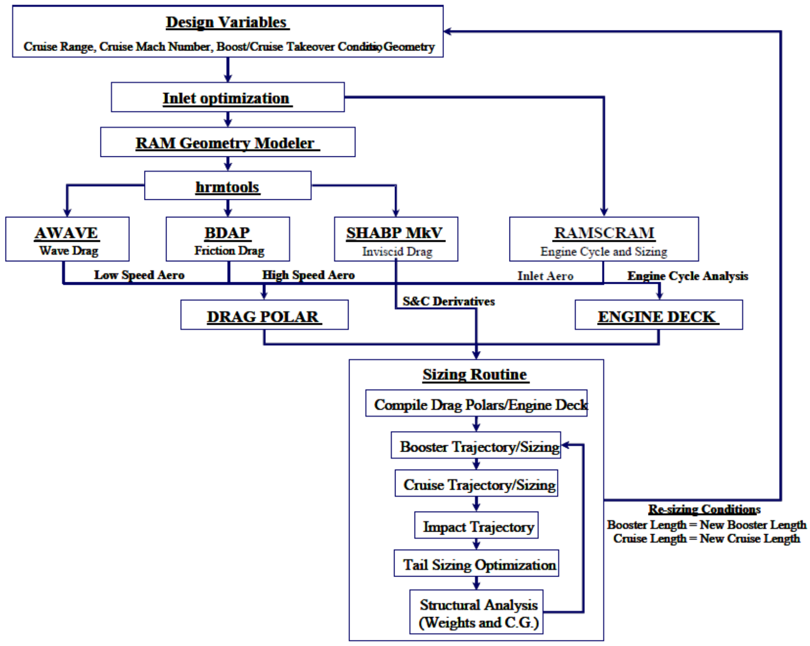


Figure 1.4 The Integrated Sizing and Synthesis Environment [6]

Once a sizing and synthesis environment was created, a design space is created. The design space investigation is for exploring which mission would be the best to design for missiles. Although there are numerous possible missions, it is impossible to run all of them in the simulation environment due to time limitations. As a result, a meta-model was created, which is based on statistical analysis of design inputs and response metrics. It is composed of simple equations in order to relate the independent variables to responses. In Georgia Tech's ASDL, the researchers preferred to use the following Response Surface Equation (RSE) for the meta-model:

$$RSE = b_0 + \sum_{i=1}^{n} b_i x_i + \sum_{i=1}^{n} b_{ii} x_i^2 + \sum_{i=1}^{n-1} \sum_{j=i+1}^{n} b_{ij} x_i x_j \qquad (1.1)$$

In RSE, n represents the number of factors, x's are the design variables, and b's represent the regression coefficients, which are determined by regressing sample data points.

The Design of Experiments (DoE) is defined as all information gathering exercises where variation is present [3]. In the response surface methodology, a predetermined DoE selected the specific cases of input variables. Although DoE provides sufficient information for creating RSE, it minimizes the number of data points at the same time. The response surface method is used for two different purposes in design. It can be used for optimizing design variables based on relative weightings as well as predicting the effects of design variables on each response which allows a sensitivity analysis at the end. In Georgia Tech's ASDL, a software package named JMP was used to generate DoE and regress the corresponding data into RSE's. As a final stage of the conceptual design, a feasibility analysis was performed by investigating constraints in the design space. By means of the meta-model, constraints on each response can be visualized two dimensionally, which helps the designers to explore the infeasible regions on design space.

The designed environment developed in Georgia Tech's ASDL was based only on supersonic missiles. Hence, subsonic mid-range cruise missiles cannot be handled by this tool.

Other than the response surface method, some genetic algorithm (GA) based search techniques are also studied for conceptual design. GA performs the effective and robust evolutionary search which can deal with the continuous, integer and discontinuous variables. As a numerical example, the design of a two-member frame and air intercept missile (AIM) design optimization problem was presented to demonstrate the accuracy and feasibility of the process developed by Nhu-Van Nguyen, Kwon-Su Jeon, Jae-Woo Lee and Yung-Hwan Byun [13]. The wing parameters of AIM 7 were reconfigured by considering its effect on the total lift, drag, and range of missile. Also, the fineness ratio of missile was involved into the design variables. The range was taken as an objective to be maximized with the three constraints. Static margin (SM), normal (C_n) and axial (C_a) coefficients were involved in inequality constraints whose upper and lower limits are taken according to the baseline missile, AIM 7. The optimization problem was in the form of:

$$\left.\begin{array}{ll}\text{Maximize} & \text{The range R} \\ \text{Subjected to} & C_n \geq (C_n)_{\text{baseline}} \\ & C_a \leq (C_a)_{\text{baseline}} \\ & SM \leq (SM)_{\text{baseline}}\end{array}\right\} \quad (1.2)$$

In addition, the hybrid algorithm was considered as the optimizer that shows robust and efficient evolutionary characteristics in its searching ability, and a gradient based method was run once the GA terminates helping in further savings in the computation searching time. In Figure 1.5, the difference between the optimum wing found and the baseline AIM 7 is illustrated. By means of optimization on wing parameters, it was achieved that the range of AIM 7 was increased from 12,632 to 16,738 meters.

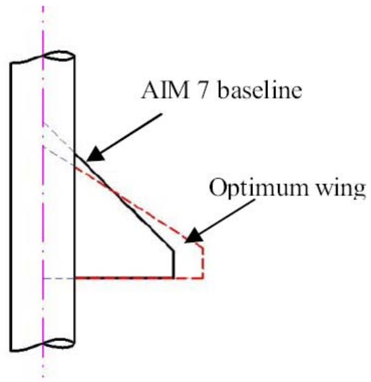


Figure 1.5 Optimum AIM Wing Configuration [13]

In the case study of by Nhu-Van Nguyen, Kwon-Su Jeon, Jae-Woo Lee and Yung-Hwan Byun, only the effect of wing dimensions on the total range of the missile was investigated. That is, other parameters like diameter, total length and wing location on the missile body could not be studied by this method since they were taken same as the original baseline missile.

In her thesis study, S. Aytar-Ortaç developed a methodology for obtaining an optimum external configuration during conceptual design of unguided missiles to satisfy the defined mission requirements [30]. Maximum range, minimum dispersion and maximum warhead effectiveness were selected as the mission requirements of this study. These requirement functions were computed for artillery shell configurations and the optimum geometry was found through both classical methods and genetic algorithm (GA) techniques.

By using equations of motion, it was impossible to obtain an analytical cost function as a function of geometric parameters only in her study. Instead, six degrees of freedom flight simulations were performed. FMCAD which is Flight Mechanics Computer Aided Design software developed by TÜBİTAK-SAGE, was used for solid modeling of different configurations and handling flight simulations. A regression analysis was performed on the simulation results and an overall cost function was achieved. The cost function was non-dimensionalized before proceeding to solution with GA. In her thesis, this non-dimensionalization was obtained by dividing each term of the cost function by the values obtained from the classical methods; namely, Conjugate Gradient, Quasi Newton. This resulted a range of [0, 1] for all objective function components, so that they can be compared.

In the case studies of the thesis of S. Aytar-Ortaç, the external geometry of artillery optimization problem was solved three times for each single objective one by one. The resulting optimum configurations found for only maximum range, only minimum dispersion and only maximum volume (warhead effectiveness) are shown in Figure 1.6.

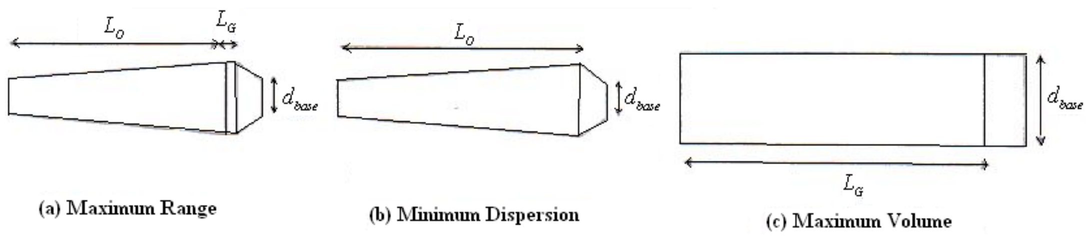


Figure 1.6 Single Objective Optimum Configurations for Artillery Shell

In order to compare the single objective optima with the multi objective optima, a multi-objective optimization is conducted with four different weighting factor sets.

These four different weighting factor sets represent four different mission requirements. The results of optimum external geometries of artillery shell for four different weighting sets (w_1, w_2, w_3, w_4) are illustrated in Figure 1.7:

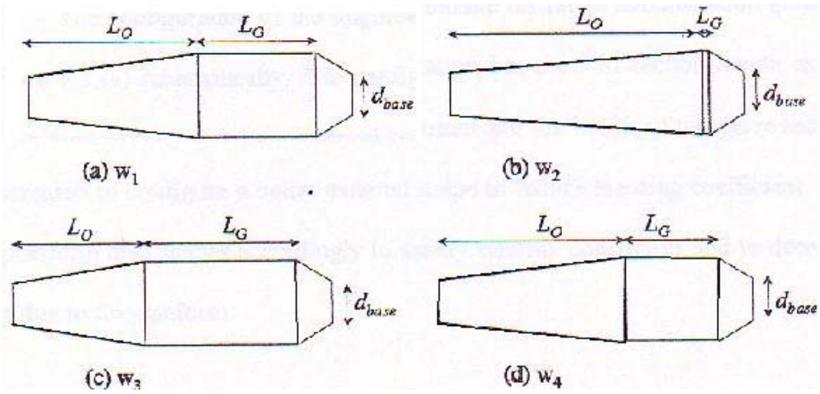


Figure 1.7 Optimum Configurations for Multi-Objective Optimization

It was observed that the resulting optimum geometries found by using different set of weightings were the combinations of the optimum geometries found by single objective optimization. That is, the geometric parameters (L_o, L_G and d_{base}) of the missiles found from the multi-objective optimization remained between the limits found from the single objective optimization. One can conclude from the case studies in the thesis of S. Aytar-Ortaç that the optimum geometries vary from one set of weightings to another.

Since only the unguided missiles were included in the scope of the thesis of S. Aytar-Ortaç, it is impossible to analyze the performances of guided missiles in such mission trajectories like cruise, climb phases.

In order to develop a conceptual design tool, McDonnell Douglas Corporation (MDC) and the NASA Langley Research Center (LaRC) have together embarked on an effort to develop a workstation-based tool to perform design synthesis, performance analysis, and optimization of hypersonic air breathing vehicles [31]. The methodology is intended to address the complex aero/propulsion integration issues which are characteristic of hypersonic vehicle design. The foundation of this tool has been built upon methods which have been developed independently by MDC and by LaRC.

The model developed by MDC and LaRC computes the vehicle performance between discrete trajectory flight conditions (Mach and altitude) defined by the user. A simple energy formula was used for calculating the propellant mass required to deliver the missile between two points on a trajectory. The Figure 1.8 shows the iteration cycle of the automated vehicle closure process.

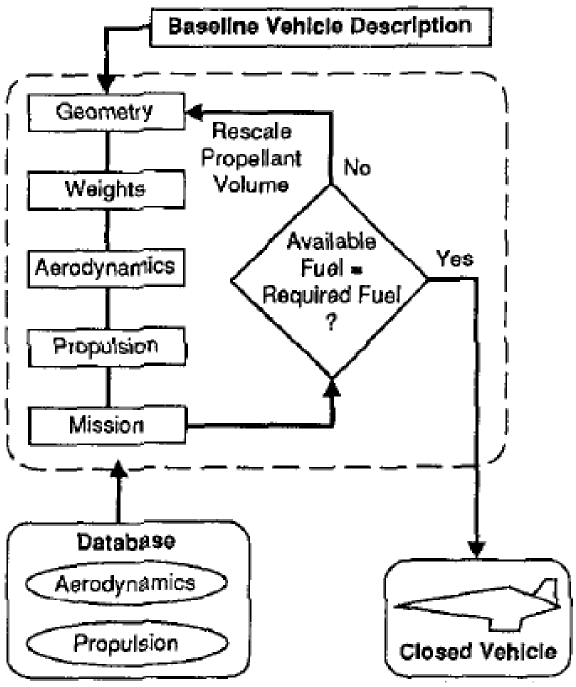


Figure 1.8 Automated Vehicle Closure Process

The iteration cycle shown in Figure 1.8 is continued until the available fuel becomes equal to the required propellant mass for missile to fly between pre-defined points in the trajectory. Since the critical design driver in the optimization was the capability of the missile to ascent between trajectory points, the problem was only limited into energy increase (acceleration) problem. Therefore, other objectives like mass minimization, maneuverability were not focused in the automatic vehicle closure process developed by MDC and LaRC.

1.3 Objectives of the Thesis

The objective of this thesis is to develop a software tool which provides automation in the conceptual design stage of missiles. The conceptual design tool has a scope of designing a mid-range, subsonic cruise missiles with a turbojet or solid fuel rocket motor. The main purpose of implementing this tool is to minimize the time spent during the conceptual design of missiles. Rapid elimination of undesirable external configurations by this software tool shortens the total design period. As seen in Figure 1.9, the conceptual design tool should be capable of performing three main subtasks.

1. A platform called as the "user interface" is needed in which both flight mission information and required weapon skills are to be defined as an input by designers.
2. There must be a "flight simulation module" by means that flight performance data of each external configuration candidate can be obtained.
3. A mechanism should be included for evaluating and grading each geometry candidates according to their flight performance results. In this mechanism, the use of a genetic-algorithm-based optimization technique is intended.

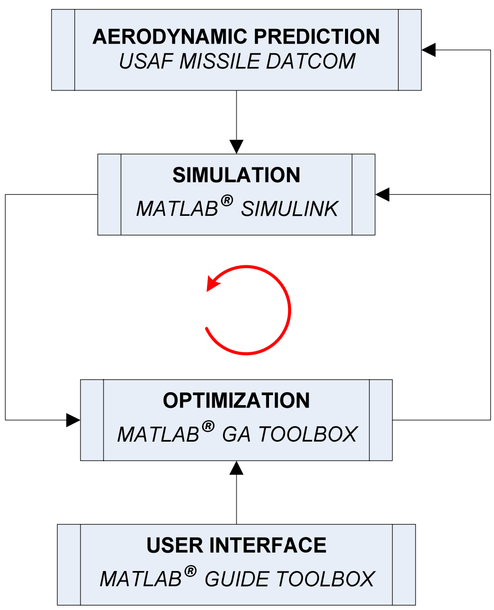


Figure 1.9 Conceptual Design Tool

The aerodynamic coefficients are required to be produced for each flight simulation. Since a large number of geometry candidates are expected to be graded according to the simulation results, the production of aerodynamic coefficients must be very fast considering the total run time of the tool. For this purpose, the use of USAF Missile DATCOM software is intended as a fast prediction tool for aerodynamic coefficients. However, this tool is more effective in subsonic regions since the experimental data used in DATCOM is based on subsonic flights. Therefore, the scope of the thesis will be limited to the design of missiles which have subsonic speeds. Also, in this thesis, the main area of interest for launch type is air-to-surface or surface-to-surface mission profiles.

In addition, the user is requested to select the trajectory profile according to mission information. Then, an optimization will be handled according to this trajectory constraint which turns the problem into a trajectory-dependent optimization of missile external geometry. In other words, the resultant best external geometry parameters will only be valid for a given trajectory at the beginning of process. By means of that, the search domain of interest can be shrunk since other possible trajectories are directly eliminated at the beginning.

1.4 Scope of the Thesis

A background introduction to the thesis topic is given in Chapter 1, including the objectives of the study with a survey of literature. The sequence of the processes that are followed in the conceptual design of missiles is defined by giving some example studies. In addition, the methods and assumptions as well as the software platforms to be used in the thesis are explained in the objectives section.

Chapter 2 presents the definition of the flight model and equations of the motions as well as the sub-models to be used in the dynamic model. These sub-models which are aerodynamics, propulsion, auto-pilot, atmosphere, and gravity are integrated and they handle the flight simulation in order to obtain flight performance data.

In Chapter 3, interactions between models mentioned in Chapter 2 are covered by showing the process cycle in the simulation. Also, possible combinations of flight trajectory phases that the simulator can handle are explained.

In Chapter 4, the variables which affect the flight performance of a missile are defined and listed into two groups; namely, shaping (configuration) and sizing (dimension) parameters. Besides, the flight performance measures of merit are explained as well as the platform and structural constraints on the external geometry of a missile are mentioned.

The optimization method used in this study is mainly addressed in Chapter 5. The external geometry optimization problem is defined by giving equations for the cost and constraints. The reasons of choosing genetic algorithm as a search method are justified. Also, the techniques of converting a constrained problem to an unconstrained problem are explained. Modifications in genetic algorithm operators which are mutation, crossover, and creation are explained as well as giving encoding method for this specific problem of external configuration sizing.

In Chapter 6, the implementation platforms and software packages used for preparing the conceptual design tool are expressed with the interactions between one another. The abilities of this tool are mentioned as well as representing the properties of the user interface.

Chapter 7 is a case study chapter in which the Harpoon AGM-84 is re-designed in the conceptual design tool prepared in this thesis. The final geometry obtained in this study is compared with the original one. In addition, the effects of search algorithm parameters on the resultant external geometries are investigated by a sensitivity analysis.

Chapter 8 is the final chapter of the thesis in which a summary of the study is provided, the conclusion of the study is stated by explaining the beneficial sides of using a conceptual design tool, and some recommendations for future studies are given.

CHAPTER 2

DYNAMIC MODEL

2.1 Definition of the Model

A dynamic model is used to express and model the behavior of a system over time [12]. In missile systems, a dynamic model represents the relation between the missile motion and the forces acting on missile body. In the conceptual design, there is no need to a high degree of freedom dynamic model since the external geometry of the missile is uncertain. After a baseline geometry is obtained by using simple models, more detailed dynamic models can be used for detailed analysis of motion. Increasing the degree of freedom means having more complex model which is unnecessary in the conceptual design stage.

By means of a simplified dynamic model of missiles, all necessary flight performance data needed for the optimization should be obtained. Therefore, the degree of freedom of the model is decided so that all the flight motions which affect the flight performance of the missile should be observed from the trajectory obtained from this model. Within the flight performance parameters, range and pull-up/down maneuverability could be obtained by at least 2 degree-of-freedom (DOF) point-mass models in vertical plane. However, there is no need to include third freedom (pitch rotation) in order to observe the control effectiveness which is ratio of fin surface deflections to angle-of-attack. It is possible to calculate the control effectiveness by using aerodynamic coefficients.

Eventually, one comes up with a 2 DOF model to represent the flight of the missile at this stage. Two degrees of freedom are the altitude and range variations of the missile body. That is, the flight motion is modeled in vertical plane having two translational freedoms which is shown in Figure 2.1:

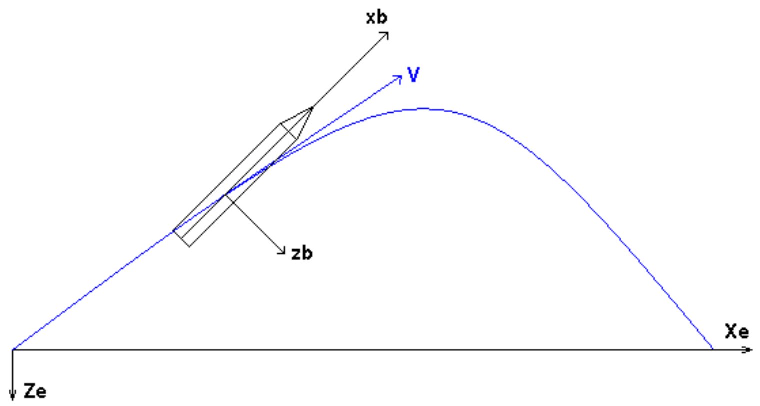


Figure 2.1 Two Degree of Freedom Model

2.2 Equations of Motions

Since the roll and yaw motions are excluded in the model, the trajectory of the model can be considered as the vertical planar motion against gravity shown in Figure 2.2:

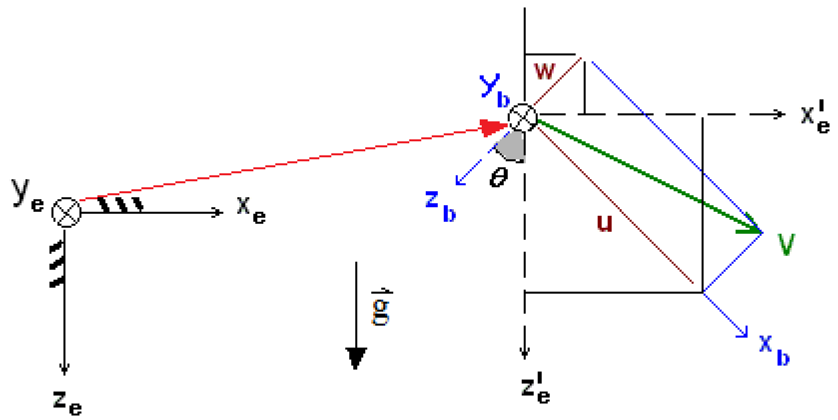


Figure 2.2 Velocity Vector in the Earth and Body Frames

There are two reference frames are used to describe the motion of the missile, which are both right handed and orthogonal. One of these reference frames is fixed to the earth and the other to the center of mass of the missile's body. The earth fixed reference frame $OX_eY_eZ_e$ can be assumed to be inertial because the range of the missile is short compared to the radius of the earth and the motion of the missile is much faster compared to earth motion. Here, X_e axis points towards north, Z_e axis points downwards to earth's center, and the Y_e axis is the complementing orthogonal axis found by the right hand rule. The body fixed reference frame $O_bX_bY_bZ_b$ has its origin at the missile's center of mass. X_b axis points towards nose of the missile, Z_e axis points downwards from the missile body.

On the other hand, the mass of the missile is decreasing as long as a propulsion system works in the system. It implies a variable inertia and a variable center of mass. The instantaneous mass m is calculated according to this propulsion model as

$$m = m_0 - \dot{m} \cdot t \tag{2.1}$$

where m_0 is the initial total mass, $\dot{m}$ is the rate of fuel burn, and t is the time.

The instantaneous location of the center of mass C_G of the missile during flight can also be computed as

$$C_G = C_{G_0} - \dot{C}_G \cdot t \tag{2.2}$$

where the constant time rate of change $\dot{C}_G$ of the location of the center of mass of the missile is expressed as

$$\dot{C}_G = \frac{C_{G_0} - C_{G_f}}{m_0 - m_f} \dot{m} \tag{2.3}$$

where C_{G_f} is the final location of the center of mass of the missile.

The velocity vector of the missile is represented in the body fixed frame as well as in the earth fixed frame as shown in Figure 2.2. The acceleration components $\dot{u}$ and $\dot{w}$

of the missile in body fixed frame can be written in terms of net forces F_x and F_z acting on missile center of mass, mass m of the missile from the Newton's second law of motion:

$$\dot{u} = \frac{F_x}{m} - \frac{\dot{m} \cdot u}{m} \tag{2.4}$$

$$\dot{w} = \frac{F_z}{m} - \frac{\dot{m} \cdot w}{m} \tag{2.5}$$

Using the transformation matrix $\overline{C}^{(e,b)}$, which transform a vector from the body fixed frame to the earth fixed frame

$$\overline{C}^{(e,b)} = \begin{bmatrix} \cos\theta & \sin\theta \\ -\sin\theta & \cos\theta \end{bmatrix} \tag{2.6}$$

The representation of missile's velocity components $\dot{x}_e$ and $\dot{z}_e$ in the earth fixed frame can be written as

$$\begin{bmatrix} \dot{x}_e \\ \dot{z}_e \end{bmatrix} = \overline{C}^{(e,b)} \begin{bmatrix} u \\ w \end{bmatrix} \tag{2.7}$$

2.3 Subsystems of Dynamic Model

2.3.1 Aerodynamics Model

An aerodynamic model is required in order to find out aerodynamic forces acting on missile which play a critical role while determining the geometric dimensions of the main body and the flat surfaces. The Missile DATCOM software is used as a semi-empirical aerodynamic coefficients estimator tool of obtaining aerodynamic coefficient and their derivatives since its run time is low and it gives outputs quickly [4]. During the simulation of flight, the aerodynamics data is to be read from a look-up table which is prepared before the simulation by Missile DATCOM for the external geometry specified. That is, in every time step of the flight simulation, the DATCOM is executed to obtain necessary aerodynamic coefficients in order to compute aerodynamic forces (lift and drag).

As seen in Figure 2.3, there is an information flow between DATCOM and other tools some of which are the inputs and others are the outputs of DATCOM. Therefore, before a flight simulation starts, the external configuration parameters and geometric dimensions of the missile body should be fed to DATCOM.

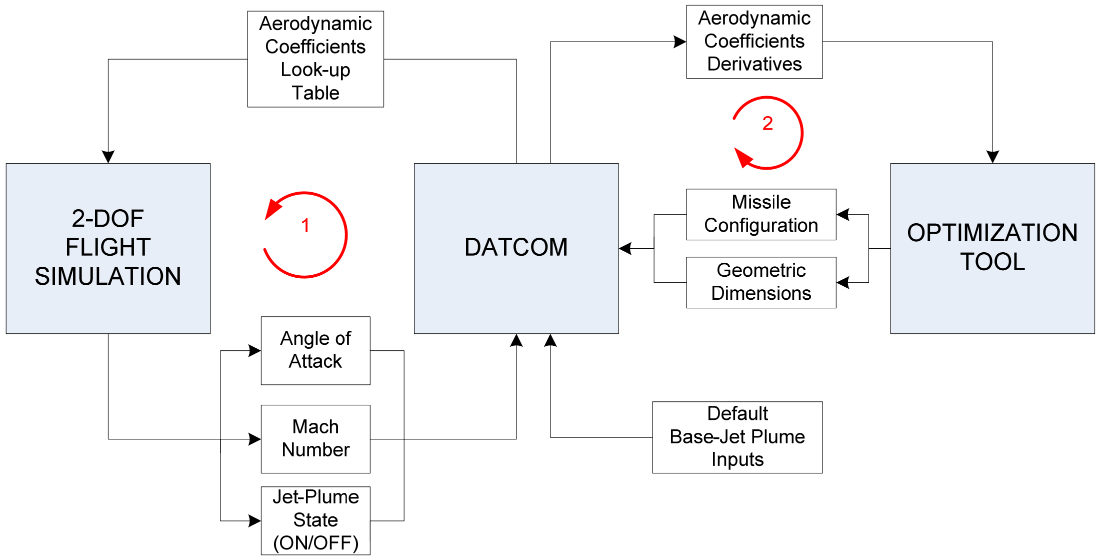


Figure 2.3 Aerodynamic Data Flow

In addition to the inputs which should be given to DATCOM, base-jet plume values must be supplied to DATCOM. In solid fuel rocket motors, the jet-plume effect occurred at base of missile affects the total drag force especially in the boost phase. The necessary conditions that DATCOM calculates the aerodynamic coefficients by considering the jet plume effect are given below [4].

$$T \neq 0 \tag{2.8}$$

$$M \geq 1.2 \tag{2.9}$$

In the above conditions, M is the Mach number of the missile and T is the thrust force given to the missile. The jet plume effect is either enabled or disabled according to the flight conditions and the engine chosen for a specific mission. For instance, in the glide phase, the jet-plume status is disabled since there is no thrust whereas in other phases it depends on the Mach number. Since most of the missiles with turbo-jet engines cannot exceed subsonic region, the effect of jet-plume can be automatically eliminated for the turbo-jet engine alternatives.

The interval of the angle of attack and Mach number should be supplied to DATCOM by the user as an input. The DATCOM produces a look-up table in which the aerodynamic coefficients C_D, C_L, and C_m and their derivatives C_{m_α} and C_{m_δ} can be interpolated. These coefficients are used for computing aerodynamic forces in flight simulation. The drag force D, lift force L and the pitch moment are calculated as [1]:

$$D = 0.5\rho V^2 C_D S \tag{2.10}$$

$$L = 0.5\rho V^2 C_L S \tag{2.11}$$

The aerodynamic coefficient derivatives which are another output of the DATCOM are needed for observing some of the flight performance data, for instance, control effectiveness and stability. Control effectiveness and stability are represented by C_{m_α} and $C_{m_\delta} / C_{m_\alpha}$ respectively. The C_{m_α} is the derivative of pitch moment coefficient with respect to angle of attack α whereas C_{m_δ} is defined as the derivative of the pitch moment coefficient with respect to fin deflection angle δ.

