Sudan University of Sciences and Technology

College of Engineering

Aeronautical Engineering Department

PERFORMANCE VALIDATION OF L-410 TURBOLET

A Project Submitted In Partial Fulfillment for the Requirements of the Degree of B.S.c (Honor) In Aeronautical Engineering

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بسم الله الرحمن الرحيم

ـ {اقْرَأْ بِاسْمِ رَبِّكَ الَّذِي خَلَقَ * خَلَقَ الْإِنْسَانَ مِنْ عَلَقٍ * اقْرَأْ وَرَبُّكَ الْأَكْرَمُ * الَّذِي عَلَّمَ بِالْقَلَمِ * عَلَّمَ الْإِنْسَانَ مَا لَمْ يَعْلَمْ ...}.

سورة العلق

آية 1–5.

DEDICATION

To those whom burnt and die daily to maintain a better life for us...

Our power supplies…

Our parents.

To our guide and our light in darkness…

Our control system…

Our teachers.

And first of all to our leader…

Dr. SAKHR BABEKIR ABU DARAG.

Acknowledgement

To everybody participates in performing our mission of representing this project in its final form or helps us by his effort, his time, by a note, a comment or any way; we are very thankful to you and very proud with your participation.

ABSTRACT

Many researches have been done in the field of analyzing the performance characteristics for different aircrafts, but by using the analytical approach.

Hence this project is introduced to simulate the performance validation of L-410turbolet by using the drag polar obtained by using computational fluid dynamics (CFD);because of the reliability of this procedure and its high ability to compute and analyze the complex systems. FLUENT package is the program that used to do the computations. The results were realistic and were in consistence with the aerodynamic theories. Also the actual manufacturer engine characteristics data have been employed in simulating the performance validation and in designing the MATLAB software in order to obtain the most proper results and to minimize as possible the error which can results from using only the theoretical formulas of the analytical approach.

Table of contents:

List of Symbols

Symbol	Parameter
D_a	Aerodynamic deceleration
D_{total}	Deceleration with aerodynamic brake
C_f	Friction drag coefficient
R_e	Rynlods number

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CHAPTER ONE

INTRODUCTION

1 ChapterOne: Introduction

1.1: Overview

The airplane did not just “happen”. When you see an air craft in the sky, You are observing the resulting action of the natural law of the nature that govern flight the human understanding of this laws of flight did not come easily. It has evolved over the past 2,500 years, starting with ancient Greek science. It was not until a cold day in December 1903 that these laws were finally harnessed by human begins to a degree sufficient to allow a heavier-than- air, powered, human carrying machine to execute a successful sustained flight through the air. On December 17 of that year, Orville and Wilbur Wright, with pride and great satisfaction, reaped the fruit of their labors and became the first to fly the first successful flying machine. But they had no way of knowing the tremendous extent to which their invention of the first successful airplane was to dominate the course of the twentieth century technically, socially, and politically.

The L-410Turbulet airplane is the subject of this project. The purpose of this project is to pass on to you appreciation of the laws of flight and the embodiment of these laws in a form that allows the understanding and predication of how the airplane will actually perform in the air (airplane performance) and how to approach the creation of the airplane in the first place in order to achieve a desired performance or mission (the creative process of airplane design). The process of airplane performance and airplane design are intimately coupled one did not happened without the other. Therefore, the purpose of this project is to present the elements of the L-410turbolet performance, and to do so in such fashion as to give you both a technical and philosophical understanding of the process.

1.2: Objectives

The main purpose in this project is to use the EXCEL software to design a program for simulating the performance of the L-410turbolet airplane in both the steady level flight under different flight conditions (temperature, altitude, density, and velocity).and to study the unsteady flight of this airplane. And to write a MATLAB code capable to investigate different performance parameters at different flight conditions.

Also a CFD approach is concerned, this approach is limited to exploring a 3D-CAD drawing of the aircraft wing, and then using a standard CFD package (FLUENT) to simulate the floe over aircraft wing airfoils under consideration in order to obtain the actual drag polar.

1.3: Problem statement

The performance simulation of any aircraft can be easily performed by the analytical approach using the traditional formal equations. But the hard task is to predict the exact performance validation by implementing the real engine characteristics data, which is difficult to obtain. The problem statement of this project is to validate the exact performance of the L-410turbolet computationally by designing performance simulation software with MATLAB.

1.4: Proposed solutions

Designing a program to generate a conceptual simulation for the performance validation of the L-410turbolet airplane using the empirical drag polar and the real engine characteristics curve. The software has the capability to investigate different performance parameters at different flight conditions.

1.5: Motivations

An airplane is a flying vehicle. Such as any other machine, it is judged by its performance. Some of the questions that usually come to mind are how fast can it fly? How high can it fly? How fast and how steep can it climb? How far can it go with tank-load of fuel? What length of runway does it need for take-off and landing? How sharp and how far can it turn? Answers to these and many other questions from the subject of this project.