2.3.2 Propulsion Model

A propulsion model is included to the system in order to overcome the drag force or accelerate the missile during the flight. At the end of the conceptual design, the designer should have an idea about the thrust-time profile of the optimum external configuration. There are mainly two propulsion types modeled which are solid-fuel rocket and turbo-jet engine.

2.3.2.1 Solid Fuel Rocket Model

Solid fuel rocket engines are capable of providing thrust across the entire Mach number range no matter what the flight altitude is. Although the specific impulses of rockets are relatively lower than that of turbojet engines, they have an advantage of having higher acceleration capability than air-breathing propulsion. Also, they can be operated in higher altitudes. The disadvantage of solid fuel rockets is that the thrust control is mostly not used in them except ramjet motors. It implies that the speed of the missile is not controlled. There are two thrust levels of rocket engines which are called as the boost and sustain phases [8].

In Figure 2.4, the thrust time characteristic of a single-phase rocket motor which provides only boost during the beginning of flight for a certain time is shown. The motor is operated at its maximum power until the fuel is consumed. This alternative is suitable only for some specific flight trajectory profiles which are surface to surface launch types like climb-glide sequence.

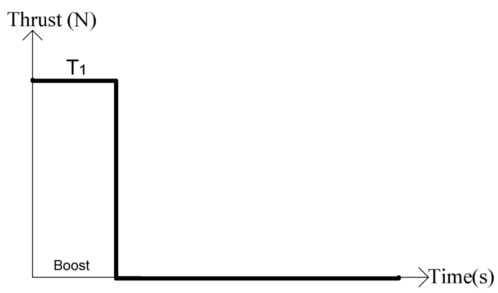


Figure 2.4 Single phase rocket: Boost

In Figure 2.5, the thrust time characteristic of single-phase rocket motor which provides only sustain during flight is shown. The motor is operated at low level which enhances the endurance of fuel during flight. It is suitable especially for flight profiles including cruise phase, for instance, glide-cruise-glide phase sequence.

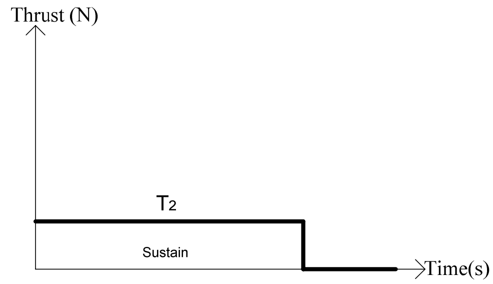


Figure 2.5 Single phase rocket: Sustain

In Figure 2.6, the thrust time profile of a hybrid-phase rocket motor which can provide two different levels of thrust, boost and sustain. This profile fits best for the flights which is the surface to surface launch type including cruise phase (Climb-Cruise-Glide).

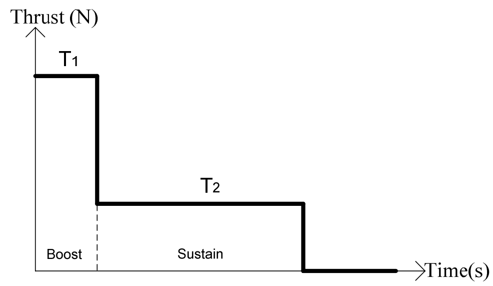


Figure 2.6 Two-phase rocket motor: Boost-Sustain

The thrust values in boost and sustain are constants depending on the motor used. Therefore, the only parameter that can be controlled in double phase rocket motor is at which phase of flight the thrust level should be changed. However, this is obvious and can be concluded as a rule of thumb if the trajectory phase sequence is known. For example, the boost is maintained during the climb phase and once the desired altitude is reached for cruise, the level can be reduced to sustain.

The speed of the missile cannot be controlled in solid fuel engine alternatives. Therefore, the desired speed should not be defined by the user once a solid fuel engine is selected in the user interface. Instead, the speed can be included within objectives to be maximized in this case. That means that the configurations which reach higher values in speed with the same thrust given are desirable. In contrast, with turbojet engine alternatives, the speed can be given as an input since the thrust profile can be adjusted during flight according that desired speed is satisfied.

2.3.2.2 Turbo-jet Model

The turbojet propulsion is suitable for subsonic cruise missiles, providing high efficiency against non-time-critical missions [1]. The advantage of turbojet engine is that the thrust can be controlled in every instant of flight which provides long range precision and controlling the speed of missile.

The thrust force needed during flight is obtained instantaneously by an auto-pilot which controls the speed. As an assumption, the thrust component in the direction of velocity is assumed to be equal to the drag force in cruise phase at steady state conditions. By means of this assumption, the magnitude of thrust can be computed in every instant of flight.

Since the value of speed can be controlled by adjusting thrust accordingly, there is no meaning to include speed into objectives to be maximized. Instead, user defines the desired velocity around which the missile should be cruised. That is, speed will be taken as a constraint in turbo-jet models. As opposed to solid fuel rocket propulsion, the thrust-time profile is not known at the beginning of simulation but will be cumulatively obtained at the end of the simulation.

After discovering the thrust profile, it should be checked out whether the given motor type can provide such a thrust profile. When the instantaneous thrust value computed is within the range specified by the upper (thrust maximum) or the lower (thrust idle) limits which the motor can supply, it remains unchanged. Otherwise, it is clipped to the upper or lower bounds. Also, the time elapsed since motor is activated should be checked if it is below the maximum run time of the motor. An example of thrust-time profile for turbo-jet motor is given in Figure 2.7. Also, it is possible to forecast the total fuel consumption in each flight trajectory phase. In other words, the propulsion model gives an opportunity of user to get information about thrust-time profile in each phase.

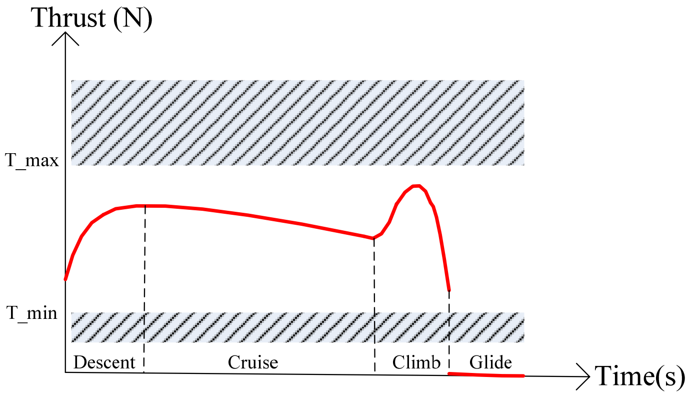


Figure 2.7 Turbo-jet Thrust vs. Time Graph

The total amount of fuel needed at the end of the flight to maintain the trajectory can be computed from the thrust profile:

$$\mathrm{T} = \mathrm{I}_{\mathrm{sp}} \frac{dm}{dt} g_0 \tag{2.12}$$

where T is the thrust, g_0 is the gravitational acceleration and I_{sp} represents the specific impulse of the motor.

Integrating the above equation with respect to time, one gets

$$\int \mathrm{T} dt = \int \mathrm{I}_{\mathrm{sp}} \frac{dm}{dt} g_0 dt = \mathrm{I}_{\mathrm{sp}} g_0 \int \frac{dm}{dt} dt \tag{2.13}$$

Hence, the instantaneous fuel mass consumed m_F can be obtained as

$$m_F = \frac{1}{I_{sp} g_0} \int_0^t T dt \tag{2.14}$$

The integral of thrust represents the total area under the thrust-time curve; therefore it can be evaluated easily. When the fuel consumed is equal to the total fuel amount given at the beginning, the motor stops and the rest of flight is performed with a glide motion. The condition that the motor operates is:

$$m_F < \frac{\eta}{100} m_L \tag{2.15}$$

where η is the ratio of total fuel mass to total launch mass m_L of the missile in percentage. It is used for calculating the amount of total fuel for a known total launch weight of the missile. This ratio is given as an input to the conceptual design tool by the user. Also, the ratio can be determined by investigating the fuel to launch mass ratio of the missiles in the literature which are in the same mission category with the missile to be designed.

In summary, the propulsion model gives an idea about the thrust profile that motor should provide. The main goal during optimization is to obtain sufficient thrust with given amount of fuel.

2.3.3 Auto-Pilot Model

Since the flight trajectory is composed of different flight phases and their sequences, there must be a control system which controls some variables during each flight phase. For example, in the cruise phase, the altitude must be kept at a specific value which is given as an input by user.

Due to existence of different phases in the same flight trajectory, the control systems can be integrated to the system as separate modules. By means of that, suitable controllers can be activated at necessary phases of the flight simulation. There are mainly three different controllers merged in auto-pilot and their activation status are given in Table 2.1.

Table 2.1 Status of Controllers in Phases

Phase	**Angle of Attack Control**	**Speed Control**	**Altitude Control**
Glide	ON	OFF	OFF
Descent	ON	ON	OFF
Cruise	OFF	ON	ON
Climb	ON	ON	OFF

Note that there will be no fin surface control in the system since the dynamics between fin surface deflections and angle of attack is assumed to be ideal since the dynamics of fin actuator system is much higher and faster than the missile dynamics. That is, it is assumed that the control system between the elevator surface and the angle of attack is so ideal that there is no time delay or steady state error in the system. As a result, the desired angle of attack is assumed to be achieved immediately.

2.3.3.1 Angle of Attack Controller

This controller is to be activated in the glide phase in order to keep the angle of attack at a certain value which makes the lift-to-drag ratio maximum. The main reason of maximizing it is to have a minimum altitude loss and a maximum range during the glide. In each time step in glide phase of the simulation, the value of trim angle of attack which makes lift-to-drag ratio maximum will be interpolated from aerodynamic look-up table. During the glide, the missile is assumed to fly with trim angle of attack which directly depends on Mach number, altitude, and external geometry.

In Figure 2.8, the desired value of angle of attack changes instantaneously according to the Mach number. In each time step, different values of trim angle of attack can be found. In addition, the angle of attack should also be controlled both for descent and climb phases for the purpose of keeping the actual flight path angle within certain limits.

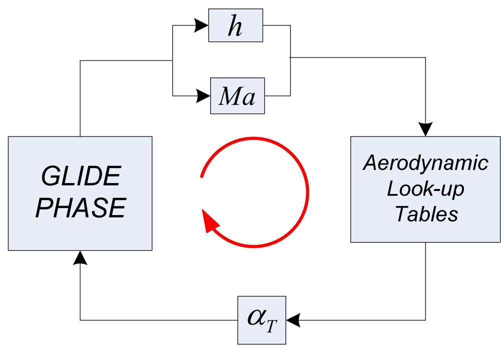


Figure 2.8 Trim Angle of Attack Interpolation

The control logic is simply proportional and shown in Figure 2.9. The controller produces such an angle of attack value that the desired flight path angle is achieved.

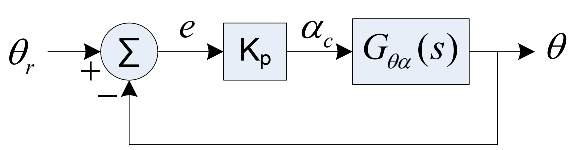


Figure 2.9 Proportional Flight Path Angle Control

2.3.3.2 Speed Controller

The speed control in missiles is usually handled by controlling the magnitude of thrust. The aim is to keep the magnitude of velocity constant at each phase except the glide when there is no thrust.

According to Figure 2.10, the net force in the x-direction of the body fixed frame is calculated and the net acceleration can also be obtained by using Newton's second law as

$$\sum F_{x_b} = m\dot{V}_{x_b} = m\dot{u} = T - mg\sin\theta - D\cos\alpha + L\sin\alpha \qquad (2.16)$$

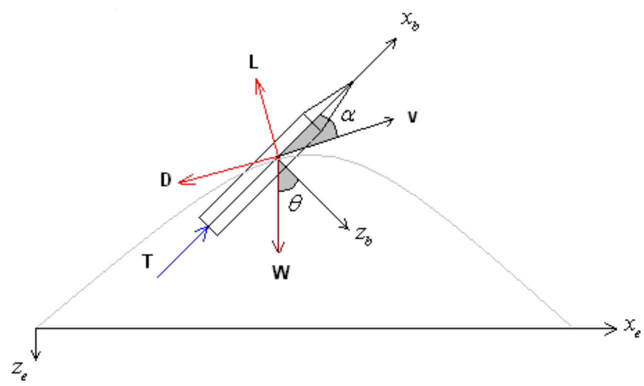


Figure 2.10 Force Diagram

In steady state condition since there is no acceleration in the direction of x-axis of the missile body ($\dot{u} = 0$), the nominal thrust force needed is found as

$$T^* = -mg\sin\theta + D\cos\alpha - L\sin\alpha \qquad (2.17)$$

In order to control the magnitude of the velocity in the direction of thrust, a simple proportional control system in Figure 2.11 is applied where the control thrust input is proportional to error in velocity component in the x-direction of body.

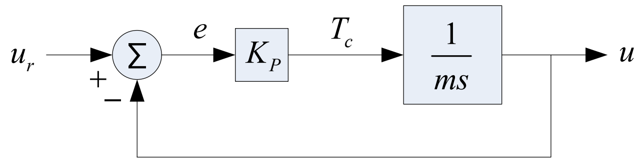


Figure 2.11 Proportional Speed Control System

The total thrust force can be represented as:

$$T = T^* + T_c \tag{2.18}$$

$$T_c = K_P e \tag{2.19}$$

$$e = (u_r - u) \tag{2.20}$$

It is assumed that the velocity component in the z-axis of body fixed frame can be neglected in case of small angle of attacks. Also, the lag between thrust and motor dynamics is not taken into account in the control system which means that the propulsion system works ideally.

In the climb, descent, and cruise phases, the speed is assumed to be constant for the simplicity in the conceptual design. However in the glide phase, since there is no thrust, the speed cannot be controlled by thrust magnitude control. In this phase, the elevator can be used as a preventer of excessive increases in speed.

2.3.3.3 Altitude Controller

The altitude control which plays significant role in cruise is handled by adjusting angle of attack according to hold altitude in cruise height. The same control theory in angle of attack control is also performed for altitude holder illustrated in Figure 2.12. According to the sign and amplitude of the error in altitude, the angle of attack will be increased or decreased which creates necessary lift force for keeping the altitude constant.

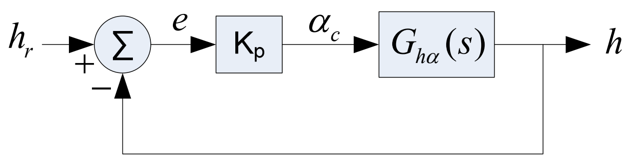


Figure 2.12 Proportional Altitude Controller

2.3.4 Atmosphere Model

An atmosphere model is necessary to include the effects of altitude on density of air and speed of sound. The COESA atmosphere model is used within MATLAB SIMULINK blocks. It implements the mathematical representation of the 1976 Committee on Extension to the Standard Atmosphere (COESA) United States standard lower atmospheric values for absolute temperature, pressure, density, and speed of sound for the input geo-potential altitude [20].

The geo-potential altitude is the only input to this atmosphere model, whereas its outputs are temperature, speed of sound, air pressure, and air density. Note that, the COESA Model extrapolates temperature linearly, but pressure and density logarithmically, beyond the interval of altitude between 0 and 84,852 meters [20]. In addition, the air density and speed of sound are calculated using a perfect gas relationship.

2.3.5 Gravity Model

The gravity model used in MATLAB SIMULINK outputs the gravitational acceleration for a given latitude and longitude by using World Geodetic System

(WGS 84) which is defined as a geocentric equi-potential ellipsoid [21]. This model has three inputs which are the latitude, longitude, and height. The acceleration of gravity acting on the missile is calculated according to these inputs.

CHAPTER 3

FLIGHT SIMULATOR

The main purpose for performing a flight simulation is to obtain some of the flight performance parameters numerically which cannot be expressed by simple analytic equations and/or empirical formulae. These flight performances include range, speed, maneuverability, and control effectiveness. Although there are some empirical formulae for calculating some of the performance parameters, they are rough estimations since there are several assumptions used in obtaining these formulae. In these expressions, the flight conditions like Mach number and angle of attack are considered to be constant implying that the missile is flying at steady-state conditions. On the contrary, there exist several transition regions along a typical flight trajectory. For example, the maneuverability of a missile should be observed especially in the pull-up maneuver which is one of the transition phases. Therefore, a numerical simulation is unavoidable when the performance metrics can be observed only in some parts of total flight profile and when their values vary during the flight. As a result, obtaining flight performance information only from analytic equations or empirical formulae might cause the designer to miss some points during design.

In this study, instead of using rough and insufficient empirical formulae for the flight performance metrics, a numerical flight simulator is designed in MATLAB SIMULINK by using models presented in the previous chapter. The main purpose of designing a flight simulator is to obtain flight performance results in an easy and systematic way.

The flow chart which shows interactions between models is given in Figure 3.1. The simulator is composed of six main modules which are EOM, Autopilot, Propulsion, Aerodynamics, Atmosphere, and Gravity.

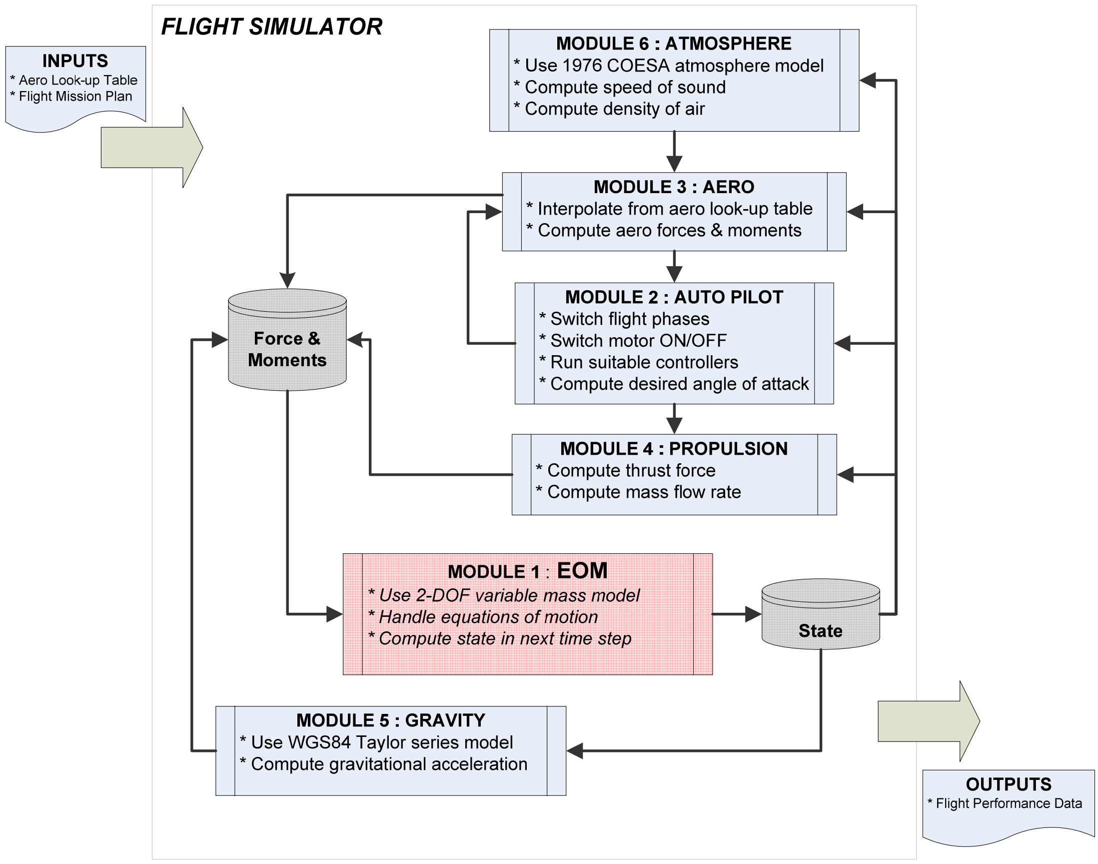


Figure 3.1 Simulator Flow Chart

The core module is called the EOM (Equations of motion) executes three degree of freedom pitch equations of motion and produces an output vector including instantaneous mass, position, orientation, linear and angular velocities in the earth-fixed frame of the missile. Also, EOM collects all forces acting on the missile body from other models which are aerodynamic, propulsion, and gravity. Autopilot module produces the angle of attack and thrust commands instantaneously by means of the controllers. Among these modules, the aerodynamic module computes aerodynamic forces and moments according to the angle of attack command received from the auto-pilot in the assumption that the commanded angle of attack is equal to the actual angle of attack. This is performed by reading aerodynamic look-up tables produced by DATCOM. The propulsion module determines the instantaneous thrust value by considering the motor status and control thrust signals received from the auto-pilot. Also, the atmosphere module supplies the air properties at a given altitude. Since the value of gravitational acceleration varies according to the missile's position in the earth, another module called Gravity is added into the system in order to compute the gravitational acceleration for a given latitude, longitude and altitude from the earth surface.

The simulation steps forward in time until a certain stop condition called ground detection is satisfied. That is, the run is stopped once the missile contacts with the hit platform which might be a land or sea platform depending on the launch type. At the end of simulation, the range, average speed, static stability, control effectiveness, and maneuverability are obtained as the output flight performance data.

3.1 Types of Flight Phases

The simulator is capable of running four different flight phases and their possible sequences. These flight phases are glide, descent, cruise, and climb. In the auto-pilot, there is a switching mechanism which can disable current phase and enable another suitable phase if necessary as determined by the auto-pilot at any instant of flight. Also, the necessary controllers are activated according to the phase enabled.

3.1.1 Glide Phase

Glide is the flight phase that the missile makes a free-fall motion, therefore it has an altitude loss without any assistance of thrust force. This phase can be enabled under two different conditions. One of them is at the beginning of flight until the motor is activated; the other is through the end of flight when the fuel is run out. During the glide, controllers are run such that the missile should fly with the least altitude loss and the maximum range. To achieve these conditions, the glide phase should be performed with a maximum lift-to-drag ratio. The forces during glide are shown in Figure 3.2 .

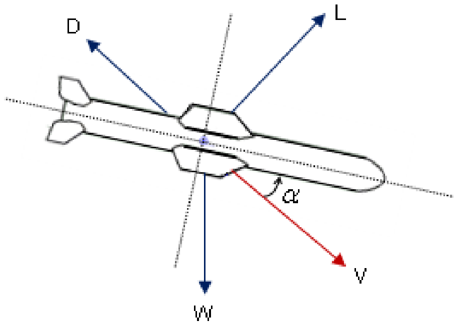


Figure 3.2 Glide Force Diagram

In order to obtain minimum altitude loss and maximum range in glide phase, a desired angle of attack should be obtained by interpolating aerodynamic look-up tables according to the instantaneous Mach number and altitude. The controller tries to keep the angle of attack at a desired value. The related equations are given below:

$$\alpha_T = \alpha\Big|_{\left(\frac{L}{D}\right)_{\max}} \tag{3.1}$$

$$T = 0 \tag{3.2}$$

where α_T represents the trim angle of attack which makes lift-to-drag ratio maximum at an instantaneous Mach number. The trim angle of attack is the angle of attack which keeps the pitch angle in equilibrium.

3.1.2 Descent Phase

In contrast to the glide phase, the missile is assisted by thrust in the descent phase for the missile to descend to the cruise altitude or to the target altitude with a desired descent speed. In this phase, it is assumed that the missile keeps its speed constant without any acceleration. Therefore, the descent controller keeps the speed of the missile constant by adjusting the value of the control thrust. However, it is possible to find out an optimum speed profile in descent or climb phases in order to minimize the fuel consumption rate. Since the scope of this thesis is optimizing the external geometry, the speed profile optimization problem is not studied. The fuel consumption is tried to be minimized implicitly by minimizing the weight of the missile. Therefore, parameters to be optimized in this study include only the external geometry configuration and dimensions.

According to Figure 3.3, there is no net force in the x-axis of missile body in steady state. The related equation is given as:

$$T = -W \sin\theta + D\cos\alpha - L\sin\alpha \tag{3.3}$$

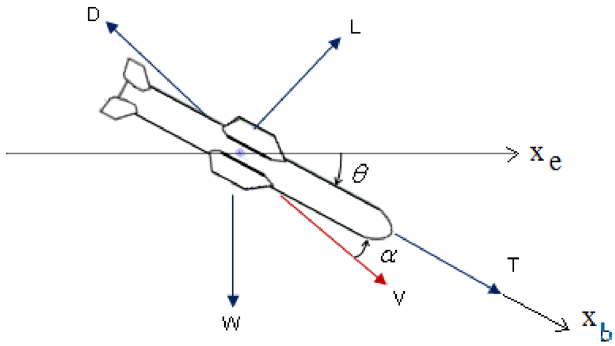


Figure 3.3 Descent Force Diagram

The flight path angle θ shown in Figure 3.3 can be specified by the user at the beginning. In order to obtain the desired flight path angle, the angle of attack will be adjusted until the actual flight path angle equals to the desired value. The proportional controller uses the following equation to produce a control angle of attack α_C:

$$\alpha_C = K_P(\theta_r - \theta) \tag{3.4}$$

where K_P is the proportional gain of the angle of attack control system and θ_r represents the desired flight path angle during descent.

3.1.3 Cruise Phase

The longest part of a trajectory is maintained in the cruise phase. This phase is the level portion of missile travel in which a constant altitude and speed are maintained. These constant values of cruise speed and altitude in the mission profile are picked up by the user.

The missile may pass through several pre-defined navigation points in cruise. In order to maneuver between these points, the missile must change its orientation by keeping both its speed and altitude constant which is called sustained maneuver. Since the dynamic model includes only the vertical planar motion, no sustained maneuver in the horizontal plane is modeled in the simulation. In the conceptual design, the navigation points are not usually determined yet. Therefore, it is unnecessary to model sustain maneuver in the dynamic model. That is, the trajectory is assumed to have no navigation points and the flight is performed only in the vertical plane.

3.1.4 Climb Phase

Climb is the phase which is aimed to raise the missile to the cruise or search altitude. In search altitude, seeker begins its search in order to detect the target. However, the climb to the search altitude motion is not necessary for the missiles having no seeker. At the beginning of this phase, the missile performs a pull-up maneuver shown in Figure 3.4 until a desired climb angle is reached. After obtaining the desired climb angle, the missile climbs at a constant speed till it is reached to a pre-defined altitude.

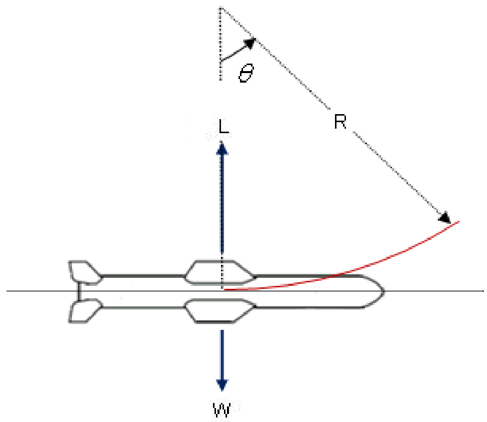


Figure 3.4 Pull-up Maneuver

According to Figure 3.4, the expressions for load factor n and turn radius R are given as

$$n = \frac{L}{W} \tag{3.5}$$

$$R = \frac{V^2}{g\left(n-1\right)} \tag{3.6}$$

where V is the speed of the missile and W is the total weight of the missile.

After reaching a desired climb flight path angle in a pull-up maneuver, the climb phase continues with a constant velocity. In order to have a more maneuverable missile the load factor should be increased by shortening the turn radius.