1.6: Contribution

The designed program enables analysis and study of performance characteristics of both steady level flight, and unsteady flight for any airplane when the required data is available.

Also we should express a method by which the drag polar variation could be estimated if not given.

1.7: Project outlines

This project contains five chapters:

Chapter one: introduction.

Chapter two: literature review.

Chapter three: Airplane performance.

Chapter four: Results and discussion.

Chapter five: conclusion and recommendations.

CHAPTER TWO
LITERATURE REVIEW

2.1: Early predications of airplane performance

The airplane of today is a modern work of art and engineering. In turn, the predication of airplane performance is sometimes viewed as relatively modern discipline. However, contrary to intuition, some of the basic concepts have roots deep in history: indeed, some of the very techniques were begin used in practice only a few years after the Wright brothers' successful first flight in 1903. This section traces a few historic paths for some of the basic ideas of airplane performance, as follows:

1. Some understanding of the power required P_R for an airplane was held by Gorge Cayley. He understood the rate of energy lost by an airplane in steady flight under gravitational attraction must be essentially the power must be supplied by an engine to maintain steady, level flight. In 1853 Cayley wrote ;
   The whole apparatus when loaded by a weight equal to that of the man intended ultimately to try the experiment and with the horizontal rudder (the elevator) described in the essay before sent, adjusted so as to regulate the oblique descent from some elevated point, to its proper pitch, it may be expected to skim down, with no force but its own gravitation, in an angle of about 11 degrees with the horizon; or possibly, if well executed, as to direct resistance something less, at a speed of about 36 feet per second, if loaded 1 pound to each square foot of surface. This having by repeated experiments, in perfectly calm weather, been ascertained, for both the safety of the man, and the datum required, let the wings be plied with the man's utmost strength; and let the angle measured by the greater extent of horizontal range of flight be noted; when this point by repeated experiments, has been accurately found.
2. The drag polar, represents simply a plot of C_D versus C_L, illustrating that C_D varies as the square of C_L. Acknowledge of the drag polar is essential to the calculation of the airplane performance. It is interesting that the concept of the drag polar was first introduced by the Frenchman M.Eiffel about 1890. Eiffel was interested in determining the laws of resistance (drag) on bodies of various shapes, and he conducted such drag measurements by dropping bodies from the Eiffel Tower and measuring their terminal velocity.
3. No general understanding of the predication of airplane performance existed before the twentieth century. At best it was understood by the time that lift and drag varied as the first power of the area, and as the second power of velocity, but this doesn't constitute a performance calculation. However this picture radically changed in 1911. In this year the Frenchman Duchene received the Month yon prize from the Paris Academy of sciences for his book entitled The Mechanics of the Airplane: A study of

the Principles of Flight. In this book, the basic elements of airplane performance are put forth for the first time. Duchene gives curves for power required and power available, he discussed airplane maximum velocity; he also gives a relation for the rate of climb. Later, in 1917, Duchene's book was translated into English by John Ledeboer and T.O'B Hubbard. Finally, during 1918-1920, three additional books on airplane performance were written, the most famous being the authoritative Applied Aerodynamics by Leonard Barstow.

- In their project "Aircraft Optimization for Maximum Endurance" Ahmed Mohammed Abd-Naby and his group have show advanced technique to improve the endurance performance for unmanned aerial vehicles, which by use of a constrained objective function minimization technique to maximize the endurance. Referring the objective function and the constraints a design program has been constructed in MATLAB that able to calculate endurance in several iterations using an initial design variables and constraints. Also the program draws diagrams illustrate the optimization process in the various iterations until the new modified design variable are available to give the best optimum endurance value.
- In their project "Performance Analysis For Airplane" Osama AbdallahAlnour and his team have ended from the technical portion and the detailed computer program for the accurate estimation of (sust-91) performance and they have represented the summarized final results.

2.2: Computational methods in air craft design

Computational methods have revolutionized the aircraft design process. Prior to the mid sixties aircraft were designed and built without the benefit of computational tools. Design information was mostly provided by the results of analytic theory combined with a fair amount of experimentation. Analytic theories continue to provide invaluable insight into the trends present in the variation of the relevant parameters in a design.

However, for detailed design work, these theories often lack the necessary accuracy, especially in the presence of non-linearity (transonic flow, large structural deflections, and real-life control systems) with the advent of the digital computer and the fast development of the field of numerical analysis , a variety of complex calculation methods have become available to the designer. Advancements in computational methods have pervaded all disciplines: aerodynamics, structures, propulsion, guidance, control systems integration, multidisciplinary optimization, etc.