The missile reaches its maximum load factor once the lift force becomes maximum in a pull-up maneuvering. However, while maximizing the lift force, the angle-of-attack should be kept below a certain critical limit above which the coefficient of lift is decreasing. Above this critical angle of attack, the missile is said to be in a stall. The stall angle of attack limits put an upper and lower bounds to the control angle of attack command which is produced by autopilot. It depends not only to the control surface type (canard, wing, tail) but also to the speed of the missile. A typical lift coefficient change with angle of attack is illustrated in Figure 3.5:

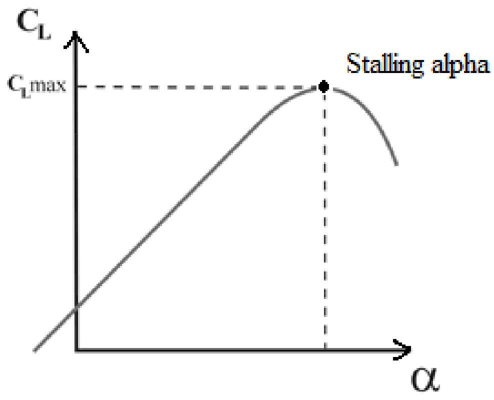


Figure 3.5 A Typical Lift Coefficient Curve

The stalling alpha should be observed from the lift coefficient lookup tables created by DATCOM for each external geometry at the beginning of flight simulation. The autopilot commands are limited by this stall angle of attack value.

3.2 Trajectories with Combined Phases

The whole trajectory is composed of possible combinations of flight phases. The sequence of these phases in a flight simulation can be defined by user. Hence, the user has the authority to select one of many possible flight profiles for different mission purposes. Therefore, the external configuration optimization becomes strictly trajectory-dependent. The feasible phase combinations and sequences are listed in the user interface. The user is requested to select one of them within the list at the beginning of design. By means of this, it is prevented the user to select unfeasible sequences and combinations of the flight profile phases. For example, there cannot be such a trajectory with a cruise-glide-climb sequence. Possible flight phase sequences can be classified according to the flight type (i.e. air-to-surface) and motor type (i.e. turbojet) as listed in Table 3.1.

Table 3.1 Flight Trajectory Combinations

	Air-to-surface	**Surface-to-surface**
Turbojet	Glide-Descent-Cruise-Climb-Descent	Climb-Cruise-Climb-Descent
	Glide-Descent-Cruise-Descent	Climb-Cruise-Descent
Solid-fuel	Climb-Glide	Climb-Cruise-Glide
	Glide-Descent-Cruise-Glide	
No Thrust	Glide	-

By means of these classifications, the most suitable motor type that fits to the missile can be predicted at the beginning according to the flight profile that user selects. Since each flight phase is modeled as an independent module, every possible combination can be simulated. This gives a freedom to the user on optimizing both air-to-surface and surface-to-surface type of missiles. Also, the flight of the air vehicles which have no propulsion system can be simulated and its trajectory is composed of glide phase only. By means of that, the impact of thrust on the external geometry optimization can be observed.

The flight simulator should be able to decide how long the missile should be maintained in the cruise phase and when the cruise phase should be stopped. In solid fuel rocket alternatives, since most amount of the fuel is consumed at the beginning of the flight, a climb phase cannot usually be operated after a cruise phase. That is, the missile can run in cruise phase until whole fuel is consumed. Therefore, the time when a cruise phase is stopped is certain. However, in turbojet alternatives, the missile can climb after a cruise phase; therefore, a necessary amount of fuel must be needed and remained at the end of cruise phase in order to handle following climb motion. That is, estimating the end of cruise phase is a bit difficult by solving backward iteration of the flight simulation. Therefore, the climb phase should be run before a cruise phase in a reverse manner. By means of this, the total amount of fuel needed for handling a terminal climb will be found out. The cruise phase is paused once the fuel amount remained is equal to the fuel amount needed to operate climb phase. The condition for pausing cruise and starting the climb is given as

$$\left(m_F\right)_{Current} = \left(m_F\right)_{Climb} \tag{3.7}$$

The critical point is to obtain the initial conditions for a climb motion. The final condition of the cruise phase is the initial condition of a climb phase at the same time. The steady state conditions of the cruise phase should be determined first and then the climb phase can be run. Therefore, the cruise phase is run until steady state conditions are satisfied. That is, steady state conditions can be satisfied once the variation in altitude of the missile drops below a certain tolerance limit (+/- 10 m.). When the steady state is reached, the cruise phase is paused and the climb and following glide or descent phase modules are executed. Once the impact is obtained, the simulation is restarted from the point at which the cruise phase is paused. The operation sequence of phases for a sample flight trajectory profile is illustrated in Figure 3.6:

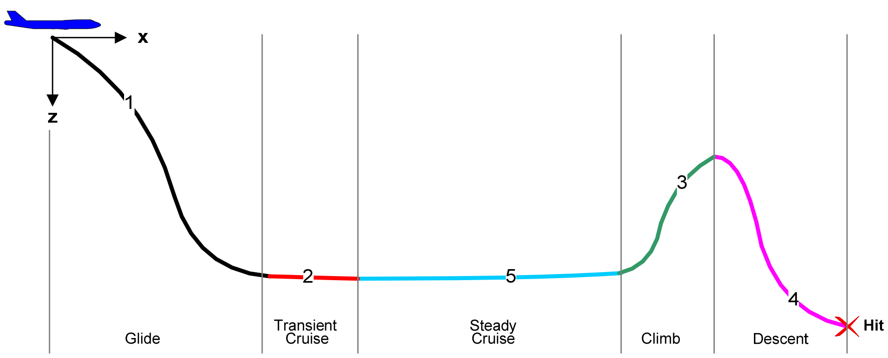


Figure 3.6 Sequences of Phases in a Trajectory

CHAPTER 4

FLIGHT PERFORMANCE

The aim of preliminary design stage is to determine the optimum baseline external geometry which satisfies the flight requirements the best. This baseline geometry is to be used as a starting point of the more detailed design stages. In order to obtain the optimum baseline geometry, the optimization parameters, objectives, and the constraints of the problem should be determined correctly. Since the external geometry parameters have a strong impact on the missile aerodynamic measures of merit and constraints, they are taken as the parameters to be optimized. On the other hand, the flight mission requirements are treated under two groups of the optimization problem; namely, the objectives and constraints.

Objectives can be called as the flight performance measures of merit which might be the weight, range, maneuver, and time to target whereas constraints can be defined as the limitations that restrict the missile from achieving its goal in a better manner. The launch platform integration weight, length, and span limitations can be examples of constraints which affect the external shape of missiles directly. Therefore, the effect of external configuration and sizing parameters on the performance measures should be defined clearly before starting an optimization algorithm.

4.1 Aerodynamic Configuration Sizing Parameters

Aerodynamic configuration sizing parameters are taken as the optimization parameters which assemble the external shape of a missile. Since the external geometry of a missile determines aerodynamic forces and moments acting on it, these parameters are considered as crucial effects on the its flight performance. They can be divided into two groups; namely, external configuration parameters and geometrical dimensions.

4.1.1 External Configuration Parameters

External configuration parameters can be selected among a pre-defined discrete set. In this thesis, the configuration parameters to be focused are the fin configuration, type of control, nose shape, and roll orientation. Their alternatives should be investigated and eliminated according to their impact on the performance measures and merits of the missile.

The fin configuration determines the number and placement of the fin surfaces on missile body. Fins can be called as canard, wing, or tail according where they are located on the main body. A canard is the small flat control surface which is located in the front section of a missile whereas a tail is on its rear. However, a wing is a comparatively large lifting surface which can be either fixed or folded out and located in the mid section of a missile. Some combinations of canard-wing-tail are illustrated in Figure 4.1.

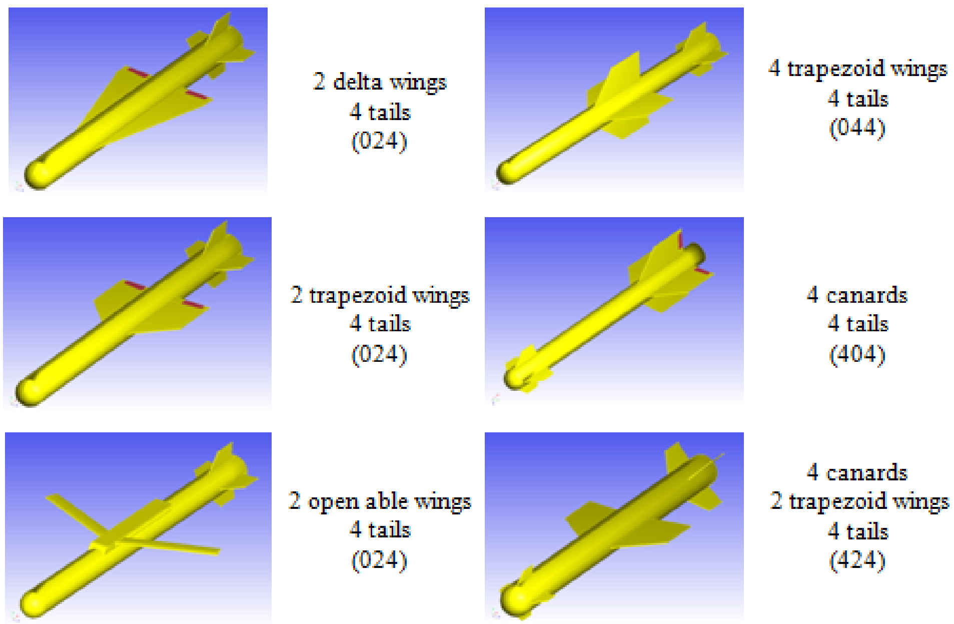


Figure 4.1 Some of the Fin Configuration Alternatives

In the representation of the fin configuration, first digit denotes the number of panels of the canard, second digit shows the number of wing panel surfaces and the last digit represents the number of tail fins. Among those canard- wing-tail configurations, only three of them, namely, 404, 024 and 044 which are most common missiles in the literature are to be investigated in this study in order to shrink the domain of the problem.

Aerodynamic control alternatives (i.e., tail fin, canard, and wing) are other leading factors on the control effectiveness of a missile. They determine the configuration of fixed lifting surfaces and movable control fin surfaces. Maneuvering like pitch, yaw, and roll rotations is performed by either all movable or flap fin surfaces which can be deflected by an angle. The canard control is preferred over tail control in missiles when a high maneuverability is required. But this configuration has a higher possibility to lead stall at high angle of attacks. In missiles, the wing control is not preferable and missiles with wing control have not been developed in recent years due to deficiencies such as large hinge moments needed and large induced roll [8]. Modern missiles use either tail or canard control. Therefore, in this study, the domain of the missile control problem is reduced to only two alternatives, by canard or by tail.

The nose shape is the dominant factor which affects the magnitude of aerodynamic drag force. A sharp nose is ideal aerodynamically, producing a small drag force. However, it is not suitable if there is a need of installing of the seeker at the nose tip due to insufficient available space for sharp noses. Seekers should be placed as close as possible toward the tip of nose in order to enhance their radar/infrared/laser detection range property. Since there is more space at the tip of blunted noses, they are more suitable for a seeker to have more precision. However, blunted noses create larger drag forces. These electromagnetic and aerodynamic limitations should be considered and balanced accordingly during the selection of nose type. Since the Missile DATCOM computes aerodynamic coefficients according to missile geometry, only the certain nose shapes can be used which are already defined in database of DATCOM [4]. They are mainly Ogive, Conical, Power, Haack and Karman. They use different formulas for nose curvatures as tabulated in Table 4.1

Table 4.1 Nose Shape Formulas [22]

Nose Shape	Equation(s)	
Conical	$y = \frac{xR}{L_N}$	-
Power	$y = R\left(\frac{x}{L_N}\right)^n$	$0 \le n \le 1$
Tangent Ogive	$y = \sqrt{\rho^2 - (L_N - x)^2} + R - \rho$	$\rho = \frac{R^2 + L^2}{2R}$
Haack	$y = R\sqrt{\frac{1}{\pi}\left(\theta - \frac{\sin(2\theta)}{2} + \frac{1}{3}\sin^3\theta\right)}$	$\theta = \arccos(1 - \frac{2x}{L})$
Von Karman	$y = R\sqrt{\frac{1}{\pi}\left(\theta - \frac{\sin(2\theta)}{2}\right)}$	$\theta = \arccos(1 - \frac{2x}{L})$

The panel orientation of fin surfaces is another factor which may enhance the lift properties but could reduce the control effectiveness [8]. Although the number of panels for a one fin set can be 2, 3, 4, 6 and 8, the scope of this study is kept limited to only 2 and 4 panels which can be oriented as plus or cross, as illustrated in Figure 4.2. In missiles, large wings usually have 2 panels because of the space limitations. Also, the most common type of tails is composed of 4 panels. These are the reasons of choosing the most common panel numbers (2 and 4) which also helps reducing the domain of the problem. Both orientations have fin panels which are assumed to be perpendicular to each other.

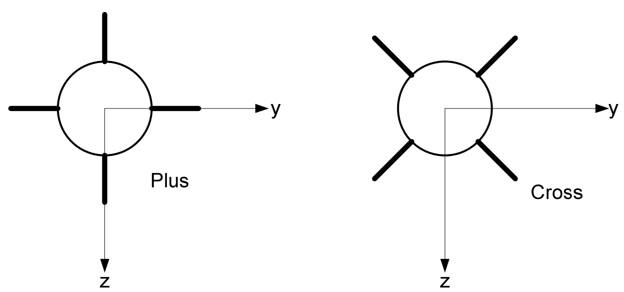


Figure 4.2 Types of Panel Orientations (Back view of missile)

Note that some of the configuration variables can also be automatically eliminated at the beginning of the design since the designer is free to start with setting some of the external configuration variables as constant. For example, in case of defining the panel orientation as plus or cross, this configuration parameter is not included into aerodynamic configuration sizing parameters. If the designer does not specify any of the configuration parameters at the beginning of the design, all possible configuration alternatives must be investigated in order to obtain optimum configuration.

Therefore, all configuration families should be created and classified for every possible configuration. Each individual in the same family will have different geometric dimensions. The best individual of each family should be determined and compared according to their costs. Comparing best individuals of each family, the particular individual with a minimum cost value should be determined and declared as the best one. Its configuration family will then be accepted as the best configuration. In summary, the characteristics that distinguish a family from other families are grouped in Table 4.2:

Table 4.2 Configuration Variables

#	**Configuration Variable**	**Possible Values Set**
1	Fin Configuration	{404, 024, 044}
2	Control Type	{Canard, Tail}
3	Nose Shape	{Ogive, Conical, Power, Haack, Karman}
4	Panel Orientation	{Plus, Cross}

According to Table 4.2, the total number of families which is composed of combinations of external configuration parameters is 60 (=3×2×5×2)

However, the canard surface control cannot be applied to 024 and 044 configuration types since there is no canard in these configurations. Therefore, the total number of families reduces down to 40 (=60–1×2×5×2) unless the user already specifies some of the configuration variables.

4.1.2 External Geometry Dimensions

External geometry dimensions are assumed to vary continuously between some upper and lower limits. They are classified under three groups in the same manner with DATCOM inputs; namely, axial body, nose, and fin set dimensions [4]. The dimensions of a missile are shown in Figure 4.3:

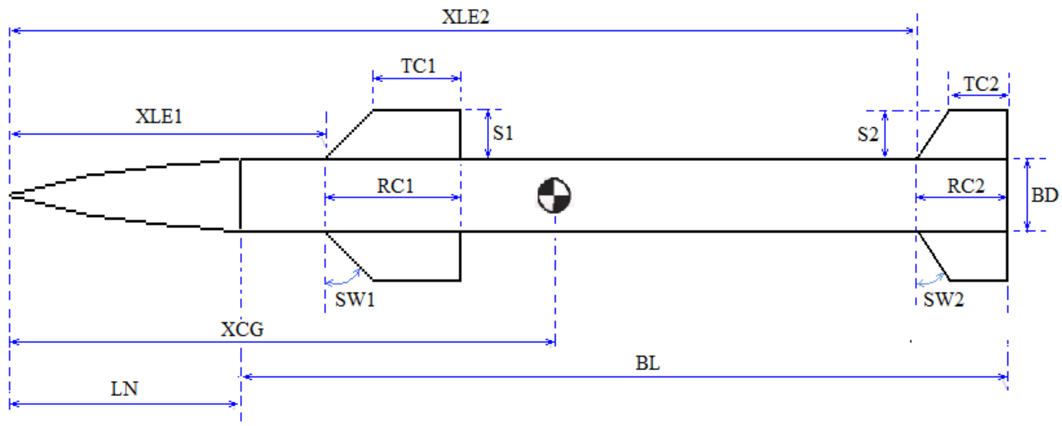


Figure 4.3 External Geometric Dimensions

Axial body parameters include the diameter and length of the main body of missile. In this thesis, the cross-section of missile is assumed to be circular and there is no diameter change along the main body. Also, the center of mass is another parameter of the main body which affects the aerodynamic moment coefficients. Although missiles having small body fineness ratio (l/d) have improved launch platform compatibility, the small body fineness ratio deteriorates aero-elasticity of the missiles and increases the body bending frequency of the missiles. These two effects of small body fineness ratio of the missile are not desirable [9].

Another factor which directly impacts the body diameter is the diameter of the motor which is fit into the rear of the missile body. As the turbo-jet motors have usually variable diameter, the maximum diameter of the motor should be taken as a lower limit of body diameter of the missile. However, the diameter of the motor in solid fuel rockets can be adjusted according to the diameter of the missile body. That is, the diameter of the solid fuel rocket motor is determined at the end of the conceptual design.

Also, the nose fineness defined as the ratio of nose length to nose diameter affects the performance of the missile. Missiles with high nose fineness are aerodynamically ideal and low observable whereas missiles having low fineness nose allows more propellant length and volume for length-limited missiles [9].

Missiles having two different fin sets will be studied in this thesis, therefore the total number of fin set parameters is 2 times of 5. These fin set parameters are leading edge sweep angle, length of root chord, tip chord, span, and the distance between leading edge and tip of nose. All the geometrical parameters are listed in Table 4.3,

Table 4.3 Geometric Dimensions

Parts	**Dimensions**	**Symbols**
AXIBODY	Length	BL
	Diameter	BD
	Center of Gravity	XCG
NOSE	Length	LN
FIN SET	Span	S
	Root chord	RC
	Tip chord	TC
	Sweep angle	SW
	X-location	XLE

4.2 Measures of Merit

At the beginning of the pre-design stage, the aim is to obtain a baseline external geometry and a configuration which satisfy the requirements of the flight. Every customer expects some skills from the missile to be designed. The statements of these features can be expressions containing “as much as” or in similar nature. These statements will constitute the objectives of design which the designer must maximize, minimize, or obey the restrictions. The performance measures which must be taken into account in the conceptual design are as objectives

- Maximum range
- Maximum speed
- Minimum total mass

and as constraints

- Stable both in statically and dynamically
- Controllable sufficiently in all three axes
- Maneuverable enough to follow a given trajectory.

Each objective should be represented in the cost function. The values of most objectives can be numerically obtained from the flight simulator. Although some of them are instantaneous, some others are cumulative objectives. The speed, control effectiveness, and maneuverability are instantaneous measures of merit since they can be obtained every instant of flight. However, the range is a cumulative objective since it is obtained only after completing the whole flight. For instantaneous objectives, the most important thing is to find out in which phase or segment of the flight they are more dominant or negligible. Once this information is available, there may be no need to check all objectives whether they are satisfied in each phase of the flight simulation. Instead, only the critical parts of the flight trajectory profile are focused for different objectives, which is a much more effective and less time consuming approach. For example, the maneuverability can only be observed from turn rates in a climb phase or only in a pull-up segment in a 2-DOF flight simulation. To get a single value, the average of turn rates can be taken as a cost value of this objective.

After completing the computations of missile flight performance, the next step is to compare the current candidate for the flight performance with the mission requirements for an optimum flight performance. To sum up, converging to a design that harmonizes the aerodynamics, propulsion, and weight as well as satisfying the flight performance requirements is a primary activity in missile configuration design.

4.2.1 Total Flight Range

The designed missile is expected to be delivered to a range which is guaranteed to be maximized for the defined flight trajectory and limitations. This strictly depends on the lift to drag ratio; therefore, is affected by the external geometry. By changing the external geometry only, extending or shortening range is possible with the same amount of fuel. The range is a function of numerous parameters; therefore, it is unwise and difficult to express it as an analytical function of all these parameters. However it is possible to predict it with some rough assumptions. For example, the Breguet Range Equation is given as [8]:

$$R = \left(\frac{L}{D}\right) I_{sp} V_{AVG} \ln\left(\frac{W_L}{W_L - W_F}\right) \tag{4.1}$$

where the specific impulse I_{sp}, launch W_L and fuel W_F weights as well as desired average velocity V_{AVG} are known at the beginning of flight.

However, the lift to drag ratio is hard to estimate since it is also a function of the angle of attack which always changes during flight. Even though the trim angle of attack can be estimated during the cruise phase, there are also other phases in a flight like climb, glide, pull-up/down in which the angle of attack cannot be considered as a constant. Therefore, it would be very rough estimate if Breguet Range Equation is used as an analytical expression of range. As a result, it is more convenient and easy to obtain the range numerically from the flight simulator.

The ratio of fuel to total launch weight both for turbojet and solid rocket engine is assumed to be constant and it can be set as an input by the designer. By means of this, the missile which is delivered to the maximum range with a pre-defined amount of fuel can be said as the optimum. Similarly, the concept of minimizing fuel consumption with a specified range has the same logic with that of maximizing the range according to a specified fuel amount. One of them should be the input and the other kept as an objective. Instead of minimizing the fuel amount and maximizing the range at the same time, one of them will be taken as a constraint defined by the user. As a result, the range is decided to be selected as an objective and the fuel amount is taken as an input. Therefore, whichever external geometry candidate of the missile is delivered to the longest range with a given fuel input will be best.

In real life, the range is not actually a factor to be maximized. Instead, the customer defines the allowable operational range [R_{min} R_{max}] for the missile. Therefore, the range should be adjusted according to its upper R_{max} and lower R_{min} limits which are defined by the user. Therefore, it can be taken as an inequality constraint instead of an objective to be maximized:

$$R \subset [R_{min} \;\; R_{max}] \tag{4.2}$$

However, missiles with solid propellant motor consume most of its fuel just at the beginning of flight, and then maintain a glide phase. Greater the lift to drag ratio is the greater the range will be. That is, the external geometry will play an important role in the range. Since there is no thrust in the glide phase, the geometries having greater range will be more acceptable. Therefore, the range is considered to be an objective to be maximized in solid fuel rocket options.

4.2.2 Average Cruise Speed

The speed can be taken as an objective to be maximized in a time-critical mission. Maximizing the speed provides faster destruction of targets. Time critical missions are mostly observed for missiles with solid propellant motor as they can be operated at high Mach numbers whereas turbojet engine missiles are actually used for stationary targets where there is no time constraint in the mission [29]. Therefore, the speed is not an objective if the user selected the turbojet engine option in the user interface. The geometry candidate whose average speed is closest to desired one is the best.

4.2.3 Total Launch Mass

There are numerous benefits of designing a missile which has low mass. The advantages include low production cost, low logistics cost, smaller size, and low radar detection. According to an investigation on 48 tactical missiles in literature which are shown in Figure 4.4, it is observed that there is approximately a specific ratio between launch mass and approximate volume of missile. According to this result, the average subsystem density of a tactical missile is found approximately 1,384 [kg/m^3] (0.05 lb/in^3) [8].

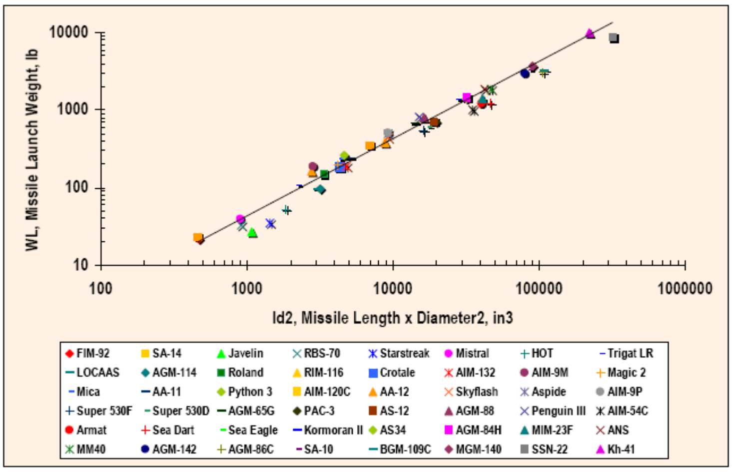


Figure 4.4 Missile mass is a function of diameter and length [8]

Except for the nose section, most missile bodies can be represented as a cylinder. Neglecting the geometry difference of the missile nose from a cylinder and neglecting the mass of surfaces, the missile launch mass m_L would be correlated as:

$$m_L = \rho_M V = 1384(\frac{\pi}{4})ld^2 = 1087ld^2 \text{ [kg]} \qquad (4.3)$$

where ρ_M is the average density of the missile, V is the volume of the missile, l is the total length of the missile and d is the diameter of the missile.

According to Equation 4.3, minimizing mass implicitly minimizes the volume of missile as well as reducing the drag force acting on the missile body. Also, in most launch platforms, there is a maximum allowable mass of a missile to be carried. Therefore, a smaller mass design makes an advantage for launch platform compatibility.

As seen in Figure 4.5, a missile has several subsystems which are nose, warhead, guidance/control (G&C), fuel tank, and motor sections. Since each subsystem has a different mass value, the center of mass of the missile is estimated by summing the mass of each subsystem times its distance from the nose tip, divided by the missile total mass.

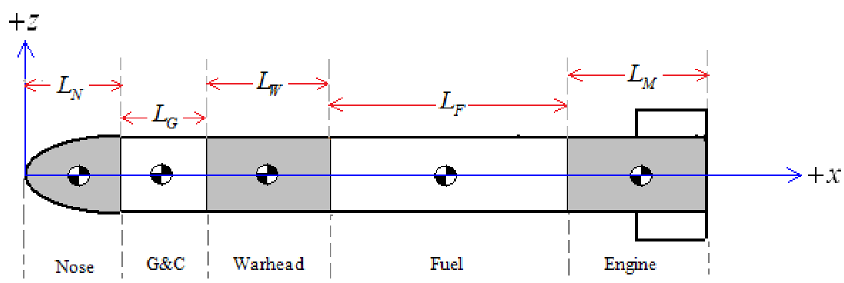


Figure 4.5 Components and their Lengths of a Missile

The center of mass of the missile x_{CG} is:

$$x_{CG} = \frac{\sum_{1}^{n}(x_i)(m_i)}{m_L} \qquad (4.4)$$

where x_i and m_i represent the center of mass location and mass of the i^{th} subsystem component of the missile, respectively.

Since the missile subsystems are composed of nose, guidance, warhead, fuel tank and motor sections, the center of mass of the missile can be expressed:

$$x_{CG} = \frac{x_N m_N + x_G m_G + x_W m_W + x_F m_F + x_M m_M}{m_L} \tag{4.5}$$

where x_N, x_G, x_W, x_F, and x_M are the center of mass locations of the nose, guidance, warhead, fuel tank, and motor sections with respect to the body-fixed frame. m_N, m_G, m_W, m_F and m_M represent the masses of each section nose, guidance, warhead, fuel tank, and motor sections.

Note that the center of mass should be calculated both for launch and burnout, which will be taken into account during flight simulation. The mass m_{burnout} of the missile when the fuel tank is empty can be obtained as

$$m_{\text{burnout}} = m_L - m_F \tag{4.6}$$

If there is a difference between burnout and launch center of mass x-locations, it implies that the center of mass is changing by burning fuel out. The difference depends on the location of fuel tank in the missile body. If the fuel tank's center of mass is very close to the missile's center of mass, the effect of center of mass change can be neglected. In such a case, a fixed center of mass location is taken during the flight.

Note that aerodynamic forces remain the same in case of a shift in the center of mass during the flight since the external geometry is fixed. However, the aerodynamic moments are changed, because the moment arm between center of pressure and center of mass is varied. But, there is no need to run DATCOM according to new location of XCG. Instead of that, some modifications in computing pitch moment should be handled.

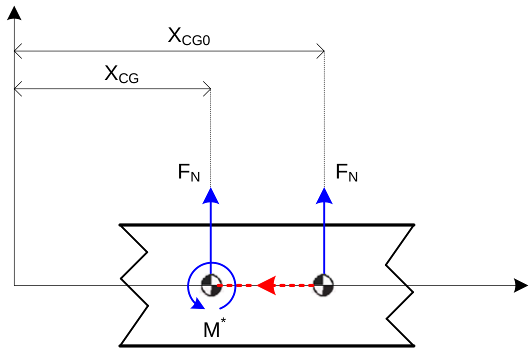


Figure 4.6 Change in Center of Mass

According to Figure 4.6, after transferring the normal force F_N to the new XCG, the new pitch moment M^* including the effect of XCG change during flight can be computed as:

$$M^* = 0.5\rho V^2 C_m S + F_N \left(X_{CG0} - X_{CG} \right) \tag{4.7}$$

where F_N is the normal force acting on missile body, X_{CG0} is the center of gravity location of the missile at launch and X_{CG} is the center of gravity location of the missile at any instant of flight.

The pitch moment is not taken into account in the calculations of the 2-DOF model. However, the center of mass variations during flight alters C_m which determines both the static stability margin C_{m_α} and the control effectiveness $C_{m_\alpha} / C_{m_\delta}$ of the missile. Therefore, the coefficients C_{m_α} and C_{m_δ} which are obtained from DATCOM should be updated. At any instant of flight, the new pitch moment coefficient C_m^* is obtained as

$$C_m^* = C_m + C_N \left(X_{CG0} - X_{CG} \right) \tag{4.8}$$

where C_N and C_m are the normal force coefficient and pitch moment coefficient, respectively, produced by DATCOM. After taking derivative of both sides of Equation 4.8 with respect to angle of attack, one gets

$$\frac{\partial C_m^*}{\partial \alpha^*} = \frac{\partial C_m}{\partial \alpha} + \frac{\partial C_N}{\partial \alpha} \left(X_{CG0} - X_{CG} \right) \tag{4.9}$$

$$C_{m_\alpha}^* = C_{m_\alpha} + \frac{\partial C_N}{\partial \alpha} \left(X_{CG0} - X_{CG} \right) \tag{4.10}$$

where $C_{m_\alpha}^*$ is the modified C_{m_α} which is obtained from DATCOM results.

Applying the Newton's backward difference method for differentiating the second term of the right side of Equation 4.10 yields

$$C_{m_\alpha}^* = C_{m_\alpha} + \left(\frac{C_N \Big|^{\alpha=\alpha^*} - C_N \Big|^{\alpha=0}}{\alpha^* - 0} \right) \left(X_{CG0} - X_{CG} \right) \tag{4.11}$$

In the same method, in finding new pitch moment derivative, the derivative of pitch moment coefficient with respect to fin deflection δ can be obtained as:

$$C_{m_\delta}^* = C_{m_\delta} + \left(\frac{C_N \Big|^{\delta=\delta^*} - C_N \Big|^{\delta=0}}{\delta^* - 0} \right) \left(X_{CG0} - X_{CG} \right) \tag{4.12}$$

By means of obtaining the new values of derivative of pitch moment coefficient with respect to the angle of attack and fin deflection analytically, there is no need to execute DATCOM every time the center of mass changes during the flight. This prevents time loss during simulation.

4.2.4 Aerodynamic Control Effectiveness

Stability and control have impacts on the aerodynamic configuration design, particularly in tail sizing; and they should be considered early in the conceptual design [8]. The aerodynamic control effectiveness is defined as the effect of control surface deflections on the pitch, roll, and yaw angles of the missile. The roll, yaw, and pitch control effectiveness of a missile are defined as:

- Roll due to rudder : $C_{l_{\delta_r}} / C_{l_{\delta_a}}$
- Yaw due to aileron : $C_{n_{\delta_a}} / C_{n_{\delta r}}$
- Roll due to side slip : $C_{l_\beta} / C_{l_{\delta_a}}$
- Roll due to roll angle : $C_{l\phi} / C_{l_{\delta_a}}$
- Pitch due to alpha : $C_{m_\alpha} / C_{m_\delta}$
- Yaw due to side slip : $C_{n_\beta} / C_{n_{\delta_r}}$

Since the flight simulator described in Chapter 3 performs only a two degree of freedom model in vertical plane, the only relevant control effectiveness ratio is pitch due to alpha:

$$\frac{C_{m_\delta}}{C_{m_\alpha}} = \frac{\Delta C_m}{\Delta \delta} \frac{\Delta \alpha}{\Delta C_m} = \frac{\Delta \alpha}{\Delta \delta} \tag{4.13}$$

Therefore, the control effectiveness related to roll and yaw rotations are not investigated in the conceptual design. The derivative coefficients of pitch moment with respect to fin deflections are produced by Missile DATCOM. To do that, deflection angle sets of each fin set should be defined in DATCOM. A rule of thumb for conceptual design of a tail or canard control missile, the ratio should be more than one because of the limitation on maximum turn capacity of the fin actuator motor that creates hinge moment [8]. The inequality constraint on the angle of attack change $\Delta\alpha$ which is created by giving a minimum fin deflection $\Delta\delta$ is:

$$\frac{\Delta\alpha}{\Delta\delta} \geq 1 \tag{4.14}$$

The motor precision of the fin actuator system is defined as the minimum deflection angle that motor can give to the control surfaces. If it is less than the maximum angle of attack and the control effectiveness ratio is less than one, there would be no way to give a full angle of attack to the missile. For example, this ratio is desired to be very high in air to air high maneuverable missiles since fast and large angle of attack changes are required with small deflections [9]. To sum up, there are mainly two factors which determine the control effectiveness of a missile; namely, fin actuator precision and flight mission. Even though the mission drives the control effectiveness to higher values, the motor precision may be a limiting factor which puts an upper bound to it. Therefore, it is more suitable to take it as a constraint rather than an objective to be maximized. The bounds of control effectiveness can be defined as an input to the optimization by balancing these three factors. The corresponding inequality expression for the control effectiveness is given as

$$1 \leq \left(\frac{\Delta\alpha}{\Delta\delta}\right)^{l} \leq \frac{\Delta\alpha}{\Delta\delta} \leq \left(\frac{\Delta\alpha}{\Delta\delta}\right)^{u} \tag{4.15}$$

where $\left(\frac{\Delta\alpha}{\Delta\delta}\right)^{l}$ and $\left(\frac{\Delta\alpha}{\Delta\delta}\right)^{u}$ are the lower and upper limits of control effectiveness.

The control effectiveness is a dominant factor especially in the regions where a flight phase transition occurs. In other words, it is effective in maneuvering when the missile is passing from cruise to climb phase or from climb to glide phase. That is, those critical regions should be focused for this performance measure of merit.

4.2.5 Maneuverability of a Missile

The aerodynamic control effectiveness is important since it determines how much an angle of attack is resulted by creating fin deflections. However, how fast this change can be occurred is another important question which is determined by maneuverability. The maneuverability determines the missile's ability of maneuver and has a direct relation to the maneuver load factor which depends on mass of the missile. In a tactical missile design, there are three criteria about maneuverability [9]. The missile should be sufficiently maneuverable in order

- to follow the flight trajectory easily
- to overcome disturbances quickly
- to be inside the safe region of the structural load factor

In order a missile to follow a trajectory with a certain speed given in the mission plan and to reject of disturbances like wind, it should have a minimum load factor (n^l) which brings a lower limit to the maneuver load factor of the missile. For example, the cruise means 1g level flight. Therefore, the missile have at least 1g load factor as a default. Also, the structural design load factor is a significant contributor to the agility of the missile. If the aerodynamic load factor exceeds the maximum structural load factor n^u that missile material can endure, there might be a structural failure. As a result, maneuverability can be considered as a constraint having desired bound as

$$1 \leq n^l \leq n \leq n^u \tag{4.16}$$

Similar to the control effectiveness case, the pull-up/down phase of flight is the most important for observing maneuverability.

4.2.6 Static Stability

The static stability in pitch is defined by the slope of the pitching moment versus the angle of attack. In order to have static stability in the pitch motion, the slope of pitching moment coefficient C_{m_α} versus angle of attack must be negative; i.e,

$$C_{m_\alpha} = \frac{\Delta C_m}{\Delta \alpha} \leq 0 \tag{4.17}$$

where C_m denotes the pitching moment coefficient and α is the angle of attack.

Increasing the angle of attack causes a negative pitching moment which drives nose down and damps the rotation of missile body. However fin deflections help the missile to preserve its angle of attack at trim. This condition is only obtained by taking center of pressure closer to the tail far away from the center of mass. Otherwise, the missile might turn over around its pitch axis.

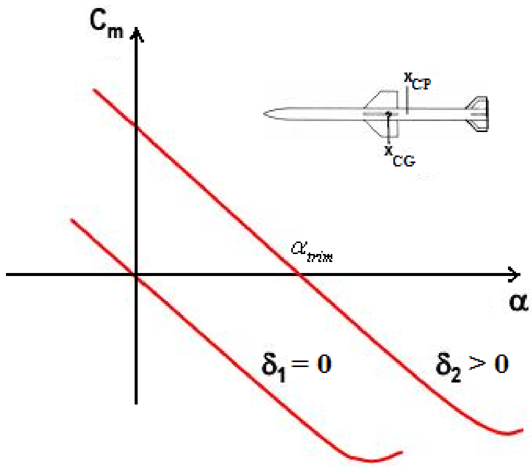


Figure 4.7 Stability Curve [8]

In Figure 4.7, the shift in the curve can be observed when the fin is deflected. Without giving any deflection to the fins, the curve passes through origin if the external geometry of the missile is symmetric. To operate the missile in a constant trim angle of attack, the restoring moment must be balanced with the moment created by fin deflections.

The stability might also affect the safety of separation of the missile from an air launch platform. Since the autopilot is not activated yet at the beginning of separation, the missile should be sufficiently stable in order to reject the disturbances which might occur just after separation. The restoring pitching moment is higher when the missile has more static stability margin. This brings an upper limit $C_{m\alpha}{}^{u}$ to the derivative of pitch moment coefficient.

However, an excessive stability makes the missile to response slower to the autopilot commands. In early design stages, this concept should be taken into account. Therefore, a constraint on the lower limit of stability margin $C_{m\alpha}{}^{l}$ can be put. As a result, the stability is taken as a constraint of which bounds are given as:

$$0 \geq C_{m\alpha}{}^{u} \geq C_{m\alpha} \geq C_{m\alpha}{}^{l} \tag{4.18}$$

4.3 Constraints on Geometry of a Missile

4.3.1 Launch Platform Compatibility

The launch platform integration sets some constraints on the missile that must be considered early in the development process. In a few cases it may be possible to modify a launch platform to accommodate a new missile; however this is not an option in most cases. For example, some weapons are modified to a compressed carriage configuration such as clipped wings and tails or deployable wings to better accommodate in launch platforms. Since the platform has a limited space, the missile should fit into a maximum length, diameter, and span limits.

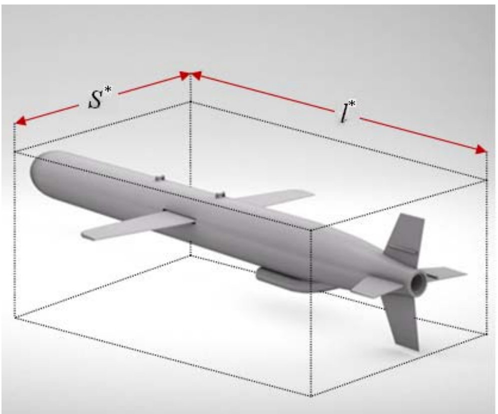


Figure 4.8 Launch Platform Space Limitations

According to Figure 4.8, there are four constraints due to platform compatibility which are given below:

$$LN + BL \le l^* \tag{4.19}$$

where LN is the nose length, BL is the body length of the missile and l^* represents the maximum total length of the missile.

$$BD \le BD^* \tag{4.20}$$

where BD represents the diameter of main body of the missile whereas BD^* represents the maximum diameter.

$$BD + 2(S1) \le S^* \tag{4.21}$$

where S1 is the span length of the first fin set and S^* is the maximum length between tips of the fin set.

$$BD + 2(S2) \le S^* \tag{4.22}$$

where S2 is the span length of the second fin set.

4.3.2 Structural Design

Buckling and bending moments are considered in the structural design of tactical missiles. The fineness ratio is defined the ratio of length to diameter that is very important for determining structural constraints. The typical interval of missile body fineness ratio is given as [8]:

$$5 \le \frac{l}{d} \le 25 \tag{4.23}$$

where l is the total length of the missile and d denotes the diameter of the missile.

Although a high fineness ratio reduces drag force, it has several disadvantages. As the missile body is thinner, the buckling risk is getting larger. In addition, the body bending frequency becomes high for a missile that has a low fineness ratio as seen in Figure 4.9.

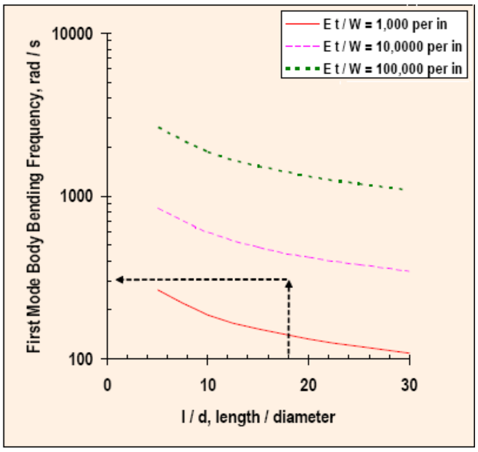


Figure 4.9 First Mode Body Bending Frequency [8]

Increasing body bending frequency improves the flight control characteristics. For example, the first mode of bending frequency should be at least twice the actuator frequency as a rule of thumb in order not to affect the actuator [8]. That brings a constraint which means there must be an upper limit of fineness ratio due to the considerations of bending moment frequency.

$$\frac{BL + LN}{BD} \leq \left(\frac{l}{d}\right)^{*} \tag{4.24}$$

4.3.3 Subsytem Constraints

The geometric constraints arisen from subsystems placement inside a missile should also be taken into account. In this thesis, the constraints due to motor section is only studied which creates a geometrical constraint. The structural case of motor must be such that the engine should fit in it. Therefore, the diameter of body in the aft has to be greater than the maximum diameter of motor. Since it is assumed that there is no diameter change between aft and main body, the constraint can be in the form of

$$BD \geq \left(d_m\right)_{\max} \tag{4.25}$$

where $\left(d_m\right)_{\max}$ denotes the maximum diameter of the motor.

CHAPTER 5

OPTIMIZATION MODEL

The main purpose of applying an optimization method in the conceptual design phase of a missile is to find the particular parameter set of missile's external geometry that maximize performance, or minimize weight or cost as while satisfying a set of design, operational, and economical constraints. Note that economical constraints are not included in this thesis study. As seen in Figure 5.1, there exists an optimization iteration cycle involving three main modules: User Interface, Optimization Model and Flight Simulator. The optimization module cooperates with the flight simulator and a user interface. Module 2 is a graphical user interface (GUI) in which the objectives and constraints of the external configuration optimization problem can be set by the designer. Also, the flight mission plan which is the input of the flight simulator module is defined in the user interface module.

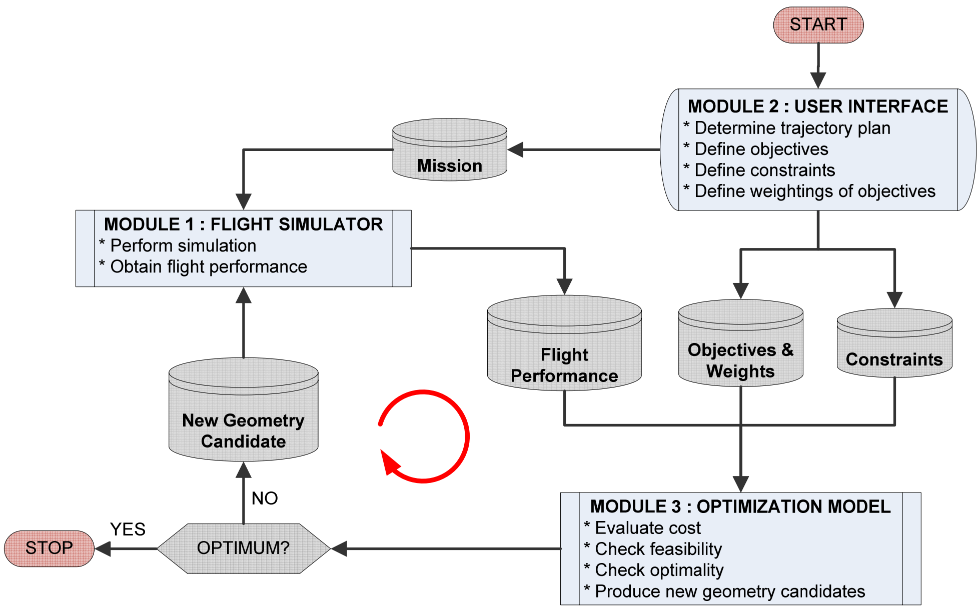


Figure 5.1 Optimization Cycle

The optimization cycle shown in Figure 5.1 is to be initiated by the designer by defining the requirements and the mission plan via a graphical user interface. By means of this GUI, the optimization module is noticed about the performance measures to be taken as objectives and their relative importance. Also, constraints on performance measures or geometric dimensions or configurations are determined.

The optimization model is fed with the flight performance values of each external geometry candidates by means of the flight simulator. Hence, the flight simulator runs in every single iteration of the optimization cycle. Module 3 computes the cost of each external geometry candidates according to its performance data and checks the feasibility of the geometry according to the constraints defined in the user interface. The iterations are stopped once the optimality conditions are satisfied and the current geometry will be declared as the optimum. Otherwise, new geometry candidates are generated and iterations will be continued.

Optimization parameters consist of a set of unknowns which affect the value of the objective function. In the external configuration optimization problem in missiles, there are two kinds of parameters; namely, external configuration parameters and geometrical dimensions mentioned in Chapter 4.

5.1 Objectives to be Minimized or Maximized in the Cost Function

In optimization problems, a cost function measures how good a particular solution of the problem is; the lower its value the better the solution is in minimization type problems. In external configuration optimization problems, cost function is composed of the terms involving the measures of merits to be minimized or maximized which are expressed in Chapter 4. Since there might be more than one objective according to the inputs defined in the user interface module, the problem can be called a multi-objective optimization problem. However, the problem with multiple objectives can be reformulated into a single-objective problem by forming a weighted combination of the different objectives.

Some of the measures of merit can be either cost or constraint according to the requirements of the design such as range and speed listed in Table 5.1. For example, the requirement on range can be set as an operational range interval as well as it can be stated as an objective to be maximized. Since there must be at least one objective in the optimization problem, the default objective term in the cost function is coming from mass minimization in this study. Since maximizing a cost means minimizing the negative of that cost, all of the objectives can be expressed in the same cost function as terms to be minimized.

Table 5.1 Status of Measures of Merit

#	**Measures of Merit**	**Status**
1	Launch Mass	To be minimized as a default
2	Operational Range	Either to be maximized or taken as a constraint
3	Cruise Speed	Either to be maximized or taken as a constraint

5.2 Equality and Inequality Constraints

While designing the external geometry of a missile, there are always some constraints that restrict the scope of the problem and determine the boundaries of feasible region. These constraints can be due to launch platform limitations and minimum desired performance requirements. The constraints in the external configuration optimization of a missile can be divided into two groups, which are performance constraints and geometrical constraints. All of the performance constraints are nonlinear constraints which cannot be expressed by a linear combination of design parameters. However, geometric constraints including bound limit of each dimension are mostly linear. A classification of constraints is listed in Table 5.2.

Table 5.2 Classification of Constraints

Constraints	Name	Type	Status
Linear	Bounds	Inequality	Default
	Launch Platform	Inequality or Equality	Optional
	Structure	Inequality	Optional
Nonlinear	Range	Inequality	Optional
	Speed	Equality	Optional
	Control Effectiveness	Inequality	Optional
	Stability	Inequality	Optional
	Maneuverability	Inequality	Optional

As mentioned in chapter 4, there are mainly 2 geometrical constraints on the diameter and length of the missile due to launch platform integration and structural design. The designer can define either some of the constraint listed in Table 5.2 or all of them according to the requirements of the specific problem. For linear constraints, there must be an interval on each geometric parameter in order to determine the scope of the dimensions. These are called bound constraints. An example for them can be as follows: "Only the missiles with length between 1 to 6 meters are to be investigated in the conceptual design of a specific missile". Therefore, the parameters $\overline{x}$ to be optimized should be chosen between their upper $\overline{x}^u$ and lower $\overline{x}^l$ limits as

$$\overline{x}^l \le \overline{x} \le \overline{x}^u \tag{5.1}$$

Bound constraints also help to define the interval of interest. All of the bound constraints on the geometric dimensions can be defined in the in the user interface

Nonlinear constraints which involve the performance requirements mostly are stability, control effectiveness, maneuverability, range and speed mentioned in

Chapter 4. Their values are obtained only after running flight simulation or executing Missile DATCOM. Therefore, there is no chance to produce a new geometry candidate which satisfies nonlinear constraints before running the flight simulation. On the contrary, the candidates linearly feasible can be easily produced without running the flight simulation. Unless the range and speed are set as objectives to be maximized, the values of operational range interval and average cruise speed must be defined as constraints in the user interface. In addition to the range and speed constraints, the designer might put additional constraints on the control effectiveness, maneuverability and stability according to the requirement of the specific problem.

5.3 Genetic Algorithm

Genetic Algorithms (GA) are search methods based on principles of natural selection and genetics. They have been applied successfully to numerous problems in business, engineering and science. In many practical applications, GA finds good solutions in reasonable amounts of time [5]. There are other alternatives of search algorithms such as conventional methods. However, conventional techniques are not preferable for complex problems such as external geometry sizing optimization problem since they are not global optimum search algorithms. Therefore, the use of genetic algorithm is considered to be more practical and effective method in this study.

GA is a stochastic search technique based on the mechanism of natural selection and natural genetics. GAs, differing from conventional search techniques, start with an initial set of solutions called population. Each individual in the population is called chromosome, representing a solution to the problem at hand. A chromosome is a string of symbols, it usually, but not necessarily, a binary bit string. The chromosomes evolve through successive iterations, called generations. During each generation, the chromosomes are evaluated, using some measures of fitness. To create the next generation, new chromosomes called offspring, are formed by either merging two chromosomes from current generation using a crossover operator or modifying a chromosome using a mutation operator. A new generation is formed by selecting, according to the fitness values and rejecting others so as to keep the population size constant. Fitter chromosomes have higher probabilities of being selected. After several generations, the algorithms converge to the best chromosome, which hopefully represents the optimum or suboptimal solution to the problem [9].

In the conceptual design of a missile, there might be more than one objective (multi-objective) which deteriorates the convexity of the optimization problem. A function is convex if and only if the region above its graph is a convex set; and, a set is convex if, given two points in the set; every point on the line segment joining these two points is also a member of the set [11]. An example of non-convex function is shown in Figure 5.2:

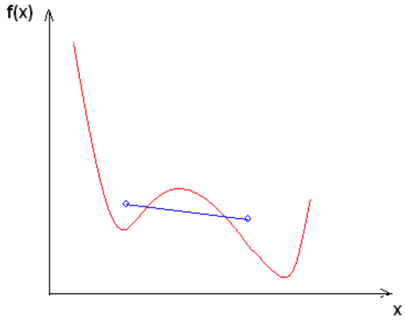


Figure 5.2 An Example of Non-convex Function

In case of non-convex cost functions, the conventional methods may fail to find out the global optimum because there is more than one local optimum. Since conventional search methods stop once they reach one of the local minima and the algorithm will have no idea about the other local minima. As a result, they fail to discover the global minimum. However, the genetic algorithm is capable of escaping local minima when they are reached and it still continues to search for other local minima by escaping mechanisms called mutations. The mutation operators lead the search point to jump randomly into another point in the feasible region once approaching to a local minimum and the algorithm continues to search from new point.

5.3.1 Fitness Evaluation

A fitness function is a particular type of objective function that quantifies optimality of a solution in genetic algorithm so that a particular chromosome may be ranked against other chromosomes. That is, it is the same with cost function f to be minimized in this problem. The mathematical representation of the problem is

$$\begin{array}{ll} \text{Minimize} & f(\bar{x}) \\ \text{Subjected to} & \bar{h}(\bar{x}) = 0 \\ & \bar{g}(\bar{x}) \leq 0 \end{array} \tag{5.2}$$

where $\bar{h}$ denotes the vector of equality constraints whereas $\bar{g}$ represents the vector of inequality constraints of the problem. Also, the vector $\bar{x}$ represents the optimization parameters.

5.3.1.1 Normalization of the Cost Terms

While converting a multi-objective optimization problem into a single objective one, each cost term in total cost expression must be converted into a non-dimensional form with respect to one another. Therefore, a multi-objective cost function can be written as

$$f(\bar{x}) = f_1(\bar{x}) + f_2(\bar{x}) + \ldots + f_n(\bar{x}) \tag{5.3}$$

where f_i $(i = 1, 2, .., n)$ denotes the cost of a single objective.

Since the units of each cost term are different such as kilogram for mass, kilometers for range, the comparison of the cost terms cannot be possible without any normalization. A general trend is to aim reducing the value of each cost terms into an interval of [0 1]. For the normalization, the desired values or desired interval limits of the performance objectives defined by the user can be used for normalization. For example, if the designer defines a desired upper and lower limit for an objective such as operational range between 200 and 250 km, desired minimum and maximum limits for the range term of the cost function are 200 and 250 km. In order to guarantee that the resulting cost value is closer to the interval of [0 1], the normalized cost function f_N is expressed as

$$f_N(\bar{x}) = \frac{f_1(\bar{x}) - f_{1_{\min}}}{f_{1_{\max}} - f_{1_{\min}}} + \frac{f_2(\bar{x}) - f_{2_{\min}}}{f_{2_{\max}} - f_{2_{\min}}} + \ldots + \frac{f_n(\bar{x}) - f_{n_{\min}}}{f_{n_{\max}} - f_{n_{\min}}} \tag{5.4}$$

where $f_{i_{\max}}$ and $f_{i_{\min}}$ $(i = 1, 2, .., n)$ represent the maximum and minimum desired values of each cost term defined by the designer.

As an alternative method to the desired values or upper limit of the desired interval of objectives, the performance values of the similar missiles in the literature may be used as the reference values for normalization. That is, the missiles which have similar mission profile and performance with the missile to be designed could be used for the normalization. In order to make them non-dimensional, every cost term should be divided by these baseline reference values as

$$f_N(\bar{x}) = \frac{f_1(\bar{x})}{f_1^*} + \frac{f_2(\bar{x})}{f_2^*} + \ldots + \frac{f_n(\bar{x})}{f_n^*} \tag{5.5}$$

where f_i^* $(i = 1, 2, .., n)$ denotes the performance values of the similar baseline missile. Since the AGM-84 type of Harpoon is used as a baseline missile for the case study of this thesis, the performance values of it is known already; therefore, this method is considered to be more suitable in this thesis study than the method with pre-defined desired upper and lower limits.

5.3.1.2 Weighting the Cost Terms

As the aerospace engineering advances, it becomes impossible to develop a missile to fully satisfy all of the requirements defined by the customer simultaneously. Hence, the designers have to deal with some tradeoffs during design which is forced by conflicting requirements with one another [6]. In cases with too many conflicting constraints, it might be infeasible to design such a missile which satisfies all the requirements. Therefore, the designer needs a method to determine specific requirements which are of high priority. In order to quantify the tradeoffs between conflicting mission requirements, the designer should optimize the missile parameters based on relative weightings of requirements.

The weightings can be considered as the importance of the objectives with respect to one another. These weightings are directly used as multiplicative coefficients for each term in the cost expression. Since the importance sequences are highly dependent on the customer's will, the user needs to define these priorities by grading each objective as G_i in the user interface. These grades given over 5 in the GUI developed in this study are converted into weightings such that their sum will be unity; i.e.,

$$w_1 + w_2 + \ldots + w_n = 1 \tag{5.6}$$

where w_i $(i = 1, 2, .., n)$ denotes the weighting of ith cost term. The relation between grading G_i and weighting w_i is given as

$$w_i = \frac{G_i}{G_1 + G_2 + \ldots + G_n} \tag{5.7}$$

As a result, the final shape of the total cost function will be like:

$$f_{N,W}(\overline{x}) = w_1 \frac{f_1(\overline{x})}{f_1^*} + w_2 \frac{f_2(\overline{x})}{f_2^*} + \ldots + w_n \frac{f_n(\overline{x})}{f_n^*} \tag{5.8}$$

where $f_{N,W}$ is the normalized and weighted cost function.