The role of computational methods in the aircraft design process is to provide detailed information to facilitate the decisions in the design process at the lowest possible cost and with adequate turnaround (turnaround is the required processing time from the point a piece of information is requested until it is finally available to the designer in a form that allows it to be used). In summary computational methods ought to:

- Allow the simulation of the behavior of complex systems beyond the reach of analytic theory.
- Provide detailed design information in timely fashion.
- Enhance our understanding of engineering systems by expanding our ability to predict their behavior.
- Provide the ability to perform multidisciplinary design optimization.
- Increase competitively and lower design and production costs.

Computational methods are nothing but tools in the aircraft designer's toolbox that allow him to complete a job. In fact, the aircraft designer is often more interested in the interactions between the disciplines that the methods apply to (Aerodynamics, Structure, Control, Propulsion, and mission profile) than the individual methods themselves. This view of the design process is often called multidisciplinary design (one can also term it multidisciplinary computational design). Moreover, a designer often wants to find a combination of design choices for the involved disciplines that produces an overall better airplane. If the computational predication methods for all disciplines are available to the designer, optimization procedures can be coupled to produce Multidisciplinary Design Optimization (MOD) tools. In a nutshell, via a combination of analytic methods and simple computational tools an optimum aircraft design for a specifically chosen mission.

The current status of computational methods is such that the use of certain set of tools has become routine practice at all major aerospace corporations (this includes simple aerodynamic models, linear structural models, and basic control system design). However, a vast amount of work remains to be done in order to make more refined non-linear techniques reach the same routine use status. Moreover, MDO work has been performed using some of the simpler models, but only a few attempts have been made to couple high-fidelity non-linear disciplines to produce optimum designs. Although computational methods are a wonderful resource the process of aircraft design, their misuse can have catastrophic consequences. The following considerations must be always in your mind when you decide to accept as valid the results of computational procedures:

- A solution is only as good as the model that is being solved: if you try to solve a problem with high non-linear content using a computational method designed for linear problems you results will make no sense.
- The accuracy of numerical solution depends heavily on the sophistication of the disceritaization procedure employed and the size of the mesh used. Lower order methods with under resolved meshes provide solutions where the margin error is quite large.
- The range of validity of the results of a given calculation depends on the model that is the heart of the procedure: if you are using an in-viscid solution procedure to approximate the behavior of attached flow, but the actual flow is separated, your results will make no sense.
- Information overload. Computational procedures flood the designer with a wealth of information that sometimes is complete nonsense! When analyzing the results provided by a computational method do not concentrate on how beautiful the color pictures are, be sure that the computational results follow the expected trends.

Let's examine the status of the more relevant aerospace disciplines to which computational methods have been applied. These include applied aerodynamics, structural analysis, and control system design.

Computational methods first began to have a significant impact on aerodynamics analysis and design in the period of 1965-75. This decade saw the introduction of panel methods which could solve the linear flow models for arbitrarily complex geometry in both subsonic and supersonic flow. It also saw the appearance of the first satisfactory methods for treating the non-linear equations of transonic flow, and the development of the hodograph method for design of the shock free supercritical airfoils.

Panel methods are based on the distribution of surface singularities on a given configuration of interest, and gained wide-spread acceptance throughout the aerospace industry. They have achieved their popularity largely due to the fact that the problems can be easily setup and solutions can be obtained rather quickly on today's desktop computers. The calculation of potential flows around bodies was first realized with the advent of the surface panel methodology originally developed at the Douglas Company. During the years, additional capability was added t these surface panel methods. These additions includes the use of higher order, more accurate formulations, the introduction of lifting capability, the solution of unsteady flows, and the coupling with various boundary layer formulations.

Panel methods lie at the bottom of complexity pyramid for the solution of aerodynamic problems. They represent a versatile and useful method to obtain a good approximation to a flow field in a very short time. Panel methods, however, cannot offer accurate solutions for a variety of high-speed non-linear flows of interest to the designer. For these kinds of flows, a more sophisticated model of the flow equations is required. Efficient flight is generally achieved by the use of smooth and streamlined shapes which avoid flow separation and minimize viscous effects; with the consequence that useful predication can be made using in-viscid models. In-viscid calculations with boundary layer corrections can provide quite accurate prediction of lift and drag when the flow remains attached, but the iteration between the in-viscid outer solution and the inner boundary layer solution becomes increasingly difficult with the onset of separation. Procedures for solving the full viscous equations are likely to be needed for the simulation of arbitrary complex separated flows, which may occur at high angles of attack or with bluff bodies. In order to treat flows at high Reynolds number. Turbulent effects by Reynolds averaging of the fluctuating …..

Elham Hassan and her work team in their project "Analyzing Flow over the Wing Using CFD" have analyze the flow over NACA 23012 airfoil and over finite wing using Computational Fluid Dynamics (CFD). The results were good enough to be used in design processes, and were inconsistence with the aerodynamic theories.