5.3.1.3 Penalty Method

The infeasibility of a chromosome originates from the nature of constrained optimization problem. For many optimization problems, a feasible region can be represented as a system of inequalities and equalities (linear or nonlinear). For such cases, many penalty or barrier methods were proposed in order to handle infeasible chromosomes. In constrained optimization problems, the optimum typically occurs at the boundary between feasible and infeasible areas. The penalty approach will force the genetic search to approach the optimum from both feasible and infeasible regions [9].

Penalty and barrier methods are two procedures for approximating constrained optimization problems by unconstrained problems. The typical penalty and barrier functions for an inequality constraint S are given respectively:

$$S = \{\overline{x} : g_i(\overline{x}) \leq 0, \;\; i = 1, 2, ..., p\} \tag{5.9}$$

where p is the total number of inequality constraints and $g_i \;\; (i = 1, 2, .., p)$ denotes the i[th] inequality constraint in an optimization problem. The general expressions of penalty and barrier functions (P and, B respectively) are given as

$$P(\overline{x}) = \frac{1}{2}\sum_{i=1}^{p}\left(\max[0, g_i(\overline{x})]\right)^2 \tag{5.10}$$

$$B(\overline{x}) = -\sum_{i=1}^{p}\frac{1}{g_i(\overline{x})} \tag{5.11}$$

The main difference between these two methods is the initial points of the search. In the barrier method, the initial solution must be in feasible region in order to obtain convergence whereas there is no need to start with a feasible point in the penalty method [11]. That is, although the penalty method has talent to bring solution inside feasible region during search, the barrier method can never achieve this if the starting point is infeasible as illustrated in Figure 5.3.

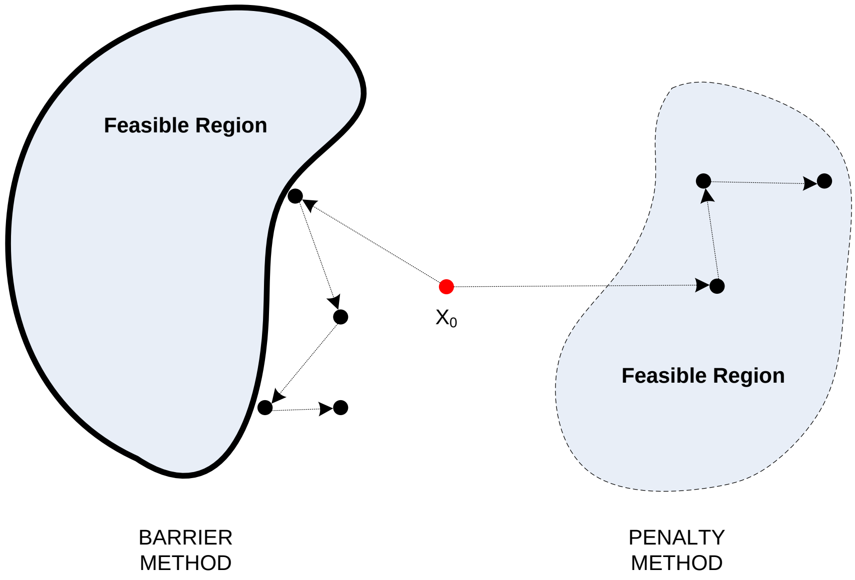


Figure 5.3 Search by Penalty and Barrier Methods

In the external geometry sizing of a missile, there are both linear and nonlinear constraints. Although the initial chromosomes are adjusted such that they are linearly feasible*, it is very difficult to choose a chromosome that ensures to satisfy all the nonlinear constraints such as range, control effectiveness, maneuverability, etc. Without running a simulation, there is no way to determine whether a chromosome satisfies the nonlinear constraints. Therefore, it is almost impossible to start with an initial solution both nonlinearly and linearly feasible at the same time. Note also that, if the simulation is run in order to check the nonlinear feasibility of each chromosome produced, there would be a waste of time.

As a result, the penalty method is concluded to be the most suitable method for handling nonlinear constraints in such complex optimization problems. The new form of the fitness function after adding penalty cost term $f_{N,W,P}$ with a penalty weight λ is given as

$$f_{N,W,P}(\overline{x}) = f_{N,W}(\overline{x}) + \lambda P(\overline{x}) \tag{5.12}$$

* In this thesis, a solution is referred to as "linearly feasible" if all linear constraints are satisfied and "non-linearly feasible" if all non-linear constraints are satisfied.

Assuming that the number of equality constraints is r and the number of inequality constraints is p, the general expression of penalty function is given as

$$P(\bar{x}) = \frac{1}{2}\sum_{i=1}^{p}\max[0, g_i(\bar{x})]^2 + \frac{1}{2}\sum_{k=1}^{r}\left|h_k(x)\right|^2 \tag{5.13}$$

However, it is more convenient to use an absolute-value penalty function in order not to deteriorate the degree of magnitude balance between objective term and penalty term. That is, the penalty terms should be in degrees of one since the objective terms of the cost function are already in degrees of one. The general form of the absolute-value penalty function is

$$P(\bar{x}) = \sum_{i=1}^{p}\max[0, g_i(\bar{x})] + \sum_{k=1}^{r}\left|h_k(x)\right| \tag{5.14}$$

There are different ways to choose a penalty weight in this problem. In case of a very large penalty weight, the search algorithm may converge just after the feasibility is guaranteed. This is called premature convergence [11]. Penalty weights directly affect the final geometry of the missile since they harmonize the dominancy between objectives and constraints.

5.3.2 Encoding Method

There are mainly two main components of genetic algorithms that are problem dependent; namely, encoding and the evaluation of fitness (cost) function. In this section, how to encode the configuration parameters is to be explained. The goal is to set to the various parameters so as to optimize some output.

Since the entire configuration parameters of missile listed in Table 4.2 are discrete or combinatorial, the only possible values for them can be selected within a finite element set. Therefore, the first assumption that is typically made is that the parameters can be represented by bit strings. This means that the parameters are discrete sized, and range of the discretization corresponds to some power of 2. For example, 10 bits per parameter means that a range of 1,024 discrete values is obtained for this parameter.

However, there is a constraint between the fin configuration and control type parameters as they are dependent ones. For example, when we select 024 configurations, it is not possible to choose canard control option because there is no canard. Therefore, these two parameters can be merged into single parameter in order to get rid of defining an extra constraint between the control type and fin configurations. In Table 5.3, the minimum number of bits is given in order to represent each configuration parameter in a bit string.

Table 5.3 Encoding Configuration Parameters

Configuration Parameter	**Possible Values Set**	**# of elements**	**Min # of bit string**
Fin & Control	{404-Canard, 404-Tail, 024-Tail,044-Tail}	4	2 (2^2)
Nose Shape	{Ogive, Conical, Power, Haack, Karman}	5	3 (2^3)
Panel Orien.	{Plus, Cross}	2	1 (2^1)

Since the number of possible values of configuration parameters does not match the total number of values that representative bit string can take, especially in fin nose shapes, there remains some excess bit patterns. If a parameter can only take a finite set of values of which number is not equal to the power of 2, the coding becomes a bit difficult issue. As a specific example, assume that the nose shape parameter can take 5 different values; however, the representative bit string can take 2^3=8 different values. There exist 3 excess unnecessary bit patterns which may result no evaluation in fitness function.

Note that for the geometric dimensional parameters in Table 4.3, the discretization is not a problem since they are continuous parameters. The only problem that might arise is that whether the discretization will provide enough resolution to obtain a desired level of precision [16]. Although a bit string encoding is suitable for continuous geometric dimensional parameters, there are some difficulties in encoding combinatorial configuration parameters.

For applying the genetic algorithm for the external configuration optimization, MATLAB Genetic Algorithm Toolbox is used in this study. However, MATLAB Genetic Algorithm Toolbox does not let putting constraints if the bit string representation is selected. As a result of these difficulties, it is necessary to design another encoding method other than bit string.

In MATLAB Genetic Algorithm Toolbox, the population type is allowed to define only in two types; double vector or bit string. In this study, there is no chance to choose other than the double vector type of chromosomes. This implies that the real number coding for constrained optimization should be used. However, randomly chosen real values do not conform to the configuration parameters because they can take only certain numbers. Therefore, it is needed to represent them with integer coding whereas the geometric dimensions are represented by real number coding shown in Figure 5.4.

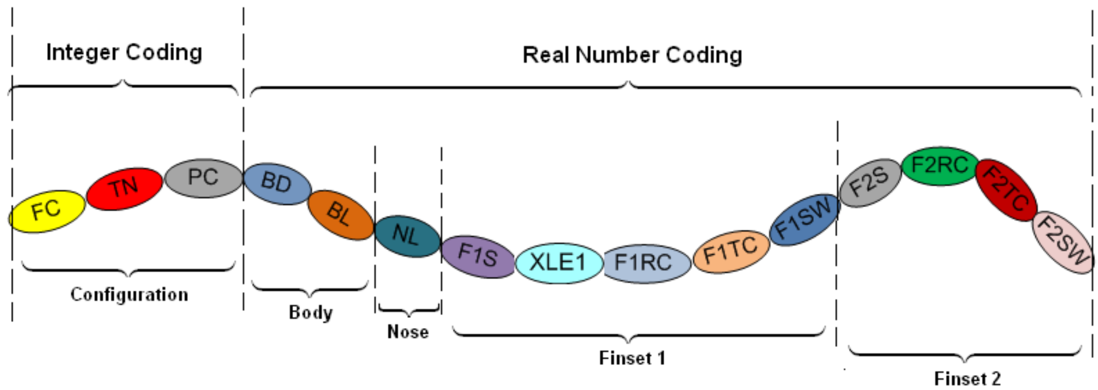


Figure 5.4 Chromosome Representing External Configuration and Geometric Dimensions of a Missile

Note that there is no gene representing the center of mass (XCG) and the leading edge location of the tail (XLE2) mentioned in Table 4.3. Since the center of mass depends on the internal component installation, it is assumed that center of mass is adjusted such that it is always between the intervals of 45-50% of total length of the missile. Also, the trailing edge of tail is assumed to be always at the base of the missile since there is no boat tail design in this study. As a result, these two parameters are assumed to be dependent of other parameters and excluded from optimization parameters. The related equations are given:

$$XCG \in [0.45(BL + NL), 0.50(BL + NL)] \tag{5.15}$$

$$XLE2 = BL + NL - RC2 \tag{5.16}$$

where NL, BL, and RC2 denote the nose length, main body length, and root chord length of the second finest, respectively.

In addition, if the user defines some equality constraints on a single dimensional parameter, this parameter is excluded from the chromosome because it is already set to a constant value. By means of that, the size of chromosome or optimization parameter vector is decreased which decreases the optimization time. If there are less parameters in the optimization, the convergence time shortens. In conclusion, it can be said that the length of the chromosome changes according to the equality constraints from one design to another.

5.3.3 Creation Function

In every optimization techniques, the process cannot be started without setting an initial point which can be expressed as the baseline geometry of missile in this specific optimization study. There are maximum 15 independent parameters on a chromosome representing external geometry and configuration of a missile. To initiate each parameter with a random number, default upper and lower limits of each parameter should be considered. In Figure 5.5, the general process of creation of the chromosomes is illustrated.

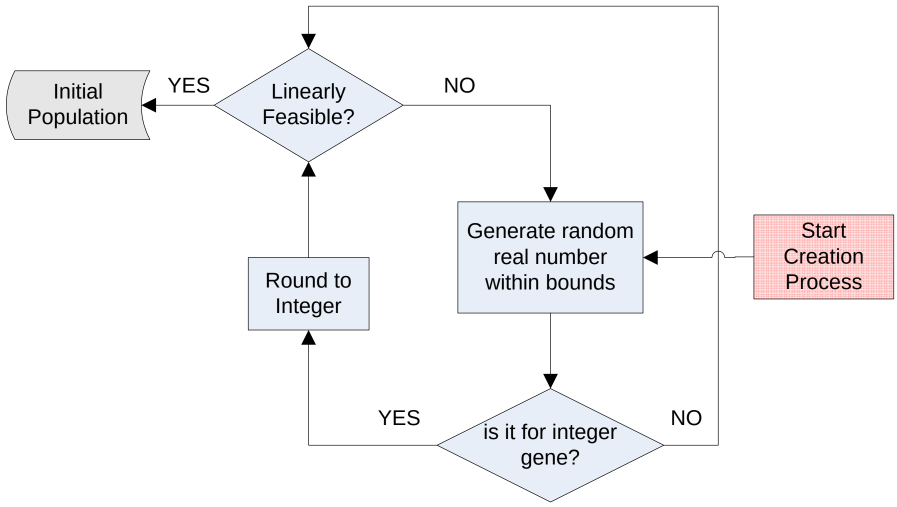


Figure 5.5 Creation Process

Some random integers for configuration parameters and some random real-valued numbers for geometric dimensional parameters should be set in the interval bounds. In this process, the random number generator command (`rand`) is used in MATLAB. At the end of creation process, the chromosomes are initialized such that they are inside their default bound limits.

As seen in Figure 5.5, a linear feasibility check is performed after the creation of each chromosome. In case of linear infeasibility, the creation process is restarted until linear constraints are satisfied. That is, the baseline geometrical parameters should be chosen so that they are inside the interval of linear constraints. By means of satisfying linear and bound constraints in creating the chromosome at the beginning, both the total iteration number and optimization time are decreased. Since the nonlinear constraints are handled by penalty function method, there is no need to produce nonlinearly feasible children.

Note that MATLAB Genetic Algorithm Tool does not have a default creation function which can handle both discrete and continuous parameters; therefore, a problem-specific custom creation function is written.

5.3.4 Crossover Function

Crossover is the process of combining or mixing two different individuals in a population [13]. The crossover operator includes three types of crossovers; namely, single-point, two-point, and scattered crossovers as seen in Figure 5.6.

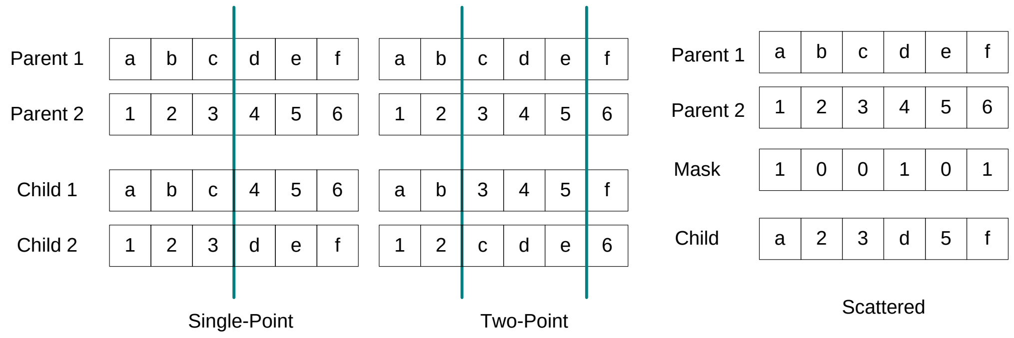


Figure 5.6 Crossover Types

Comparing the performance of scattered crossover with single and two-point crossover, genomes that are near each other tend to survive together whereas genomes that are far apart tend to be separated in single or two-point crossover. However, this effect is eliminated by the technique of scattered crossover in which each gene in a chromosome has an equal chance of coming from either parent. This increases the diversity of the population which helps to create feasible children especially in constrained optimization. Therefore, it enables the genetic algorithm to converge faster and produce better solution.

In the external configuration optimization problem, the scattered crossover function is used. Its mask array is composed of a random binary vector to select the genes. If a binary number is 1, this function selects the gene from the first parent otherwise from the second parents. By means of this, the method combines the genes to form a child [10]. The process of crossover is shown in Figure 5.7:

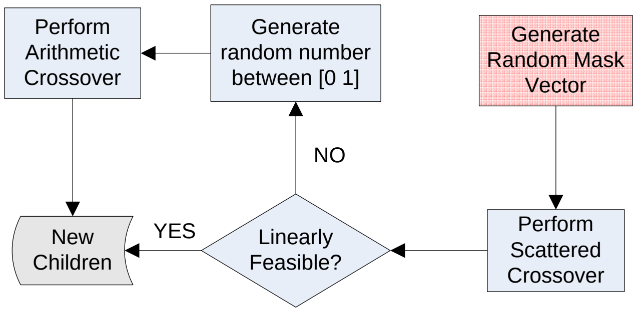


Figure 5.7 Crossover Process

The next step is to check the linear feasibility of each child after reproduction by scattered crossover as seen in Figure 5.7. Instead of repeating the scattered crossover in case of an infeasible child is produced, an arithmetic crossover method is used which guarantees the linear feasibility of a child. This method is based on the idea that the linear combination of two feasible solutions is also linearly feasible. The following equations express how to obtain two children $\overline{x}_{C1}$ and $\overline{x}_{C2}$ from a parent $\overline{x}_1$ and $\overline{x}_2$:

$$\overline{x}_{C1} = \upsilon \overline{x}_1 + (1-\upsilon)\overline{x}_2 \tag{5.17}$$

$$\overline{x}_{C2} = (1-\upsilon)\overline{x}_1 + \upsilon \overline{x}_2 \tag{5.18}$$

The factor υ is a random weighting number which is chosen within [0 1] before each crossover process. If the scattered crossover was repeated until the linear feasibility is satisfied, there would be too much time consumed. Note also that, the arithmetic crossover method might bring non-integer values for configuration parameters; it is only applied on genes representing geometrical dimensions.

5.3.5 Mutation Function

Mutation is the genetic operator that alters one or more gene values in a chromosome from its initial state. This can result in entirely new gene values being added to the gene pool. With these new gene values, the genetic algorithm may be able to arrive in a better solution that was previously possible. In addition, the mutation is also important part of a genetic search as it helps to prevent the population from stagnating at local optima.

The mutation occurs during evolution according to a user-definable mutation probability. MATLAB Genetic Algorithm Toolbox selects it as a default value of 0.01. If it is set to be too high, the search will turn into a primitive random search whereas algorithm has a premature convergence at any local optima if it is chosen to be very low.

The mutation process particular to the external configuration optimization problem is shown in Figure 5.8:

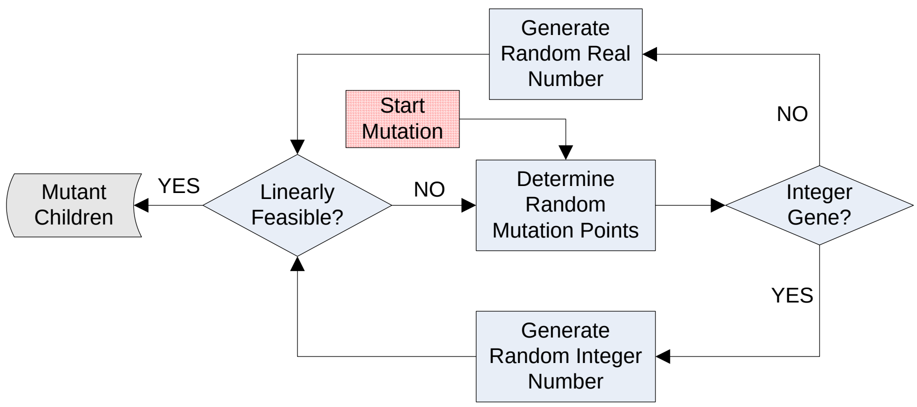


Figure 5.8 Mutation Algorithm

In the conceptual design optimization, the uniform mutation algorithm is used. It is a mutation operator that replaces the value of the chosen gene with a uniform random value selected between the user-specified upper and lower bounds for that gene. This mutation operator can only be used for integer and float genes. Since there is no chance to represent integer and float genes in the same chromosome in MATLAB Genetic Algorithm Toolbox, a custom mutation operator is developed in this study which questions whether the mutation gene is for discrete configuration parameters or continuous geometric dimensions. The mutation process cycle shown in Figure 5.8 is repeated until the linear feasibility is satisfied.

For other genetic operators except mutation, crossover, and creation functions, the default type of methods which is offered in MATLAB Genetic Algorithm Toolbox are used without any modifications.

5.3.6 Hybrid Algorithm

GAs have been used successfully in order to explore optimal or near-optimal solutions for a wide variety of optimization problems. Although GAs often gives good results in global search, but they are relatively slow in converging to a local optimal. On the other hand, the local improvement methods, such as gradient-based (line search, LS) procedures, can achieve to find the local optimum in a small region of the search space, but they are typically poor in a global search [9].

Therefore, a hybridization strategy was suggested to improve performance of simple GAs. In order to ensure that the solution found by simple GA method is optimum, the algorithm is switched to the hybrid method when the stop condition is satisfied at the end of GAs. Hybrid function is applied to the near-optimal offspring generated by GA in order to push it to a local optimum as shown in Figure 5.9. By means of that, the optimality of the final solution is guaranteed.

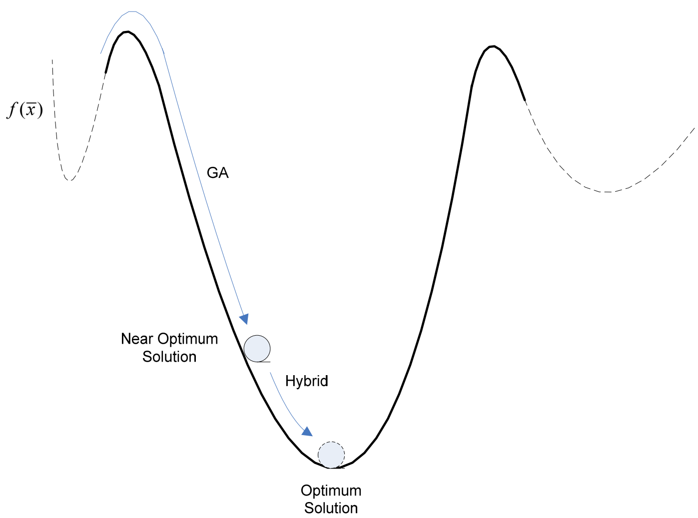


Figure 5.9 Hybrid Genetic Algorithm

CHAPTER 6

SOFTWARE DESIGN

A conceptual design tool named EXCON (External Configuration) is developed as a product of the thesis. The main purpose of implementing a tool is to minimize the time spent during the conceptual design of missiles in military projects. It also provides an automation of the process which decreases the human interference while designing a missile. Rapidly eliminating undesirable geometry configurations by using EXCON, the detailed design stage would start earlier shortening the total design period. EXCON is an integrated software tool which has a user interface that the users can define design requirements and constraints as well as the mission profile. It also implements the genetic algorithm for optimization and a flight model which is supported by an aerodynamic coefficient prediction package, Missile DATCOM. The conceptual design by EXCON is performed by four basic actors which are graphical User Interface (GUI), Optimization, DATCOM Processor and Simulation. The sequence of operations in initialization, iteration and finalization phases is illustrated in Figure B. 7 of Appendix B.

In the initialization phase, the user submits the minimum set of inputs which are sufficient to start an optimization. Also, the genetic algorithm parameters are initialized as well as creating initial population in this phase. In the iteration phase, the genetic algorithm tries to find better solution for the external geometry optimization problem. It collects the flight performance of each geometry candidates and evaluates the fitness of them. Once any of the stop conditions mentioned in Table 6.3 is satisfied, the iteration phase is terminated and finalization begins. In this phase, the geometric dimensions and configuration of the optimum solution is printed out in the user interface. Also, the user is informed about the optimum thrust profile.

6.1 Conceptual Design Tool Skills

In this thesis, the conceptual design tool intended has a scope of designing a mid-range, with a turbojet or solid fuel rocket motor. Since the optimization is trajectory dependent, there are some types of trajectories is studied in the scope of EXCON which might be air-to-surface, surface-to-surface and air-to-air. However, most of the air-to-air missiles are high maneuverable supersonic ones whereas air-to-surface and surface-to-surface missiles can have speeds in subsonic regions. Note that the reliability of the outputs produced by EXCON is limited by the reliability of the Missile DATCOM. Therefore, users of the EXCON should be aware that Missile DATCOM gives relatively more reliable aerodynamic coefficients' outputs in subsonic regions than in supersonic and transonic regions. In case that a detailed Computational Fluid Dynamics (CFD) package were used such as Fluent which analyzes and estimates aerodynamic coefficients instead of Missile DATCOM, the reliability. There is one disadvantage of using such an advanced software package: it requires much more time to analyze and predict aerodynamic coefficients compared to Missile DATCOM. Therefore, it is more convenient to use them in more detailed design stages following the conceptual design.

In addition to the missiles which have either turbojet or solid fuel rocket engines, there are some missiles which have multi-stage thrust system with a booster and turbojet as a sustainer in the same flight. They are usually launched as surface to surface. The booster capsule is burned in order to give a sufficient altitude to the missile. The turbojet system is activated at the beginning of launch. The turbojet motor is put into use when the solid fuel is run out at the end of boost phase. In EXCON, the hybrid motor is not modeled. Instead of using a multi-stage motor, the boost effect, which is created by the solid fuel motor at the beginning of the flight, is modeled by defining an initial speed to a missile having only a turbojet motor. By means of that, the hybrid motor alternatives can also be checked in the conceptual design by EXCON.

Other than the mission profile and motor constraints explained in the previous paragraphs, there are default bounds of geometric dimension which determines the scope in which EXCON works. The bounds of the geometrical dimensions and configuration should be clearly stated. The bound interval inputs such as designing the missiles with lengths between 1 and 6 meters and diameters between 0.1 and 0.7 meters should be set in EXCON. The geometric dimension bounds used in the case study are defined in Table A.2 in Appendix A.

6.2 Development Environment and Software Packages

There is an integrated development environment in EXCON which includes MATLAB toolboxes and other supporting packages. EXCON software includes four main sub-models which are

- user interface,
- optimization model,
- flight simulation, and
- DATCOM processor.

In this section, the toolboxes used for implementing these four main sub-models are to be explained.

6.2.1 MATLAB Toolboxes Used

In this section, MATLAB Toolboxes used in design and performance optimizations are explained.

6.2.1.1 SIMULINK

For the flight simulator, MATLAB SIMULINK toolbox is used. The details of SIMULINK block diagrams including aerodynamics, propulsion, EOM (Equations of Motion), autopilot, gravity and atmosphere are shown in APPENDIX B.

As the simulator runs every single iteration of optimization cycle, it should be as fast as possible. In order to improve the run time properties of the simulator, the profiler option of SIMULINK is used. By means of that, the designer can observe run time performance of SIMULINK model developed. The SIMULINK simulation profiler collects performance data while simulating the model and generates a report, called a simulation profile, based on the run time data. This simulation profile shows to the designer how much time SIMULINK spends on executing each function required to simulate the model. The profile enables designers to determine the parts of your model that require the most time to simulate and hence where to focus your model optimization efforts [17].

By observing the profiler results for the flight simulation model, one can state that the most of the simulation time was spent on interpolation of the aerodynamic coefficients.

Interpolation is handled by two different method and the performances of these two methods are compared according to their contribution to the total run time of the simulation. The first method is writing a code inside MATLAB Fcn SIMULINK block in order to handle interpolation of the aerodynamic coefficients from a look-up table. In the code, a built-in MATLAB function `interp2` is used. In the second method, the interpolation and pre-lookup blocks in SIMULINK library shown in Figure 6.1 are used in order to handle interpolation.

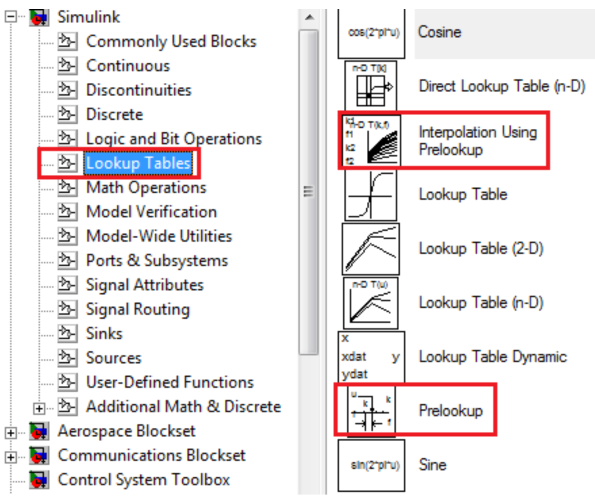


Figure 6.1 Look-up Tables in SIMULINK Library

For the case study missile, the simulation is run both for these two cases and the total simulation times are obtained in Table 6.1:

Table 6.1 Total Simulation Time According to Interpolation Type

#	**Interpolation with**	**SIMULINK Run Time [s]**
1	`interp2`	21.38
2	SIMULINK Look-up Tables Blocks	4.94

As seen in Table 6.1, the total simulation time is dramatically decreased in the second case. That is, the simulation is almost 5 times faster without calling `interp2` function inside SIMULINK. As a result, the Look-up Tables blocks are prefered to be used in the SIMULINK in order to improve the simulation speed.