2.310turbolet airplane

The L 410 aircraft is an all metal upper-wing aircraft powered by two turboprop engines GE H80-200, which is configured as a civil version for 19 passengers, transport or special mission aircraft. L 410 aircraft are operated in more than 50 countries on five continents. Most of them were delivered to Russia, others to Africa, South East Asia, and Southern America and also to Europe. The manufacturer has produced more than 1,100 L 410 aircraft so far. At present, the current version L 410 UVP E20 is on the market as a successor of L 410 family. Its unique flight characteristics and operational benefits are fully recognized such as follows:

- Short take off and landing on short airstrips
- Operation from unpaved runway
- Low operational and maintenance costs
- Excellent performance in high temperature and high altitude environment
- Wide variety of equipment with possible installation of special removable kits
- Large cargo compartments
- Exceed safety standards and easy flight operation

The most spacious cabin in its aircraft category

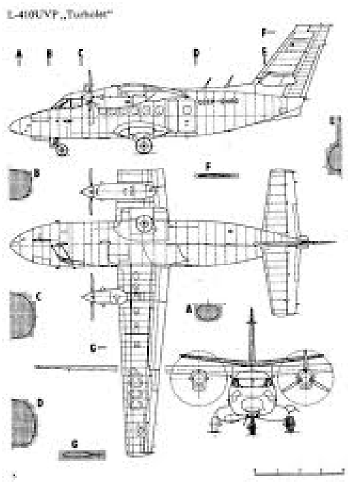
The L410 is very successful Czech commuter which was first built in response to Soviet requirements, but has sold widely around the globe.

First design studies of the original 15 seat L 410 began in 1966. The resulting conventional design was named the Turbo let, and was developed to be capable of operations from unprepared strips. The power plant chosen was the all new Walter or Motor let M 601, but this engine was not sufficiently developed enough to power the prototypes, and Pratt & Whitney Canada PT6A27s were fitted in their place. First flight occurred on April 16 1969, and series production began in 1970. Initial production L 410s were also powered by the PT6A, and it was not until 1973 that production aircraft L 410Ms featured the M 601.

The basic L 410 was superseded from 1979 by the L 410 UVP with a 47cm (1ft 7in) fuselage stretch, M 601B engines and detail refinements. The UVP was in turn replaced by the M 601E powered UVPE in which the toilet and baggage compartments were repositioned allowing more efficient seating arrangements for up to 19 passengers. The UVPE is the current production version.

The L 420 is an improved variant with more powerful M 601F engines, higher weights and improved performance designed to meet western certification requirements. It first flew on November 10 1993 and was awarded US FAA certification in May 1998.

Ayres took control of Let in September 1998 and plans to further develop the 410/420 line.



CHAPTER THREE

AIRPLANE PERFORMANCE

3.1: Introduction

The aerodynamic forces and moments exerted on a body moving through a fluid stem from two sources:

1. The pressure distribution
2. The shear stress distribution

Both acting over the body surface. The physical laws governing such phenomena were examined, with various applications to aerodynamic flows. The airplane will be considered a rigid body on which is exerted four natural forces: lift, drag, propulsive thrust, and weight. Concern will be focused on the movement of the airplane as it responds to these forces. Such considerations form the core of flight dynamics, an important discipline of aerospace engineering. Studies of airplane performance fall under the heading of flight dynamics.

In these studies, we will no longer be concerned with aerodynamic details; rather, we will generally assume that the aerodynamicists have done their work and provided us with the pertinent aerodynamic data for a given airplane. These data are usually packaged in the form of a drag polar for the complete airplane, given as

$$C_D = C_{D.e} + \frac{C_:^2}{\pi e AR} \quad (3.1a)$$

Here, C_D is the drag coefficient for the complete airplane; C_L is the total lift coefficient, including the small contributions from the horizontal tail and fuselage; and C_D is defined as the parasite drag coefficient, which contains not only the profile drag of the wing, but also the friction and pressure drag of the tail surfaces, fuselage, engine nacelles, landing gear, and any other component of the airplane which is exposed to the airflow. At transonic: and supersonic speeds, $C_{D.e}$ also contains wave drag. Because of changes in the flow field around the airplane -especially changes in the amount of separated flow over parts of the airplane – as the angle of attack is varied, $C_{D.e}$ will change with angle of attack; that is, $C_{D.e}$ is itself a function of lift coefficient. A reasonable approximation for this function is

$$C_D = C_{D.0} + rC_L^2$$

Where r is an empirically determined constant. Hence. Eq. (3.la) can be written as

$$C_D = C_{D.0} + \left(r + \frac{1}{\pi eAR}\right) C_L^2 \tag{3.1b}$$

In Eqs. (3.1a) and (3.1b), e is the familiar span efficiency factor, which takes into account the nonelliptical lift distribution on wings of general shape. Let us now redefine e such that it also includes the effect of the variation of parasite drag with lift; i.e., let us write Eq. (3.1b) in the form

$$C_D = C_{D.0} + \left(\frac{C_L^2}{\pi eAR}\right) \tag{3.1c}$$

Where $C_{D.0}$ is the parasite drag coefficient at zero lift and the term $C_L^2/\pi eAR$includes both induced drag and the contribution to parasite drag due to lift. In Eq. (3.1c), our redefined e, which now includes the effect of r from Eq. (3.1b), is called the Oswald efficiency factor (named after W. Bailey Oswald, who first established this terminology in NACA report no. 408 in l933). In this chapter, the basic aerodynamic properties of the airplane will be described by Eq. (3.1c). and we will consider both $C_{D.0}$ and e as known aerodynamic quantities, obtained from the aerodynamicist. We will continue to designate $C_L^2/\pi eAR$ by $C_{D.0}$,where $C_{D.0}$ now has the expanded interpretation as the coefficient of drag due to lift, including both the contributions due to induced drag and the increment in parasite drag due to angle of attack different than$\alpha = \alpha_L$. For compactness, wewill designate $C_{D.0}$ as simply the parasite drag coefficient, although we recognize to be more precisely the parasite drag coefficient at zero lift, i.e., the value of $C_{D.0}$when$a = a_{L=0}$.

To study the performance of an airplane, the fundamental equations, which govern its translational motion through the air, must first be established, as follows.

3.2: Equations of Motion

Consider an airplane in flight. The flight path (direction of motion of the airplane) is inclined at an angle θ with respect to the horizontal. The flight path direction and the relative wind are along the same line. The mean chord line is at a geometric angle of attack $\alpha$with respect to the flight path direction. There are four physical forces acting on the airplane:

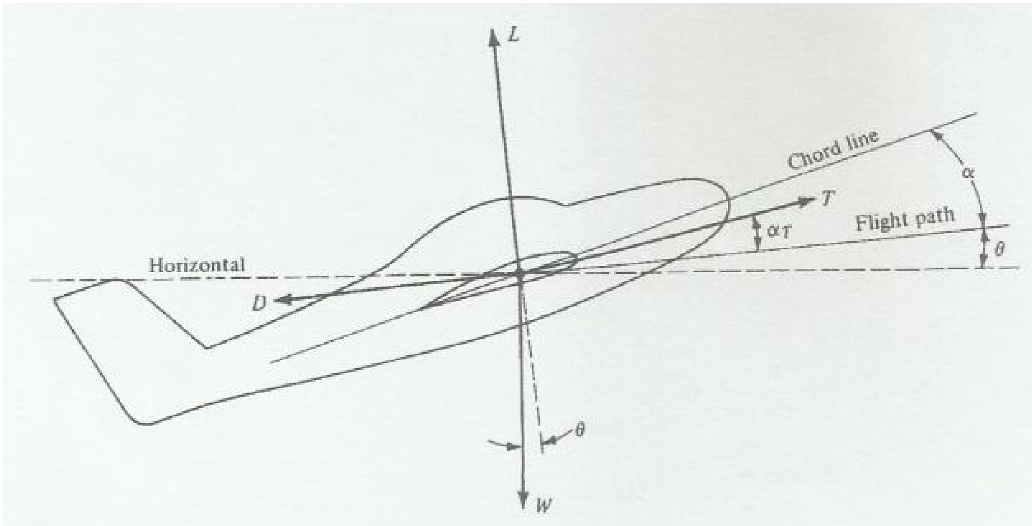


Figure 3.1 Force diagram for an airplane in flight.

1. Lift L, which is perpendicular to the flight path direction
2. Drag D, which is parallel to the fight path direction
3. Weight W, which acts vertically toward the center of the earth (and hence is inclined at the angle $\theta$with respect to the lift direction)
4. Thrust T, which in general is inclined at the angle α_T with respect to the flightpath direction

When an object moves along a curved path, the motion is called curvilinear, as opposed to motion along a straight line, which is rectilinear. Newton's second law, which is a physical statement that force = mass X acceleration, holds in either case. Consider a curvilinear path. At a given point on the path, set up two mutually perpendicular axes, one along the direction of the flight path and the other normal to the fight path. Applying Newton's law along the flight path,

$$\sum F_{||} = ma = m\frac{dV}{dt} \tag{3.2}$$

where$\sum F_{||}$is the summation of all forces parallel to the flight path, $a = dV/dt$isthe acceleration along the flight path, and V is the instantaneous value of the airplane's flight velocity. (Velocity V is always along the flight path direction, by definition.) Now, applying Newton's law perpendicular to the flight path, we have

$$\sum F_{\perp} = m\frac{V^2}{r_c} \tag{3.3}$$

where$\sum F_{\perp}$ is the summation of all forces perpendicular to the flight path and$\frac{V^2}{r_c}$ is the acceleration normal to a curved path with radius of curvaturer_c. Thisnormal acceleration V^2/r_c should be familiar from basic physics. The right-handside of Eq. (3.3) is nothing other than the "centrifugal force."