6.2.1.2 Genetic Algorithm Toolbox

In order to handle the optimization, MATLAB Genetic Algorithm functions are mostly used in this study. The main function which is built in MATLAB is `ga` and it implements the genetic algorithm at the command line to minimize an objective function. However, since the mutation, crossover and creation operators used in genetic algorithm are specific to one problem to another; some custom operators are coded considering the needs of external configuration optimization problem. As a hybrid algorithm, `fmincon` is used since the problem is constrained optimization. `fmincon` attempts to find minimum of the fitness function starting at a final solution generated by GA. It uses the techniques which is generally referred to as constrained nonlinear optimization or nonlinear programming. The details of these operators are mentioned in Chapter 5.

In addition to the MATLAB built-in functions, several additional functions are written in order to handle the interaction between genetic algorithm and user interface. Since some of the constraints and the mission profile are determined by the user as the requirements of the design; genetic algorithm parameters such as linear constraints' parameters, chromosome length should be prepared according to the user inputs. Therefore, the additional functions listed and explained in Table 6.2 are written.

Table 6.2 Additional Functions Used in Genetic Algorithm

Function Name	**Explanation**
`getucon.m`	Gets user constraints on geometry and configuration.
`createGenePool.m`	Initializes all the genes within default bounds.
`createChromosome.m`	Selects the required genes in the pool for which the user does not define equality constraints. Appends selected genes tip to tip in order to compose a single chromosome.
`lincon.m`	Sets linear constraint's parameters A and b matrices such that $Ax < b$.
`getSimResults.m`	Runs the flight simulation for the current chromosome and obtains its flight performance data to be used in the evaluation of its fitness.
`evalFitness.m`	Evaluates fitness by using user-defined weights, normalization dividers and penalty weights.
`isFeasible.m`	Checks the linear feasibility of the chromosome.

Within the functions shown in Table 6.2, only `evalFitness.m` and `getSimResults.m` are called in every iteration of optimization whereas the rest of them are called only once at the beginning of optimization.

6.2.1.3 Guide Toolbox

MATLAB Guide toolbox provides the GUI (Graphical User Interface) design environment tools that allow the designer to create or edit GUIs interactively. It is used for the development of the user interface menus of EXCON.

EXCON software uses a GUI to enter inputs, control the analysis process, and view results. The GUI incorporates a variety of menus, scroll bars, buttons, and dialog boxes available. An example of EXCON GUI is shown in Figure A. 1 of Appendix A. The developed GUI is composed of mainly six panes which represents Launch Specs, Motor Specs, Flight Specs, External Configuration Specs, Objectives and Optimization Results.

Some of the inputs in panels are optional whereas others are obligatory to be entered. The GUI orients the user about the minimum set of inputs that should be submitted in panels. In case of incomplete or missing data, the GUI gives error messages after pushing the START OPTIMIZATION button, and the missing inputs will be highlighted. In addition, the popup menu lists are automatically updated by interacting with each other. For example, if air to surface launch type is chosen by the user, the flight phase's popup menu is updated such that there is no climb-glide phase sequence in the list. Also, some of the edit boxes may be disabled or enabled according to other inputs entered already. As a result, the consistency within inputs is controlled by GUI before starting an optimization.

User can also define the maximum process time in time limit editbox. In case of exceeding the time limit specified, the algorithm stops due to time limitation and the best geometry found so far will be taken as the optimum.

In the optimization results panel, the user is informed about the progress of the optimization process. The optimum geometry of each generation is printed in the left figure. Also, the convergence history can be seen in fitness vs. generations figure in the right. The minimum and mean fitness values of each generation are also printed in that figure. After the optimization is converged, the performance and external geometry data of optimum are shown as well as the elapsed time and the reason of termination. There might be several reasons of termination of optimization process listed in Table 6.3:

Table 6.3 Possible Termination Messages [18]

Message #	Termination Reason
1	Time limit exceeded
2	Stall time limit exceeded
3	No feasible point found
4	Maximum number of generations exceeded
5	Magnitude of step smaller than machine precision and constraint violation less than options.
6	The value of the fitness function did not change in options. Stall generation. Limit generations and constraint violation less than options.
7	Fitness limit reached and constraint violation less than options.
8	Average cumulative change in value of the fitness function over options. Stall generation. Limit generations less than options.

Also, the thrust-time profile of the optimum geometry during specified mission profile is plotted in the optimization results panel.

6.2.2 Supporting Packages

Missile DATCOM software package is used for estimating aerodynamic coefficients for a given geometry [4]. A batch run processor is coded for the executable version of DATCOM 5/97 produced by USAF (United States Air Force). This processor is developed in MATLAB and the main purpose of this script is to prepare proper input sets for DATCOM and read the output file produced by DATCOM. For two degree of freedom system, there is no need to read aerodynamic roll and yaw coefficients' output. This saves time elapsed during file reading operations. Following the execution of DATCOM and reading outputs, the processor creates 3D aerodynamic coefficients arrays which depend on Mach number and angle of attack. These coefficient arrays will be used as the aerodynamic look-up tables in the flight simulation.

Although using DATCOM as an aerodynamic coefficient estimator is a quick and economical way in preliminary design phase, there is disadvantage of using it. For certain geometries, it can produce positive drag coefficients which are impossible. Since the drag force is the resistance of air on a moving object, there is no way to produce a drag force which helps the motion. The reason for producing positive drag coefficients especially observed for the thick body missiles might be due to deficient experimental data used in semi-empirical formulae of DATCOM algorithms. Therefore, there must be a check mechanism after creating aerodynamic coefficients array shown in Table 6.4:

Table 6.4 Output Feasibility Check Functions of DATCOM Processor

Function Name	**Explanation**
`isPenaltyStability.m`	Checks if each element of $C_{m\alpha}$ array is negative. If not, it punishes that geometry by giving very large penalty without running simulation.
`isPenaltyDrag.m`	Checks if there is no negative element in C_D array. If not, it punishes that geometry by giving very large penalty without running simulation.

As seen in Table 6.4, the stability must be guaranteed by checking aerodynamic pitch moment coefficients derivative with respect to angle of attack $C_{m\alpha}$ as well as drag coefficients C_D. There is no need to simulate the flight with unstable geometries. Instead, a large penalty cost is given to unstable geometries in order to eliminate them quickly.

If the geometries which have positive drag coefficients are included to the optimization without penalizing, they can even mislead the optimization algorithm. Since positive drag plays a role like additional thrust, the total range of these missiles would be resulted higher than in real. This mishap acts like helping in maximization of range which is totally false. Therefore, it is crucial to check the consistency of axial coefficients' arrays of the missile geometries before using them in the flight simulation.

CHAPTER 7

CASE STUDY FOR THE CONCEPTUAL DESIGN

As a case study, one of an air to surface turbo-jet missile named AGM-84A Harpoon is re-designed for the verification of the conceptual design tool (EXCON). The design output is to be compared with the original missile's external configuration parameters. The mission profile, physical data, constraints and requirements for AGM-84A type of Harpoon are obtained from open sources in the internet. Note that, the external geometric dimensions which are not found from open sources are roughly measured from the Harpoon images in the internet. Also, the unknown performance data of Harpoon (stability, control effectiveness, maneuverability) is obtained by running the 2 DOF flight simulation of the EXCON.

As brief information, the Harpoon is an all-weather, over-the horizon, anti-ship missile system, developed and manufactured by McDonnell Douglas (Boeing Integrated Defense Systems). In 2004, Boeing delivered the 7000th Harpoon unit since its introduction in 1977 [23]. It is low-level missile with sea-skimming cruise capability, equipped with active radar guidance and warhead design to assure high survivability and effectiveness.

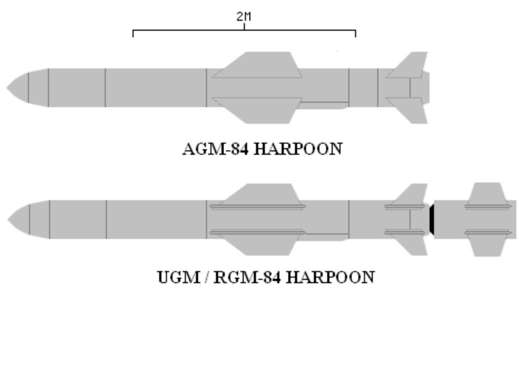


Figure 7.1 Harpoon with Booster and without Booster [24]

There are three types of Harpoon missiles according to their launch types. They can be launched from a shipboard, submarine and an aircraft. Except the aircraft launched types (AGM), there is an extra booster motor at the aft as shown in Figure 7.1 for submarine (UGM) or surface ship launched (RGM) versions. In this case study, only the air-launched AGM-84 version of Harpoon shown in Figure 7.2 is used.

Figure 7.2 AGM-84 Harpoon

7.1 Physical Constraints

The physical information of Harpoon is important since some data can be defined as the problem constraints in the user interface of the conceptual design tool. Since the information of launch platform integration is not known, there is no platform compatibility constraint like span, length, or diameter used during the case study. However, there are some physical constraints due to the engine used in Harpoon. Its engine specifications are given in Table 7.1:

Table 7.1 Physical Properties of Teledyne CAE J402 [27]

Properties	**Values**		**Unit**
Length	74.8		[cm]
Maximum Width	31.8		[cm]
Average Specific Impulse	≈ 2000		[s]
Thrust	Idle	500	[N]
	Max	2937	

The AGM-84 type of Harpoon is powered by a Teledyne CAE J402 turbojet engine [27]. Since the largest diameter of the engine is known, there is a constraint on the missile aft body diameter. That is, missile's diameter should be large enough such that a Teledyne CAE J402 engine can fit in it. Therefore, a minimum diameter constraint

$$BD \geq 0.318 \text{ [m]} \tag{7.1}$$

should be entered in the *External Configuration Specs* pane in the user interface.

Also, the specific impulse, maximum and minimum thrusts given in Table 7.1 must be entered as a motor capacity constraint in the *Motor Specs* pane of the GUI.

7.2 Operational Information

Operational information is necessary for constructing the mission profile of the Harpoon. The flight simulation will be performed according to a desired trajectory profile and flight conditions as well as launch conditions. The typical trajectory profiles of Harpoon are illustrated in Figure 7.3:

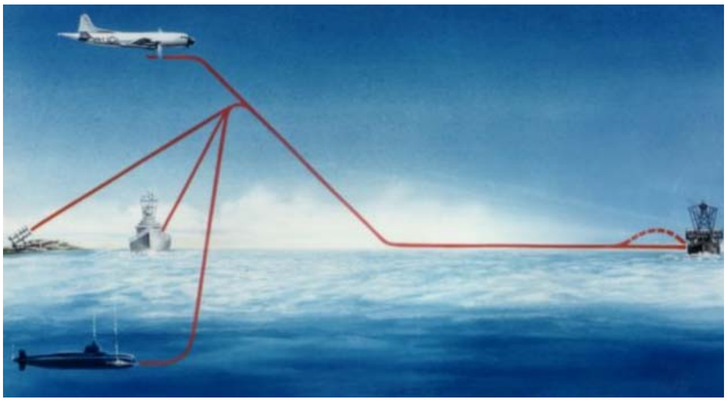


Figure 7.3 Possible Trajectories of Harpoon

The Harpoon is the only dedicated anti-ship missile in service with the U.S. Armed Forces. It has been developed into several advanced versions, including the SLAM (Stand-off Land Attack Missile) derivatives for high-precision attacks on land targets. Current U.S. platforms for the AGM-84 are the Navy's F/A-18, P-3C and S-3B and also a few B-52Hs of the USAF. The AGM-84E/H/K SLAM is currently used by the F/A-18 only [23].

Within the trajectory profiles illustrated in Figure 7.3, only the air to launch trajectory is focused in this case study. In this trajectory profile, there are glide, descent, cruise and climb phases are performed and the phase combination is typically similar with the flight phase sequence of Glide-Descent-Cruise-Climb-Descent within the alternatives listed in Table 3.1. The launch information given in Table 7.2 is entered in the *Launch Specs* pane in the user interface.

Table 7.2 Operational Information of Harpoon AGM-84 [25] [23]

Operational Information	**Values**	**Units**
Launch Mass	523	[kg]
Fuel Mass	49	[kg]

Table 7. 2 (Cont'd)

Range	220		[km]
Speed	Launch	240	[m/s]
	Cruise	240	[m/s]
Altitude	Launch	1,066	[m]
	Cruise	25	[m]
	Search	1,750	[m]

The approximate launch speed at the altitude of 1,000 m is 240 m/s. Also, it is assumed that there is no initial elevation angle θ_0. Considering the design inputs in *Flight Specs* pane, the corresponding desired interval of operational range is 200-225 km and the average desired cruise speed is assumed to be 240 m/s. Also, the altitudes in cruise and search are defined the same with Table 7.2.

7.3 Mission Requirements

In order to forecast the flight performance measures of the original Harpoon which cannot be reached from open sources such as maneuverability, stability and control effectiveness, the flight of the original Harpoon is simulated by using its original dimensions. After obtaining necessary performance information, the approximate intervals including these values will be set as the input of the conceptual design tool in *Objectives* pane. A 2-D drawing of Harpoon AGM-84 estimated from the Harpoon images in the internet is given in Figure 7.4:

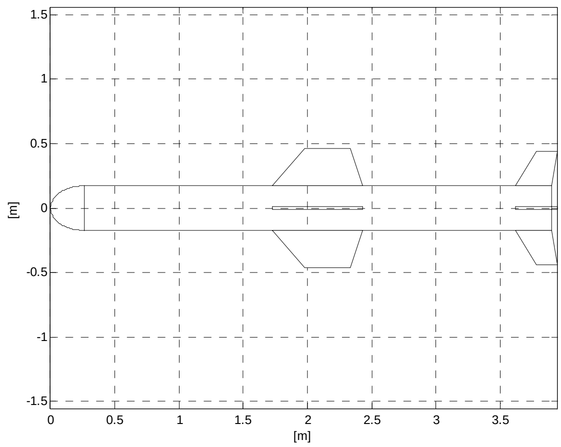


Figure 7.4 2-D View of Baseline Missile

The aerodynamic coefficients are produced by Missile DATCOM according to the original geometry of Harpoon. Then, the two degree of freedom simulator mentioned in Chapter 3 is used for simulating its flight according to the given launch and flight conditions in Table 7.2. The flight phase combination of the original Harpoon is Glide-Descent-Cruise-Climb-Descent. From the simulation results, the altitude change of original Harpoon with respect to time is plotted in Figure 7.5.

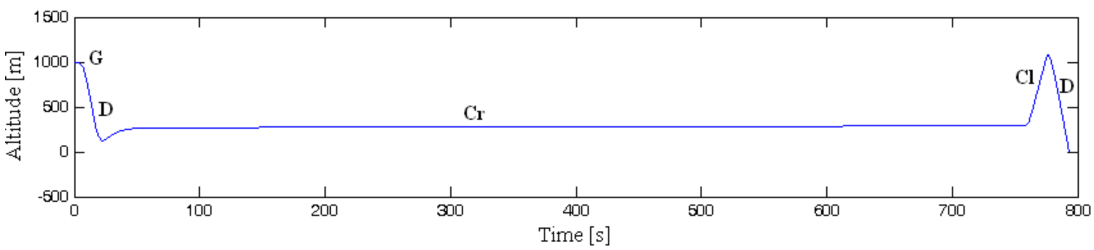


Figure 7.5 Altitude vs. Time Plot of the Baseline Missile in 2 DOF Simulation

The simulation results also give an idea about the load factor of the original Harpoon AGM-84. According to the Figure 7.6, the maximum load factor is observed in pull-up maneuver at the beginning of the climb phase. That means that the original

Harpoon can maneuver by a load factor which is 2.3g. Therefore, the missile which is to be designed in EXCON should be at least the same capacity of maneuvering with that of the original one in order to achieve the trajectory profile in the mission. This brings a requirement to the re-design process of the Harpoon.

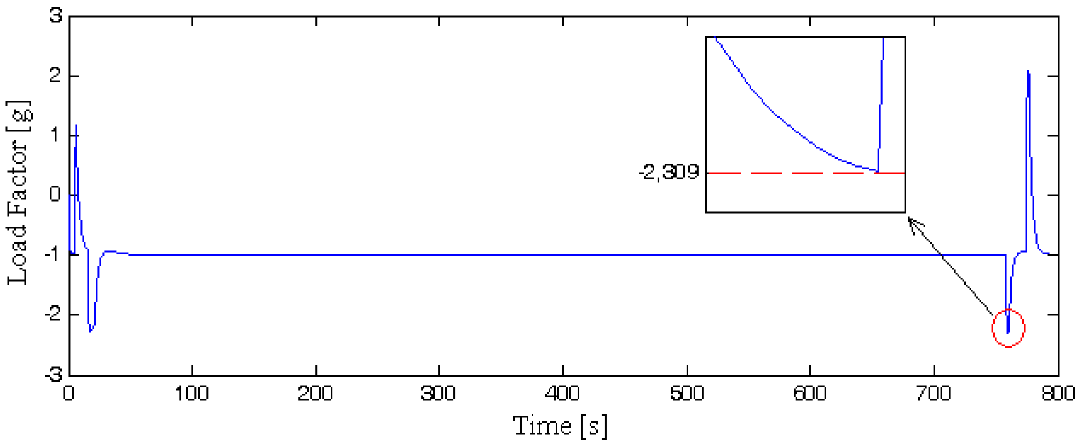


Figure 7.6 Load Factor of Baseline Missile in Pull-up Maneuver

In addition to the flight performance measures, static stability margin SSM is found as

$$\text{SSM} = \min\left|\frac{\Delta C_m}{\Delta \alpha}\right| = 7.36 \tag{7.2}$$

where C_m is the pitch moment coefficient and α is the angle of attack.

Also, the average control effectiveness CE during cruise phase is observed as

$$\text{CE} = \text{mean}\left|\frac{\Delta C_m}{\Delta \delta}\right|_{\text{CRUISE}} = 4.16 \tag{7.3}$$

where δ denotes the fin deflection angle.

Performance measures which are obtained from the simulation of the original Harpoon are to be used for normalizing the fitness function terms later.

7.4 Modeling in Conceptual Design Tool

In the operational information shown in Table 7.2, the range and speed of the missile are set as the constraints of the problem. Therefore, the only remaining performance measure to be minimized is the total mass of the missile. As a result, only the *Minimum Mass* option is checked in *Objectives* pane of the GUI. Also, the interval of desired maneuver load factor should be selected such that it involves the value of load factor of the original harpoon

$$n_{\text{baseline}} = 2.309 \subset [1 \;\; 3] \tag{7.4}$$

Since the actual constraints on control effectiveness and static stability margin of the original Harpoon are not known in conceptual design phase, the intervals for the control effectiveness and stability are selected arbitrarily as

$$SSM_{\text{baseline}} \subset [4 \;\; 6] \tag{7.5}$$

$$CE_{\text{baseline}} \subset [2.0 \;\; 2.5] \tag{7.6}$$

By means of putting these constraints on the static stability SSM and control effectiveness CE, the search area of interest gets smaller which reduces the total time of optimization in the case studies. However, the optimization can be run for different combinations of the SSM and CE intervals. For example, SSM = [8 10] and CE = [3.0 3.5].

In the *Flight Specs* pane of the GUI, the operational range should be entered such that the interval of desired range involves the range of the baseline missile AGM-84

$$R_{\text{baseline}} = 220 \subset [200 \;\; 225] \text{ km} \tag{7.7}$$

Also, the desired speed should be chosen as the nearest value of speed of the baseline missile AGM-84 as

$$\mathrm{V}_{\text{baseline}} = 240 \text{ m/s} \tag{7.8}$$

In the mass model, the fuel-to-launch mass ratio is taken as a constant for all external geometries in the optimization. In order to determine this ratio in re-designing Harpoon, the ratio of fuel $\mathrm{m_F}$ and launch $\mathrm{m_L}$ mass of the original Harpoon is calculated as

$$\left(\frac{\mathrm{m_F}}{\mathrm{m_L}}\right)_{\text{baseline}} = \frac{49}{523} \approx 0.1 \tag{7.9}$$

and used in the conceptual design calculations. According to the above equation, it is assumed that the fuel mass is always 10% of the total launch mass.

The fitness function $f_{N,W,P}$ is composed of the objective term $f_{N,W}$ and the additional penalty term P with a penalty weight λ as

$$f_{N,W,P}(\overline{x}) = f_{N,W}(\overline{x}) + \lambda P(\overline{x}) \tag{7.10}$$

Since there is only one objective to be minimized in the problem, the weighting of the objectives except mass will be taken zero. Therefore, the normalized objective function of the problem becomes

$$f_{N,W}(\overline{x}) = \frac{\mathrm{m_L}}{(\mathrm{m_L})_{\text{baseline}}} \tag{7.11}$$

In addition to the pure objective function term, there are penalty function terms which consist of 4 inequalities and 1 equality constraint terms. The inequality constraints are control effectiveness, load factor, static stability margin, and operational range. They are limited both by lower and upper bounds. The only equality constraint is the average cruise speed during cruise phase. The total penalty term in the fitness function is expressed as

$$P(\bar{x}) = \sum_{i=1}^{5} P_i(\bar{x}) \tag{7.12}$$

The penalty term which is related to the aerodynamic control effectiveness is

$$P_1(\bar{x}) = \frac{\max\left[0, \mathrm{CE}_{\min} - \mathrm{CE}\right] + \max\left[0, \mathrm{CE} - \mathrm{CE}_{\max}\right]}{\mathrm{CE}_{\text{baseline}}} \tag{7.13}$$

The penalty term which represents the load factor is

$$P_2(\bar{x}) = \frac{\max\left[0, n_{\min} - n\right] + \max\left[0, n - n_{\min}\right]}{n_{\text{baseline}}} \tag{7.14}$$

P_3 denotes the constraints on the static stability margin of the missile as

$$P_3(\bar{x}) = \frac{\max\left[0, \mathrm{SSM}_{\min} - \mathrm{SSM}\right] + \max\left[0, \mathrm{SSM} - \mathrm{SSM}_{\min}\right]}{\mathrm{SSM}_{\text{baseline}}} \tag{7.15}$$

The speed of the missile is tried to be converged to the desired baseline speed $\mathrm{V}_{\text{baseline}}$ of AGM-84 Harpoon by adding a fourth penalty term as

$$P_4(\bar{x}) = \left|\mathrm{V} - \mathrm{V}_{\text{baseline}}\right| \tag{7.16}$$

The last term P_5 of the penalty function enforces the operational range of the missile into the interval of desired upper and lower limits:

$$P_5(\bar{x}) = \frac{\max\left[0, R_{\min} - R\right] + \max\left[0, R - R_{\min}\right]}{R_{\text{baseline}}} \tag{7.17}$$

The baseline values based on AGM-84 Harpoon and bound values of the objectives and constraints of the fitness function are listed in Table 7.3:

Table 7.3 Fitness Function Constants

Fitness Terms		**Baseline**	**Lower**	**Upper**
Objectives	Mass	523	-	-
Constraints	Control Effectiveness	4.16	2.0	2.5
	Static Stability Margin	7.36	4	6
	Load Factor	2.31	1	3
	Speed	240	-	-
	Operational Range	220	200	225

EXCON is executed several times for the same problem of conceptual design of AGM-84 Harpoon. The only parameter which is changed for each execution is the penalty weight. EXCON is run for each elements of the set of penalty weights S_λ given below.

$$S_\lambda = \{0.25 \; , \; 0.50 \; , 1.0 \; , 2.0 \; , 4.0\} \tag{7.18}$$

The effects of penalty weights on optimum external geometries and convergence of the optimization are studied. Also, the minimum objective values and the constraint violations for each penalty weights are compared with each another.

7.4.1 The Effect of Penalty weights on Optimum Solution

EXCON is executed for re-designing the baseline geometry Harpoon AGM-84 several times by using each penalty weight used for establishing the fitness function.

Penalty weights determine the dominancy between the objective term and penalty term in the fitness function. The diversity in optimum external geometries for the set of penalty weight in S_λ is illustrated in Figure 7.8:

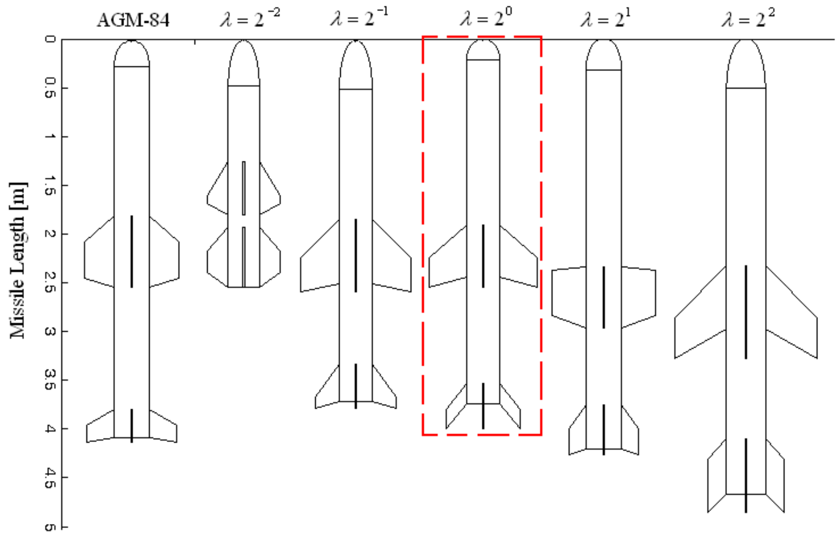


Figure 7.7 Optimum Geometries vs. Penalty weights

In Figure 7.8, variations in optimum mass and constraint violation values with respect to the penalty weights are shown.

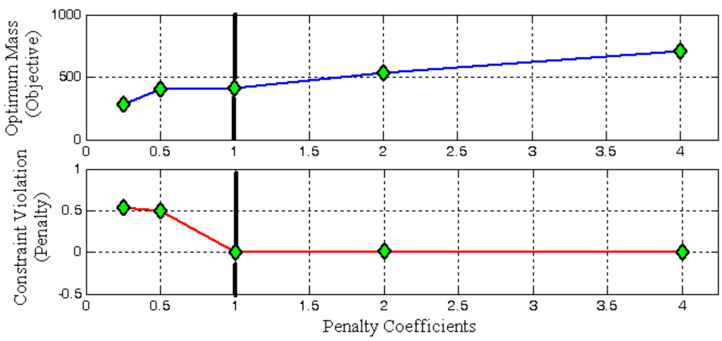


Figure 7.8 Effect of Penalty Weights on Fitness Function

The results of optimum launch mass for each penalty weights represents the objective term of the fitness function whereas the constraint violation values for each penalty weights denotes the penalty term of the fitness function. In order to have a feasible optimum solution, the constraint violation should be zero. However, it is observed that the constraint violation is increased when the penalty weight is getting smaller. Small penalty weights reduce the impact of penalty term in a fitness function, therefore, constraints might be violated.

When the optimum mass plot is observed, a decreasing trend is observed when the penalty weight is reduced. A small penalty weight gives a dominant effect on the pure objective term (mass). Therefore, the optimization algorithm focuses on objective term more than the penalty term which helps finding an optimum missile which has lighter weight. However, there is no improvement in the optimum mass value below a certain limit of penalty weight **(**$\lambda = 1$**)**. This is due to the fact that range requirement is not satisfied below a certain limit of penalty weight.

According to the Figure 7.8, the value of the penalty weight on which the constraints have started to be violated is 1. Therefore, penalty weight cannot be smaller than 1. Since optimum mass increases with increasing penalty weights, design point should be on 1 on which the launch mass is the smallest. As a result, the penalty weights for the external configuration problem of Harpoon should be chosen as 1 and the corresponding design point can be called the optimum solution in the feasible region.

With the penalty weight value 1, the mass convergence history of the best individuals in each generation is illustrated in Figure 7.9. One can observe that the launch mass is converged at the 21st generation.