$$\sum F_{\parallel} = T\cos\alpha_{\mathrm{T}} - D - W\sin\theta \tag{3.4}$$

And the forces perpendicular to the flight path (positive upward and negative downward) are

$$\sum F_{\perp} = L + T\sin\alpha_T - W\cos\theta \tag{3.5}$$

Combining Eq. (3.2) with (3.4) and (3.3) with (3.5) yields

$$T\cos\alpha_T - D - W\sin\theta = m\frac{dV}{dt} \tag{3.6}$$

$$L + T\sin\alpha_T - W\cos\theta = m\frac{V^2}{r_c} \tag{3.7}$$

Equations (3.6) and (3.7) are the equations of motion for an airplane in translational flight. (Note that an airplane can also rotate about its axes. Also note that we are not considering the possible sidewise motion of the airplane.

Equations (3.6) and (3.7) describe the general two-dimensional translational motion of an airplane in accelerated flight. The performance of an airplane for such unaccelerated flight conditions is called static performance. This may, at first thought, seem unduly restrictive; however, static performance analyses lead to reasonable calculations of maximum velocity, maximum rate of climb, maximum range, etc. — parameters of vital interest in airplane design and operation.

With this in mind, consider level, unaccelerated flight. Level flight means that the flight path is along the horizontal: that is,$\theta = 0$. Unaccelerated flight means that the right-hand sides of Eqs. (3.6) and (3.7) are zero. Therefore, these equations reduce to

$$T\cos\alpha_T = D \tag{3.8}$$

$$L + T\sin\alpha_T = W \tag{3.9}$$

Furthermore, for most conventional airplanes, α_T is small enough that $\cos\alpha_T = 1$andsin $\alpha_T = 0$. Thus, from Eq. (3.8) and (3.9),

$$T = D \tag{3.10}$$

$$L = W \tag{3.11}$$

Equations (3.10) and (3.11) are the equations of motion for level, unaccelerated flight. In level, unaccelerated flight, the aerodynamic drag is balanced by the thrust of the engine, and the aerodynamic lift is balanced by the weight of the airplane-almost trivial, but very useful, results.

3.3: Thrust Required For Level, Unaccelerated Flight

Consider an airplane in steady level fight at a given altitude and a given velocity. For flight at this velocity, the airplane's power plant (e.g., turbojet engine or reciprocating engine—propeller combination) must produce a net thrust which is equal to the drag. The thrust required to obtain a certain steady velocity is easily calculated as:

$$T = D = q_\infty S C_D \tag{3.12}$$

$$L = W = q_\infty S C_L \tag{3.13}$$

$$\frac{T}{W} = \frac{C_D}{C_L} \tag{3.14}$$

Thus, from Eq. (3.14), the thrust required for an airplane to fly at a given velocity in level, unaccelerated flight is

$$T_R = \frac{W}{C_L/C_D} = \frac{W}{L/D} \tag{3.15}$$

(Note that a subscript R has been added to thrust to emphasize that it is thrust required.)Thrust required T_R for a given airplane at a given altitude, varies with velocityV_∞.

3.4: Thrust Available and Maximum Velocity

Thrust requiredT_R, it dictated by the aerodynamics and weight of the airplane itself, it is an airframe-associated phenomenon. In contrast, the thrust available T_A is strictly associated with the engine of the airplane; it is the propulsive thrust provided by an engine-propeller combination, a turbojet, a rocket, etc. Suffice it to say here that reciprocating piston engines with propellers exhibit a variation of thrust with velocity. Thrust at zero velocity (static thrust) is a maximum and decreases with forward velocity. At near-sonic flight speeds, the tips of the propeller blades encounter the same compressibility problems and the thrust available rapidly deteriorates. On the other hand, the thrust of a turbojet engine is relatively constant with velocity. These two power plants are quite common in aviation today; reciprocating engine-propeller combinations power the average light, general aviation aircraft, whereas the jet engine is used by almost all large commercial transports and military combat aircraft.

3.5: Power Required For Level, Unaccelerated Flight

Power is a precisely defined mechanical term; it is energy per unit time. The power associated with a moving object can be illustrated by a block moving at constant velocity V under the influence of the constant force F.

$$Power = \frac{energy}{time} = \frac{force \times distance}{time} = force \times \frac{distence}{time}$$

$$power = F\left(\frac{d}{t_2 - t_1}\right) = FV \tag{3.16}$$

Where $d/(t_2 - t)$ is the velocity V of the object. Thus, Eq. (3.16) demonstrates that the power associated with a force exerted on a moving object is force *X* velocity, an important result.

Consider an airplane in level, unaccelerated flight at a given altitude and with velocityV_∞. The thrust required isT_R. From Eq. (3.16) the power required P_R is therefore

$$P_R = T_R V_\infty \tag{3.17}$$

The effect of the airplane aerodynamics ($C_L$andC_D) on P_R is readily obtained by

$$P_R = T_R V_\infty = \frac{W}{C_2/C_D} V_\infty \tag{3.18}$$

$$L = W = q_\infty S C_L = \frac{1}{2}\rho_\infty V_\infty^2 S C_L$$

Hence

$$V_\infty = \sqrt{\frac{2W}{\rho_\infty S C_L}} \tag{3.19}$$

$$P_R = \frac{W}{C_L/C\_D}\sqrt{\frac{2W}{\rho_\infty S C_L}}$$

$$P_R = \sqrt{\frac{2W^3 C_D^2}{\rho_\infty S C_L^3}}\,\alpha\,\frac{1}{C_L^{3/2}/C_D} \tag{3.20}$$

In contrast to thrust required, which varies inversely as C_L/C_D power required varies inversely as$C_L^{3/2}/C_D$.

The power-required curve is defined as a plot ofP_R, note that it qualitatively resembles the thrust-required curve of. As the airplane velocity increases, P_R first decreases, goes through aminimum, and then increases. At the velocity for minimum power required, theairplane is flying at the angle of attack which corresponds to a maximum$C_L^{3/2}/C_D$.

We demonstrated that minimum T_R aerodynamically correspondsto equal parasite and induced drag. An analogous but different relation holds at minimum P_R

$$P_R + T_R V_\infty = D V_\infty = q_\infty S\left(C_{D.0} + \frac{C_L^2}{\pi e AR}\right)V_\infty$$

$$P_R = \underset{\text{Parasite power required}}{q_\infty S C_{D.0} V_\infty} + \underset{\text{Induced power required}}{q_\infty S V_\infty \frac{C_L^2}{\pi e AR}} \tag{3.21}$$

Therefore, as in the earlier case ofT_R, the power required can be split into therespective contributions needed to overcome parasite drag and drag due to lift.Also as before, the aerodynamic conditions associated with minimum R can be obtained from Eq. (3.21) by setting $dP_R/dV_\infty = 0$. To do this, first obtain Eq. (3.21) explicitly in terms of V_∞

Recalling that $q_\infty = \frac{1}{2}\rho V_\infty^2$ and $C_L = W/\frac{1}{2}P\rho_\infty V_\infty^2 S$.

$$P_R = \frac{1}{2}\rho_\infty V_\infty^3 S \frac{\left(W/\frac{1}{2}\rho_\infty V_\infty^2 S\right)^2}{\pi e AR}$$

$$P_R = \frac{1}{2}\rho_\infty V_\infty^3 S C_{D.0} + \frac{\left(W^2/\frac{1}{2}\rho_\infty V_\infty^2 S\right)^2}{\pi e AR} \tag{3.22}$$

For minimum power required, $dP_R/dV_\infty = 0$ Differentiating Eq. (6.28) yields

$$\frac{dP_R}{dV_\infty} = \frac{1}{2}\rho_\infty V_\infty^2 S C_{D.0} - \frac{W^2/\frac{3}{4}\rho_\infty V_\infty^2 S}{\pi e AR}$$

$$= \frac{3}{2}\rho_\infty V_\infty^2 S\left(C_{D.0} - \frac{W^2/\frac{3}{4}\rho_\infty^2 S^2 V_\infty^4}{\pi e AR}\right)$$

$$= \frac{3}{2}\rho_\infty V_\infty^2 S\left(C_{D.0} - \frac{\frac{1}{3}C_L^2}{\pi e AR}\right)$$

$= \frac{3}{2}\rho_\infty V_\infty^2 S\left(C_{D.0} - \frac{1}{3}C_{D.i}\right) = 0$For minimum P_R

Hence, the aerodynamic condition that holds at minimum power required is

$$C_{D.0} = \frac{1}{3}C_{D.0}$$

The point on the power-required curve that corresponds to minimum $T_R$iseasily obtained by drawing a line through the origin and tangent to the P_R curve. The point of tangency corresponds to minimum T_R (hence maximumL/D).

3.6: Power Available and Maximum Velocity

Note again that P_R is a characteristic of the aerodynamic design and weight of the aircraft itself. In contrast, the power available P_A is a characteristic of the power plant. However, the following comments are made to expedite our performance analyses.

A Reciprocating Engine-Propeller Combination

A piston engine generates power by burning fuel in confined cylinders and using this energy to move pistons, which in turn deliver power to the rotating crankshaft, The power delivered to the propeller by the crankshaft is defined as the shaft brake power P, (the word "brake" stems from a method of laboratory testing which measures the power of an engine by loading it with a calibrated brake mechanism). However, not all of *P* is available to drive the airplane; some of it is dissipated by inefficiencies of the propeller itself. Hence, the power available to propel the airplane, P_A, is given by

$$P_A = \eta P \tag{3.24}$$

Whereη is the propeller efficiency,$\eta < 1$. Propeller efficiency is an importantquantity and is a direct product of the aerodynamics of the propeller. It is alwaysless than unity. For our discussions here, both$\eta$and *P* are assumed to be known quantities for a given airplane.