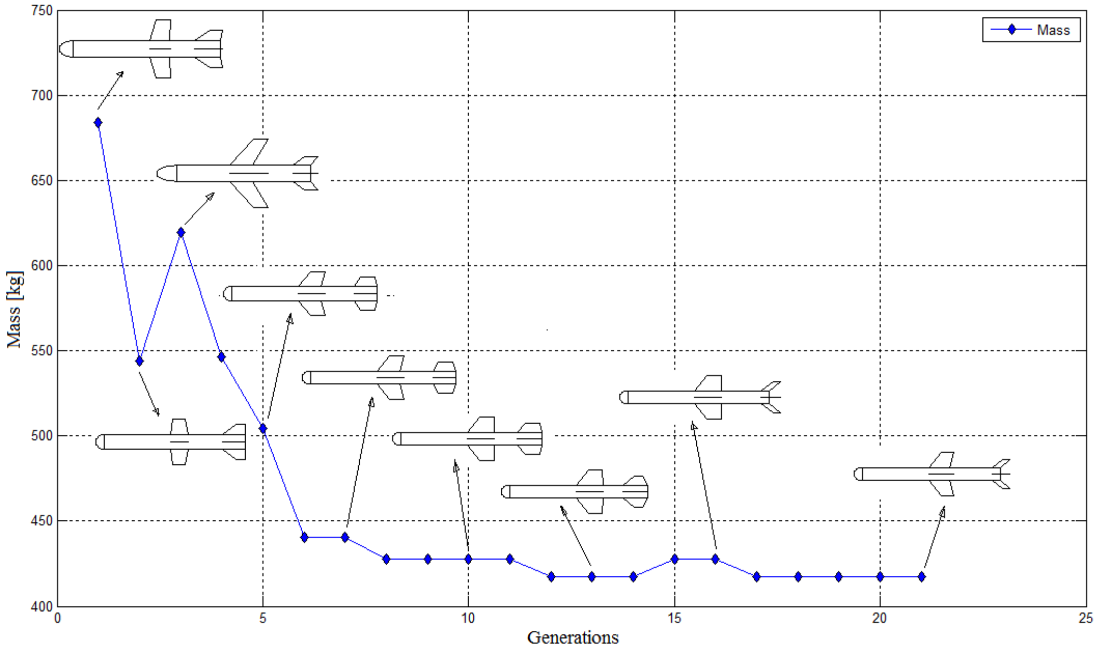


Figure 7.9 Mass Convergence History of the Optimum Geometry (λ = 1)

The final value of the launch mass of the missile is found as 417 kg. The re-designed Harpoon can be delivered to 215 km with 41.7 kg fuel. The flight performance variations in time during the flight simulation are given in APPENDIX B.

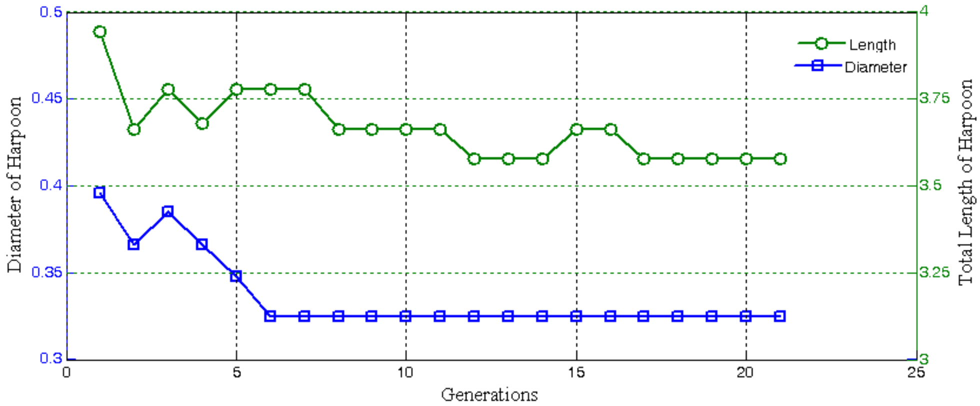


Figure 7.10 Convergence of Diameter and Total Length of the Harpoon (λ = 1)

As seen in Figure 7.10, the total length converges before the diameter of the missile. The improvement in mass until 6^{th} generation is supported by both diameter and length changes. However, the only contributor to the mass improvement is the length after 6^{th} generation since the diameter of the missile keeps constant after 6^{th} generation. As mass is assumed to be linear with volume ($l \times d^2$) of the missile, the mass convergence trend can be also observed from the trends of diameter and length.

7.4.2 Possible Reasons for the Deviations From Baseline Geometry

The flight performance comparison between optimum design of EXCON and baseline design of AGM-84 Harpoon is shown in Table A.1 of Appendix B. The ranges of these two missiles are very close to each other such that the redesigned Harpoon in EXCON has a range which is 2.3% less than the range of original one. However, the total launch mass is achieved to be reduced to 20% of total launch mass of the original harpoon. This implies that the total length and the diameter of the Harpoon which is re-designed in EXCON are decreased. Therefore, the launch platform compatibility is enhanced since there is less space needed for the Harpoon.

The fineness ratio of redesigned missile is 11.0 whereas it is 11.5 for the original AGM-84. Reducing the fineness ratio is beneficial for the structural rigidity of the missile.

The optimum design has the same fin configuration (044) and control type (tail) with the baseline harpoon AGM-84. However, it has a plus panel orientation whereas the original Harpoon has cross panel orientation. This leads the optimum missile to have less control effectiveness. As a result, the control effectiveness of the re-designed missile is 55% of the AGM-84. However, it is still above the value of 2 which can be considered the minimum design criteria for the control effectiveness.

There might be some reasons for obtaining a smaller size missile than original baseline size. Although, the constraints are tried to be forecasted during original design of AGM-84 Harpoon, there might be some missing design constraints which are unknown. For example, EXCON can take into account only the constraints due to motor diameter. However, there might be also other subsystem constraints due to the sections of warhead, fuel tank, and guidance. The difference in the lengths of redesigned and baseline missiles may arise from these subsystem constraints. For instance, the length of the missile cannot be smaller than a certain lower limit in case that the inputs such as minimum warhead length are given.

Another reason of the deviations in external geometry parameters between original and re-designed Harpoon might be due to the constraints in manufacturing phase. The engineers can have to change the final design of a product even during the difficulties in manufacturing phase.

Since it is impossible to know all the constraints during the design of the original AGM-84 Harpoon, the redesigned harpoon should not be expected to look like almost the same with the original design. Also, it is meaningless to compare the external geometry of the missile which is obtained by EXCON with the final design of the original AGM-84 Harpoon since the conceptual design outputs are usually modified in the detail design stages.

7.4.3 Convergency of Hybrid Algorithm

By means of hybrid algorithm, the solutions found by GA are improved. It helps the solution to be converged to the optimum. In the Table 7.4, the improvements in mass values obtained by hybrid function (`fmincon`) are shown.

Table 7.4 Mass Improvements for Different Penalty Coefficient

Penalty Coefficient	**GA Solution**	**`fmincon` Solution**	**Improvement**	**Unit**
$\lambda = 1$	417.27	417.27	0.00	[kg]
$\lambda = 2$	540.86	535.70	5.16	
$\lambda = 4$	718.60	712.45	6.15	

According to Table 7.4, there are little reductions in mass for penalty coefficients 2 and 4 whereas the hybrid function could not improve the mass found by GA. There might be 2 reasons of no improvement in mass: One might be due to insufficient time for running `fmincon`, other might be because that GA already found the optimum solution.

In order to observe the global convergence of GA, the program is run more than once with the same inputs and penalty coefficient 2. The mass convergence plots for different runs are given in Figure 7.11:

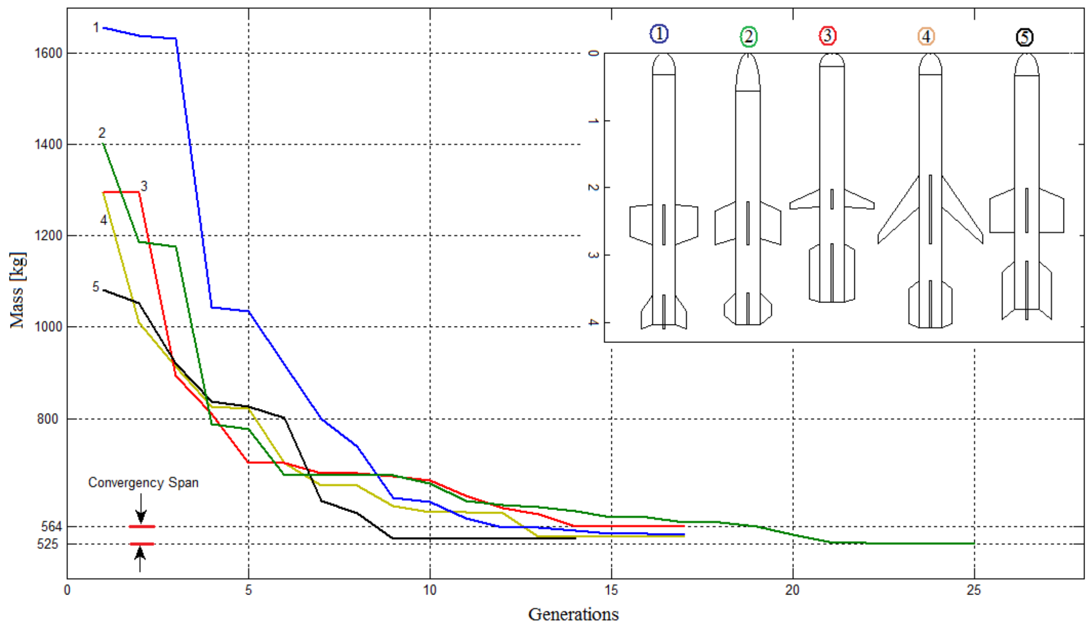


Figure 7.11 Convergence History of Mass for $\lambda = 2$

As seen in the above figure, one can observe that the final solutions converge within a certain span of convergency between 525 and 564 although each of 5 runs start with different initial points in search space. This implies that the genetic algorithm manages to escape from local optima and results in a global minimum. The small deviations between final solutions for different runs of the same case can be due to either from the tolerance limits and maximum number of function evaluation limits for stopping GA or `fmincon`.

The images of corresponding best solutions for these 5 different run are also shown in the Figure 7.11. According to images, the length and diameter of all missiles are observed to be very close to each other. This is an expected result since the mass is a function of diameter and length only. The contributions of fin surfaces to the total mass can be neglected. The shape and size of the wing and tail fins are different according to images. The reason is obviously due to the fact that the performance metrics (control effectiveness, maneuverability) which is highly dependent on the fin surfaces are taken as inequality constraints. It gives a freedom of shaping the fin surfaces within acceptable bounds of inequalities. Therefore, the deviations in wing-tail size and shape for 5 different solutions are not surprising.

CHAPTER 8

RESULTS AND CONCLUSION

8.1 Summary and Results

This thesis presents the methods used to create an integrated external configuration and geometry sizing environment for the design of missiles. This environment including a flight simulator, an optimization model, and a graphical user interface is aimed to have the ability to provide the impact of changing mission requirements on the conceptual design of the missiles. A software tool named EXCON is developed as a conceptual design environment. By means of EXCON, the external configuration and geometry of the missiles having either turbojet or solid propellant motor can be designed according to the mission profile specified by the designer in the graphical user interface. Therefore, the external geometry optimization is obviously mission-dependent. That is, the optimum configuration of the missile is affected by the user-defined mission profile such as flight trajectory, motor type, and launch type.

In order to get rid of auto-pilot design for each geometry candidates in conceptual design stage, the proportional gains of the controllers are kept constant. This means that they depend on neither the mission plan nor the external geometry. As long as stabilities of all controllers are satisfied, the values of these gains have an impact only on the transition parts (i.e. pull up/down) of flight trajectory profiles since they determine the behavior of the system while it reaches to steady state conditions (i.e. trim angle of attack, constant speed). Therefore, changing their values only shapes the transition flight profile parts in a different way. After steady state conditions are reached like in cruise, climb, descent etc., they loose their effects on the flight trajectory. Since the longest part of the trajectory is composed of the cruise phase, the contribution of the transition parts (pull-up/down) can be neglected compared to that of cruise part. That implies that the effect of the auto-pilot gain values on range can be ignored. Similarly, since the control effectiveness and load factor are functions of only aerodynamic forces, moments and weight, they are not affected by value of the controller gains, neither.

The conceptual design process developed in this thesis utilizes a genetic algorithm based technique in order to determine the design parameters to result in a minimum mass, maximum range, and maximum speed missile which meets a set of constraints on several aerodynamic performance parameters such as maneuverability, control effectiveness, and stability. In addition to the aerodynamic performance constraints, other constraints based on launch platform compatibility and structural limitation issues are also taken into account.

Since the conceptual design of launch vehicles involves various disciplines in a highly coupled manner, a weighting factor strategy is applied to the cost function which takes into account the severity of each objective with respect to one another. Also, in order to achieve a feasible optimum solution which satisfies the constraints, the penalty method is used. The additional penalty term is added to the cost function expression with a suitable penalty weight.

In the optimization algorithms utilized in EXCON, the outcome geometries are produced by using different penalty weights in the cost function in order to survey the impact of them on the optimum results. Launch masses of the optimum external geometry solutions are observed to decrease when penalty weights used are decreasing. However, the constraint violation is getting higher with decreasing penalty weights. In order to have a feasible optimum solution, there must be a zero constraint violation. Therefore, a balance should be made between two opposing concepts: minimization of launch mass and constraint violation. As a result, the design point is selected as the optimum external configuration solution which is on the boundary of constraint violation.

In order to check whether the solution of GA is a near-optimum or not, the hybrid algorithm is used in order to push the near-optimum solution to the optimum. In some of the case studies, it is observed that the hybrid function (`fmincon`) improves the solution found by GA.

As a verification of the conceptual design tool developed, a case study is conducted to determine a set of optimal external configuration and geometric dimension parameters for an air-to-air turbo-jet engine anti-ship missile. The Harpoon AGM-84 missile reconfiguration problem is formulated and integrated into the proposed software tool (EXCON) in order to improve the launch mass of the baseline missile, Harpoon AGM-84.

The main purpose of performing such a case study is to verify the software tool developed. For the verification of EXCON, there are two success criteria targeted to obtain. One of them is to confirm whether EXCON is capable of reducing the launch mass of the original baseline missile with known constraints without deteriorating the range and maneuverability characteristics of the baseline missile dramatically. The process in the case study yields an external configuration that is significantly lighter (approximately 20%) than the launch mass of the original baseline missile. During the improvement in launch mass of the original baseline missile, the maneuverability is also improved by 16% of the original Harpoon. Although the range was smaller by 2.3% than the range of baseline missile, this difference in the range is at a negligible level compared to the improvements obtained in the launch mass and maneuverability. As a result, the first criterion of verification is proved.

Other success criterion of EXCON is to observe whether the external geometry of the optimum missile is still similar with that of the original baseline missile roughly. This is needed for the verification of the code written in EXCON. Comparing the external geometry view of a newly design missiles with that of a baseline missile in literature should be beneficial for proving that the missiles designed in EXCON can be producible and fly in a real atmosphere conditions. Although the size of the optimum missile obtained by EXCON, is smaller than that of original one because of the reduced launch mass, the proportion of the dimensions for both of the missiles are observed to be similar. For example, the fineness ratios of optimum and original baseline missile are approximately 11.0 and 11.5, respectively, which can be considered to be close to each other. Also, the wings of the optimum missile are located at 51% of total length; similarly, the wings of the original Harpoon are located at 50% of total length. Since both success criteria for verification of EXCON are achieved by the case study, EXCON is proven to be an effective tool for cruise missile performance analysis and configuration sizing.

8.2 Conclusions

Although the external geometry of optimum missile is found to be similar to the original baseline missile Harpoon AGM-84, one should not always expect to obtain such a similarity in the external geometries. There would be reasonable distinctions in the external shapes of the missiles if the re-configuration problem were applied to another missile. The difference between the external geometries can be higher when the number of unknown constraints in the re-design process is increased. In general, it is impossible to know all the real-life constraints faced with during a real design and manufacturing process of a missile. In EXCON, most of the subsystem constraints related to warhead, guidance-control section, and seeker are not taken into account. However, these constraints have additional impacts on the warhead effectiveness, lethality as well as producibility of the missile. In addition to them, the radar cross section area calculations which affects the radar detection probability of the missile are not included to the cost function evaluations during the case study of this thesis. Although they have impacts on the final geometry of the optimum missile, these constraints on the baseline missile Harpoon AGM-84 can not be obtained from open sources. Therefore, it is impossible to include these unknown constraints into the re-configuration process in EXCON. As a result of that, the total length of the optimum missile in the case study is smaller by 8.5% than that of original baseline missile.

There is another reason of possible geometry difference between baseline missile Harpoon AGM-84 and the optimum missile found by EXCON. In real projects, some of the constraints can appear in the later detail design stages following the conceptual design. Manufacturing feasibility is one of an example of this case. The decisions taken in the conceptual design stages could be suddenly changed due to the constraints faced with in the detailed design stages. Since the optimum Harpoon found by EXCON is the outcome of the conceptual design process and the original baseline Harpoon AGM-84 is the ultimate product, the deviation of the optimum harpoon geometry from original Harpoon AGM-84 geometry is not astonishing.

Since the main focus of EXCON is obtaining an external geometry of a missile of which aerodynamic performance is optimum, EXCON can be used as an aerodynamic analysis and synthesis tool in the conceptual design of missiles in the defense industry. It brings an automation to the conceptual design of the missiles which helps to decrease the total time and to minimize the interference of the designers in conceptual design of the missiles.

8.3 Recommendations for Future Work

EXCON is capable of performing conceptual design of air-to-surface and surface-to-surface missiles which have either a solid propellant or turbo-jet motor. However, the scope of the missiles can be enlarged through the high maneuverable air-to-air low range missiles as a future study. In order to achieve this, the ramjet and scramjet motor models should be included into the propulsion modules in order to obtain high Mach numbers for air-to-air flight types.

The external geometry optimization presented in this thesis is a multi-objective problem. However, some of the objectives like selection of the optimum propulsion type, minimizing radar cross-section area, maximizing warhead effectiveness are not taken into account. Therefore, the span of the objectives can be enlarged as a future study. Also, some of the extended constraints related to subsystem requirements and some of the structural factors can be included into the graphical user interface menus.

Since Missile DATCOM can produce aerodynamic coefficients only for circular and elliptical cross sectional missile bodies, the state-of-art missiles having cross sections like polygon cannot be modeled in EXCON. Therefore, the aerodynamic estimations formulae for arbitrary body cross-sections can be added into the algorithms utilized in EXCON.

The case studies investigated in the thesis are focused only in the penalty weights' effect on the optimum external geometry. The effects of genetic algorithm parameters like mutation rate, population size, elite count on the optimization performance are not studied. The recommended genetic algorithm parameters by MATLAB are selected for the case studies. Therefore, the performance tests can be handled by changing the genetic algorithm parameters in order to minimize the total optimization time and speed up the convergence rate. Also, the mutation rate can be changed adaptively according to the convergence rate of the algorithm during run. In case of slowness of the convergence rate, either arbitrary new seeds can be created and added to the population or the mutation rate can be increased. These interferences in the algorithm during run time might improve the speed of convergence.

Since the flight simulation is handled by using 2 DOF model in vertical plane, only the effect of pitch on the external geometry of missiles are investigated. However, the effect of yaw and roll properties such as control effectiveness and stability in yaw and roll rotations can also be taken into account while designing the external geometry by increasing the degree of freedom of the model.

REFERENCES

[1] Anderson, J. D., "Aircraft Performance and Design", McGraw-Hill, USA, 1999.

[2] Bennett, B. K., "Conceptual Design Synthesis Tool for Arbitrary-body Missiles", 15th Applied Aerodyanmics Conference, Atlanta, GA, USA, AIAA 1997-2281, June 23-25, 1997.

[3] Bowcutt, K. G., "Multidisciplinary Optimization of Airbreathing Hypersonic Vehicles", Journal of Propulsion and Power, Vol. 17, No. 6, 1184-1190, 2001.

[4] Blake, W. B., "Missile DATCOM User's Manual", Wright Patterson Air Force Base, Ohio, USA, 1997.

[5] Cantú-Paz, E., "Efficient and Parallel Accurate Genetic Algorithms", Springer, Massachusetts, USA, 2001.

[6] Ender, T. R., McClure, E. K., Mavris, D. N., "Development of an Integrated Parametric Environment for Conceptual Hypersonic Missile Sizing", AIAA's Aircraft Technology, Integration and Operations (ATIO) Technical Forum, Los Angeles, California, USA, AIAA 2002-5856, 2002.

[7] Engelund, W. C., Stanley, D. O., Lepsch, R. A., McMillian, M. M., "Aerodynamic Configuration Design Using Response Surface Methodology Analysis", Aircraft Design, Systems and Operations Meeting, Monterey, California, USA, AIAA 1993-3967, 1993.

[8] Fleeman, E. L., "Tactical Missile Design", AIAA Education Series, USA, 2001.

[9] Gen, M., Cheng, R., “Genetic Algorithms and Engineering Design”, Wiley Interscience, USA, 1997.

[10] Gouwanda, D., Ponnambalam, S. G., “Evolutionary Search Techniques to Solve Set Covering Problems”, Vol. 39, World Academy of Science, Engineering and Technology, ISSN: 2070-3724, 2008.

[11] Luenberger, D. G., “Linear and Nonlinear Programming”, Addison-Wesley, USA, 2003.

[12] Meerschaert, M. M., “Mathematical Modeling”, Academic Press, USA, 2007.

[13] Nguyen, N., Jeon, K., Lee, W., Byun, Y., “The Repetitive Optimization Design Strategy Using Neural Network and Hybrid Algorithm”, Research, Innovation and Vision for the Future IEEE, International Conference, Ho Chi Minh City, Vietnam, ISBN: 978-1-4244-2379-8, 2008.

[14] Potosnak, K., “Conceptual Design”, [online] http://www.interfaceconcepts.com/concept.htm, last accessed date: 31/08/2009.

[15] Summerfield, M., “Tactical Missile Aerodynamics”, AIAA, New York, 1986.

[16] Whitley, D., “A Genetic Algorithm Tutorial”, Vol. 4, Statistics and Computing, 65-85, ISSN: 0960-3174, 1994.

[17] Mathworks, Inc., “SIMULINK toolbox for use with MATLAB®”, User’s guide, Natick, MA, USA, 2007.

[18] Mathworks, Inc., “Termination Criteria and Optimization Results”, MATLAB®. User’s guide, Natick, MA, USA, 2007.

[19] Mathworks, Inc., "Simple Variable Mass 3DOF (Body Axes)", MATLAB®. User's guide, Natick, MA, USA, 2007.

[20] Mathworks, Inc., "COESA Atmosphere Model", MATLAB®. User's guide, Natick, MA, USA, 2007.

[21] Mathworks, Inc., "WGS84 Gravity Model", MATLAB®. User's guide, Natick, MA, USA, 2007.

[22] Crowell, A.G., "The Descriptive Geometry of Nose Cones", 2006, [online] https://projetosulfos.if.sc.usp.br/artigos, last accessed date: 31/08/2009.

[23] "The Ultimate F-16 Reference, AGM-84 Harpoon", 2009, [online] http://www.f-16.net/f-16_armament_article12.html, last accessed date: 31/08/2009.

[24] "Antiship Missiles, AGM-84 Harpoon", 2009, [online] http://www.vectorsite.net/twbomb_10.html, last accessed date: 31/08/2009.

[25] "Integrated Defense System, Harpoon", 2009, [online] http://boeing.com/defense_space/missiles/harpoon, last accessed date: 31/08/2009.

[26] "Strategic-Air-Command, AGM-84 Harpoon Missile, 2009, [online] http://www.strategic-air-command.com/missiles/Aircraft Launched_Missiles/agm-84_harpoon_missile.htm, last accessed date: 31/08/2009.

[27] "National Air and Space Museum, Teledyne CAE J-402-CA-400 Turbojet Engine", 1989, [online] http://collections.nasm.si.edu, last accessed date: 31/08/2009.

[28] Mavris, D., “Peregrine Hypersonic Strike Fighter Weapons System”, A proposal to the AIAA foundation in response to the AIAA Foundation Graduate Team Aircraft Design Competition request for proposals, Georgia Institute of Technology, Atlanta, Georgia, 2000.

[29] Kutluay, Ü., Ortaç, S. A., “Flight Mechanics Conceptual Design Report”, TÜBİTAK-SAGE, Ankara, Turkey, 2007.

[30] Ortaç, S.A., “Optimal External Configuration Design of Missiles”, M.S. Thesis in M.E.T.U. Mechanical Engineering Dept., Ankara, Turkey, 2002.

[31] Alberico, J., “The Development of an Interactive Computer Tool for Synthesis and Optimization of Hypersonic Airbreathing Vehicles”, American Institute of Aeronautics and Astronautics, AIAA-92-5076, 1992.

APPENDIX A

OPTIMIZATION RESULTS

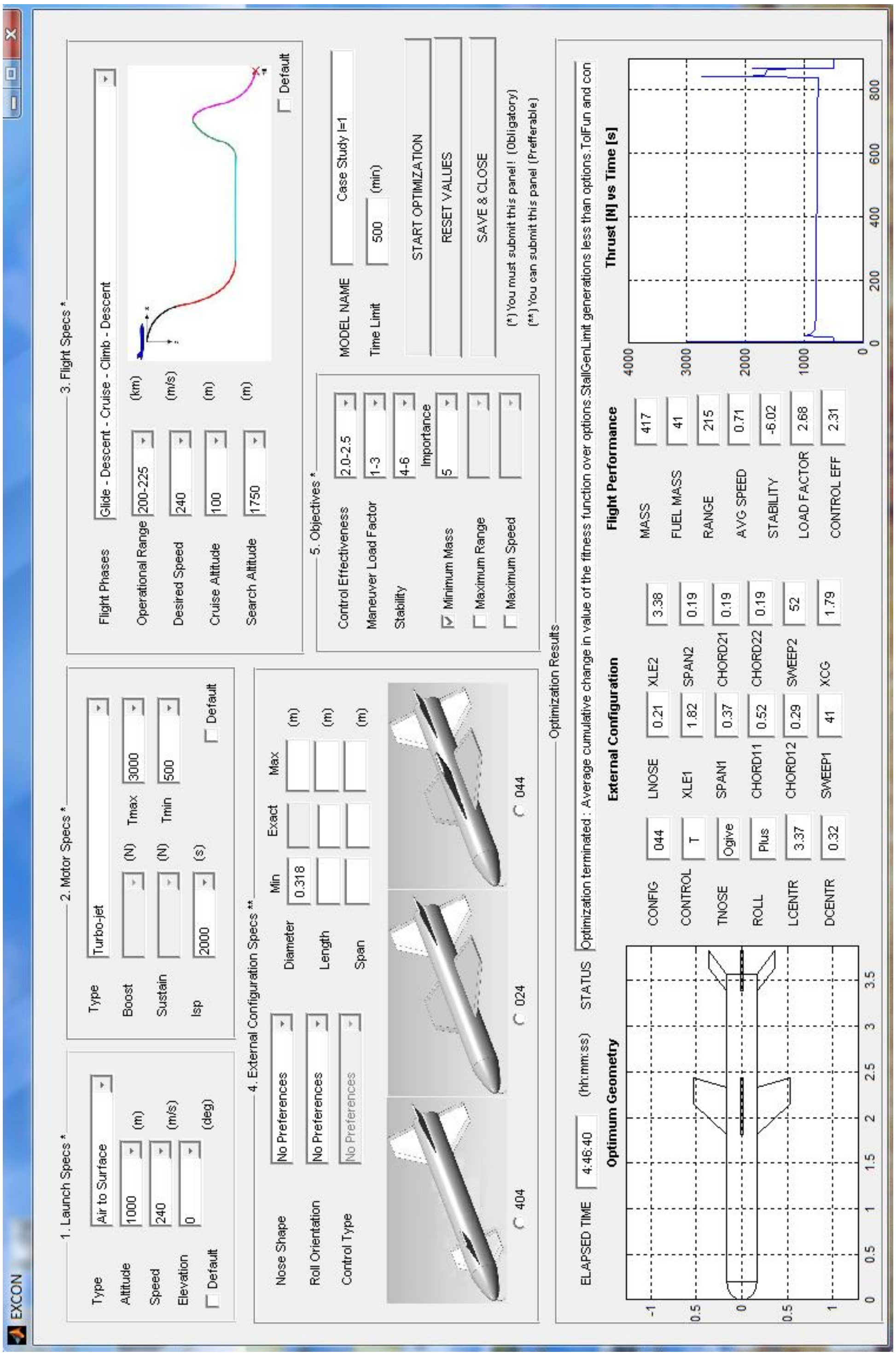


Figure A. 1 EXCON Graphical User Interface (GUI)

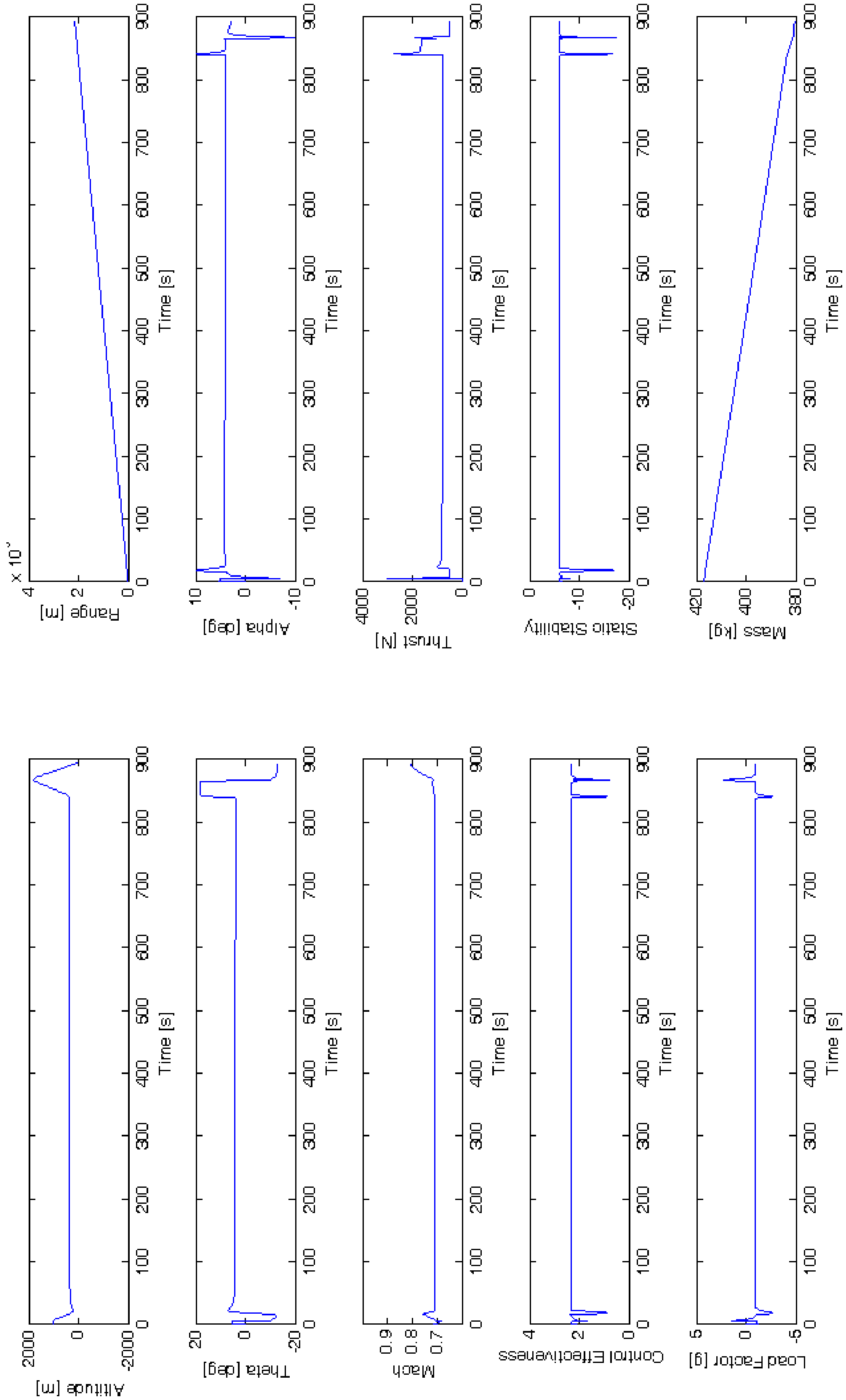


Figure A. 2 Flight Simulation Results of EXCON Optimum at λ = 1

Table A. 1 Flight Performance Comparison of Baseline and EXCON Missile

Flight Performance Data	EXCON Optimum $\lambda = 1$	Baseline AGM-84	Unit
Mass	417	523	[kg]
Fuel Mass	41.7	49	[kg]
Control Effectiveness	2.31	4.16	-
Static Stability Margin	-6.02	-7.36	-
Load Factor	2.68	2.31	[g]
Speed	240	240	[m/s]
Operational Range	215	220	[km]

Table A. 2 External Geometry Dimensions of EXCON Results and Baseline Missile

External Geometry Parameters	EXCON Optimum $\lambda = 1$	Baseline Missile	Default Bounds		Unit
			Upper	Lower	
D	0.33	0.34	0.1	0.6	[m]
L	3.57	3.90	1.00	6.00	[m]
LN	0.21	0.25	0.10	0.70	[m]
XCG	1.79	1.95	-	-	[m]
S1	0.37	0.30	0.05	1.60	[m]
RC1	0.52	0.70	0.05	1.00	[m]
TC1	0.29	0.35	0.05	1	[m]
SW1	41	40	0	60	[deg]
XLE1	1.82	1.70	-	-	[m]
S2	0.19	0.30	0.05	0.5	[m]

Table A. 2 (Cont'd)

RC2	0.19	0.30	0.05	1	[m]
TC2	0.19	0.15	0.05	1	[m]
SW2	52	30	0	60	[deg]
XLE2	3.38	3.60	-	-	[m]

Table A. 3 External Configuration Parameters of EXCON results and Baseline Missile

External Configuration Parameters	**EXCON Optimum $\lambda = 1$**	**Baseline AGM-84**
Nose Shape	Ogive	-
Fin Configuration	044	044
Panel Orientation	Plus (+)	Cross (x)
Control Type	Tail	Tail

Table A. 4 MATLAB Genetic Algorithm Parameters

GA Parameters	**Value**
`PopulationType`	`'doubleVector'`
`PopInitRange`	`100`
`populationSize`	`200`
`CrossoverFcn`	`@crossovercustom`
`CrossoverFraction`	`0.8`
`SelectionFcn`	`@selectionstochunif`
`MutationFcn`	`@mutationcustom`

Table A. 4 (Cont'd)

CreationFcn	@gacreationcustom
Generations	100
EliteCount	10
TimeLimit	inf
StallGenLimit	10
StallTimeLimit	Inf
TolFun	1e-3
TolCon	1e-4

APPENDIX B

MODELS

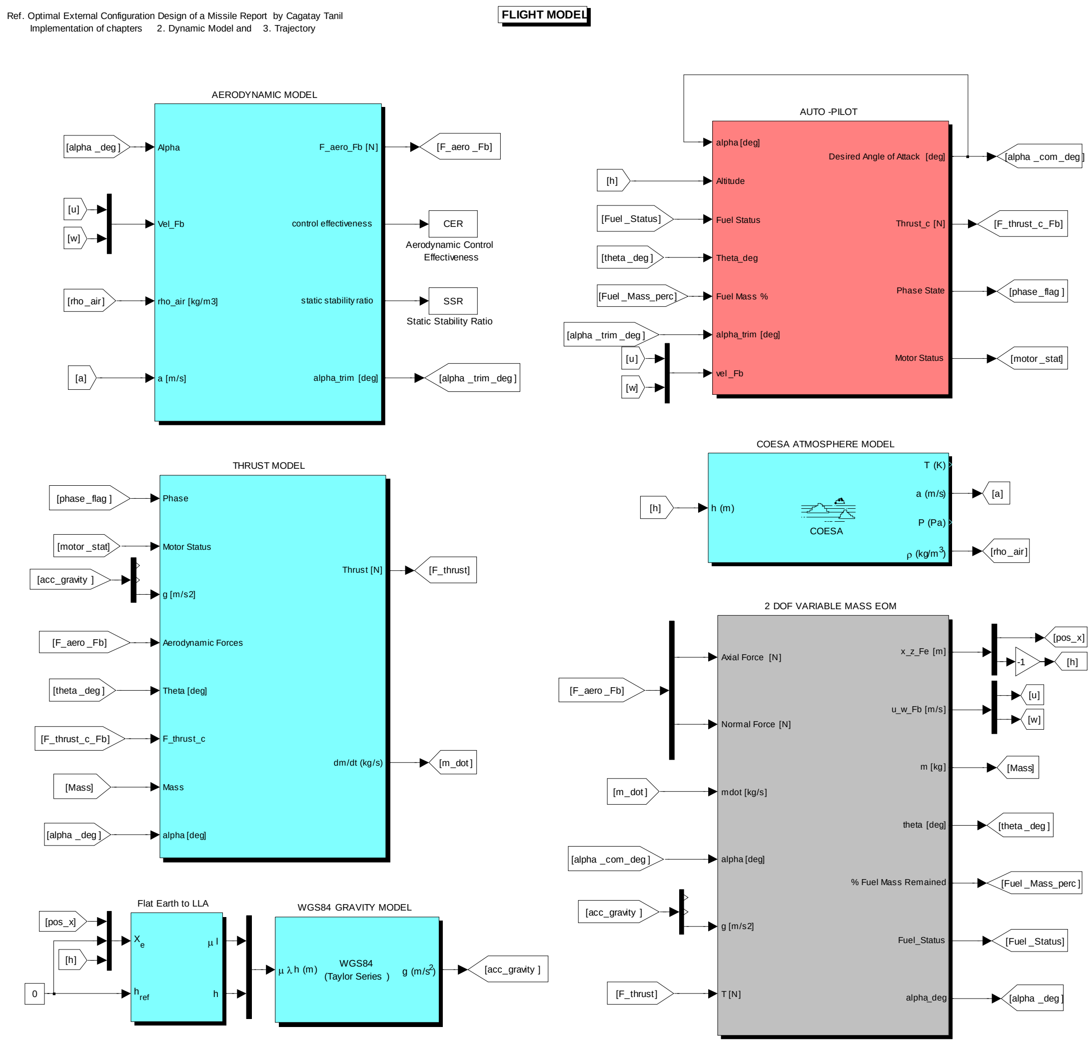


Figure B. 1 Simulink Block Diagram of Flight Simulator

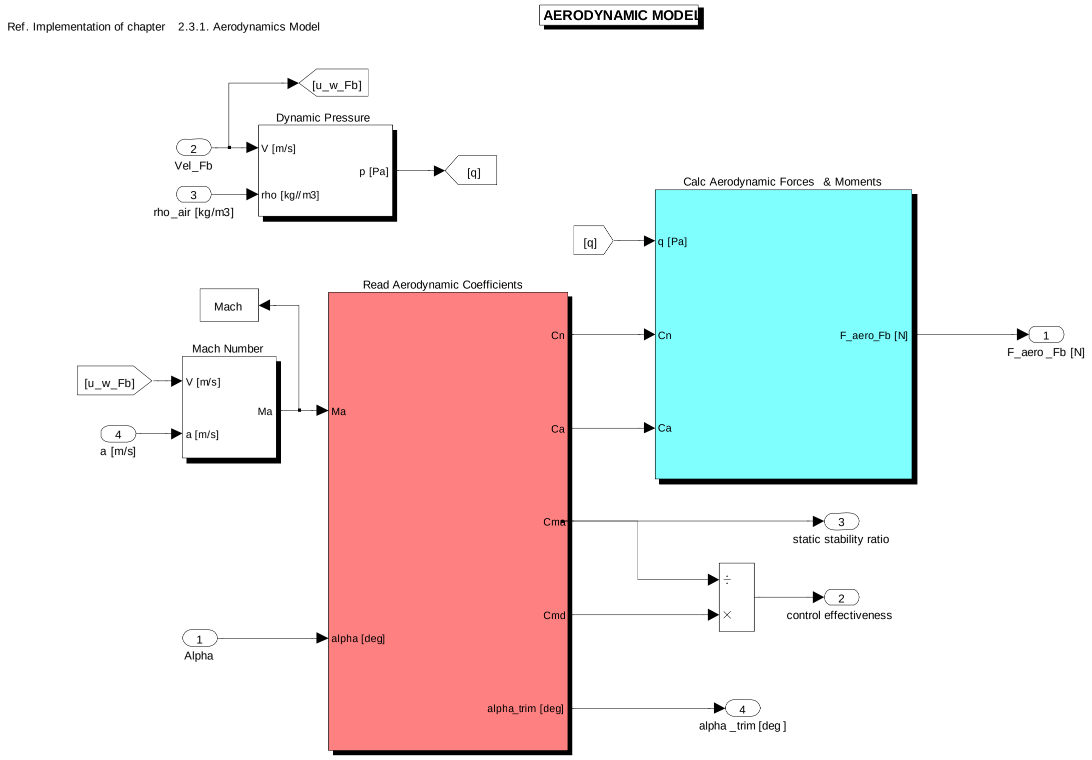


Figure B. 2 Simulink Block Diagram of Aerodynamic Model

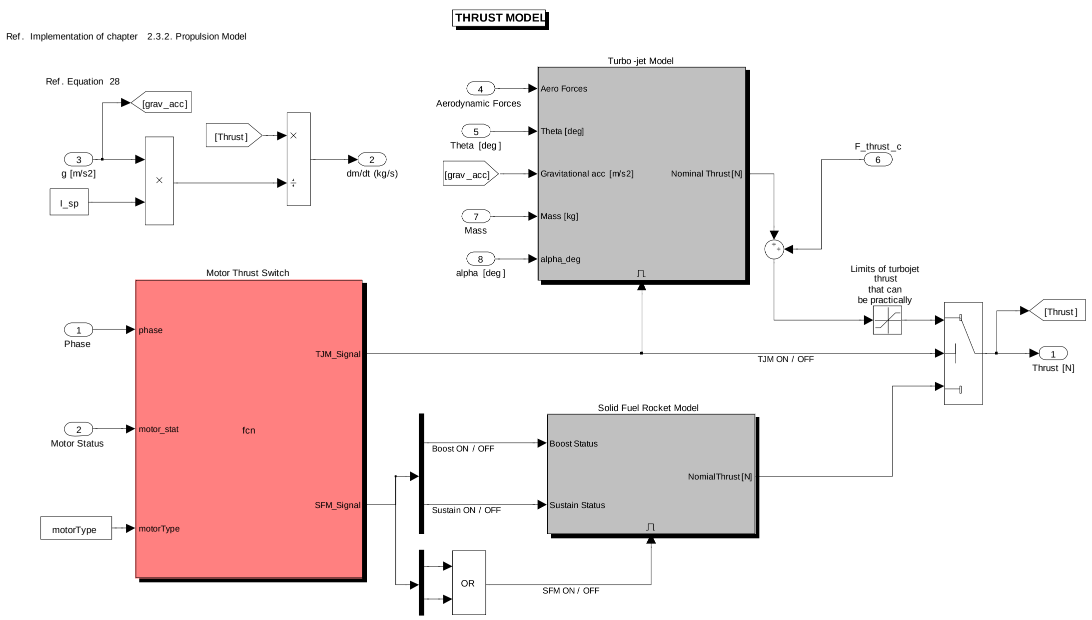


Figure B. 3 Simulink Block Diagram of Thrust Model

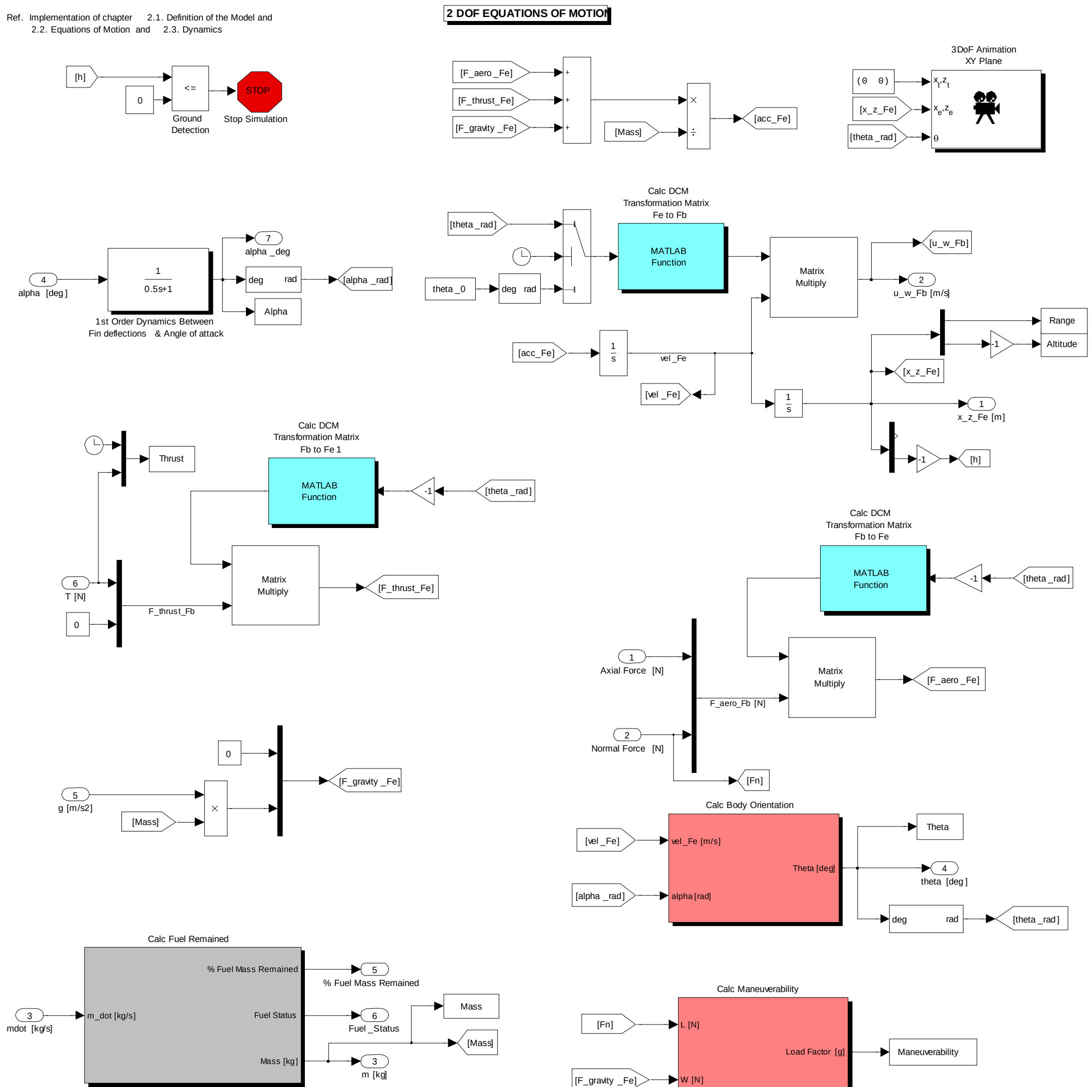


Figure B. 4 Simulink Block Diagram of EOM (Equations of Motion)

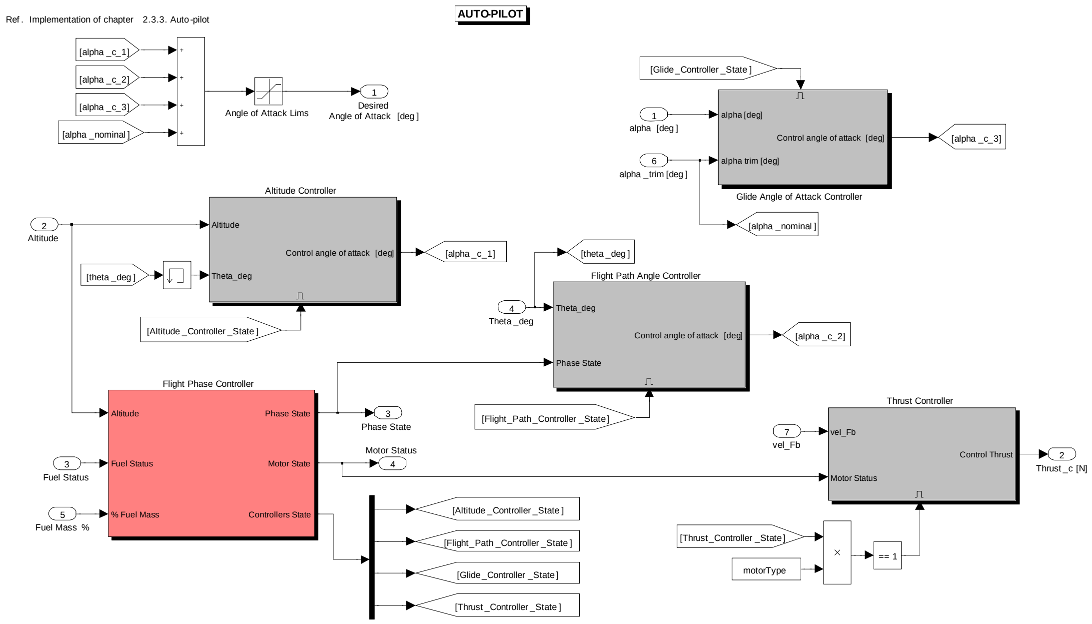


Figure B. 5 Simulink Block Diagram of Autopilot

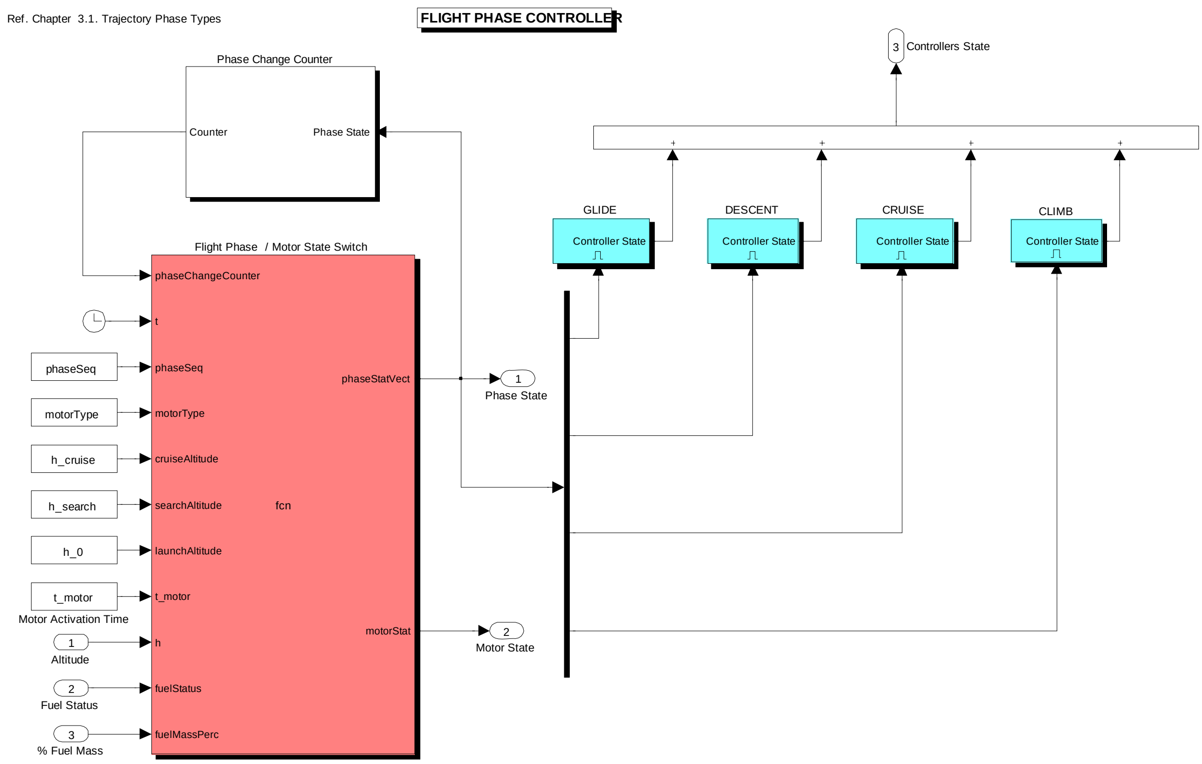


Figure B. 6 Simulink Block Diagram of Flight Phase Controller

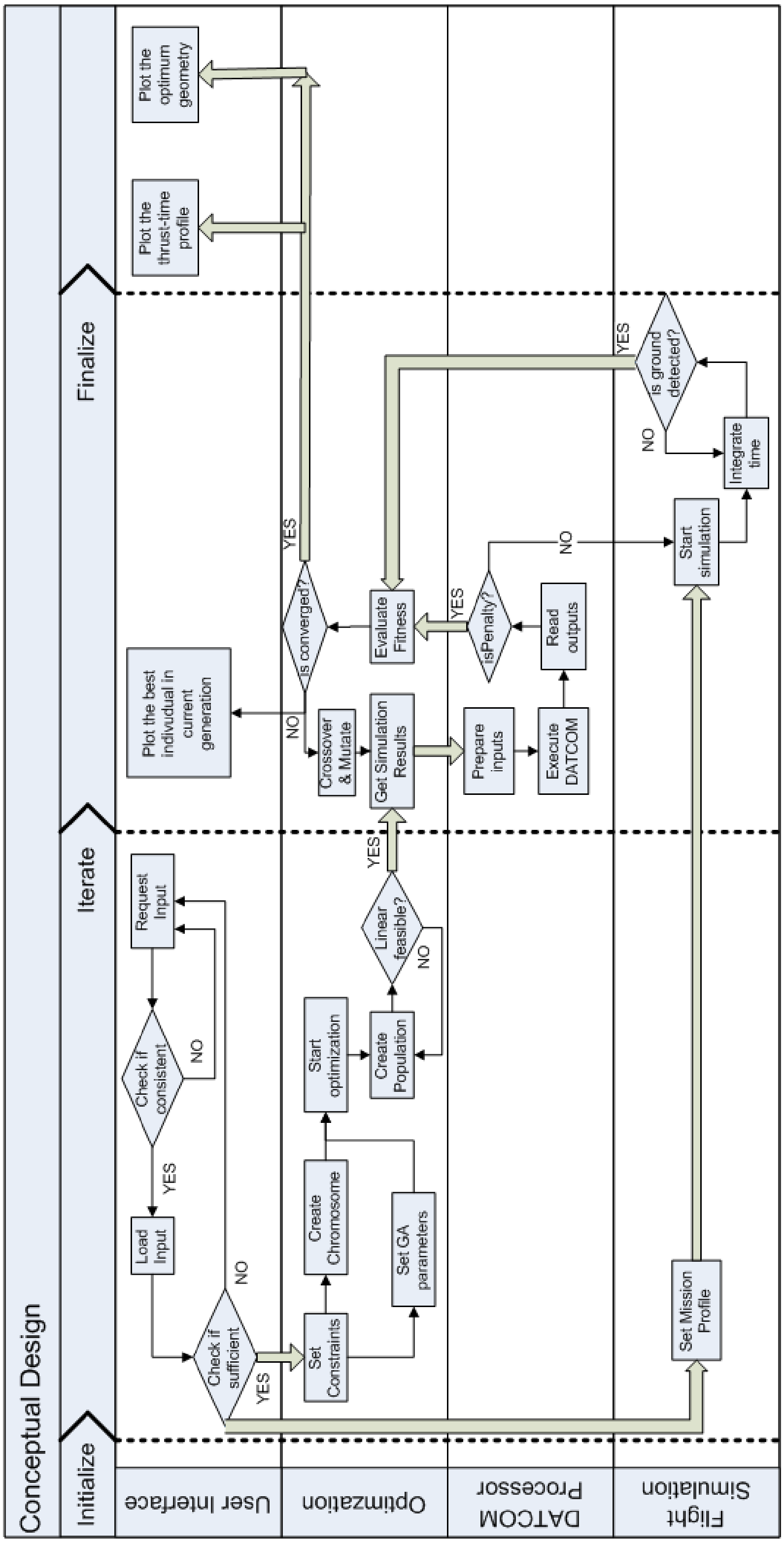


Figure B. 7 Software Architecture Design