A remark on units is necessary. In the engineering system, power is in$ft.lb/s$; in the SI system, power is in watts [which are equivalent to N.m/s. However, the historical evolution of engineering has left us with a horrendously inconsistent (but very convenient) unit of power, which is widely used, namely, horsepower. All reciprocating engines are rated in terms of horsepower and it is important to note that

1 horsepower = 550 ft.lb/s

=746 W

Therefore, it is common to use shaft brake horsepower bhp in place of *P*, and horsepower available hp_A in place of P_A. Equation (3.24) still holds in the form

$$hp_A = (\eta)(bhp) \tag{3.24}$$

3.7: Altitude Effects on Power Required and Available

With regard to P_R, curves at altitude could be generated by repeating the calculations of the previous sections, with ρ appropriate to the given altitude. However, once the sea-level P_R curve is calculated by means of this process, the curves at altitude can be more quickly obtained by simple ratios, as follows. Let the subscript "0" designate sea-level conditions.

$$V_0 = \sqrt{\frac{2w}{\rho_0 S C_L}} \tag{3.25}$$

$$P_{R.0} = \sqrt{\frac{2W C_D^2}{\rho_0 S C_L^2}} \tag{3.26}$$

whereV_0, $P_{R.0}$, and ρ_0 are velocity, power, and density at sea level. At altitude, where the density is ρ, these relations are

$$V_0 = \sqrt{\frac{2w}{\rho S C_L}} \tag{3.27}$$

$$P_{R.0} = \sqrt{\frac{2W^3 C_D^2}{\rho_0 S C_L^3}} \tag{3.28}$$

Now, strictly for the purposes of calculation, Let$C_L$remain fixed between sea level and altitude. Hence, because $C_D = C_{D.0} + C_L^2/\pi eAR$ $C_D$also remains fixed.

$$V_{alt} = V_0 \left(\frac{\rho_0}{\rho}\right)^{1/2} \tag{3.29}$$

and

$$P_{alt} = P_{R.0} \left(\frac{\rho_0}{\rho}\right)^{1/2} \tag{3.30}$$

3.8: Rate of Climb

Consider an airplane in steady, unaccelerated, climbing flight. The velocity along the flight path isV_∞ and the flight path itself is V_∞

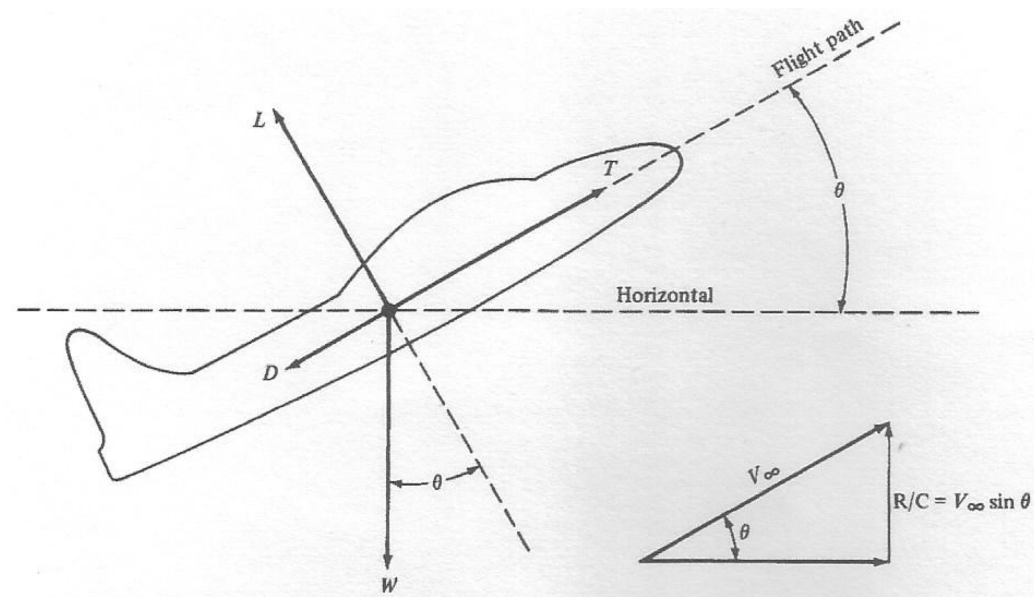


Figure 3.2 Airplane in climbing flight.

Inclined to the horizontal at the angleθ. As always, lift and drag are perpendicular and parallel to V_∞, and the weight is perpendicular to the horizontal. Thrust Tis assumed to be aligned with the flight path. Here, the physical difference from our previous discussion on level flight is that *T* is not only working to overcome the drag, but for climbing flight it is also supporting a component of weight. Summing forces parallel to the flight path, we get

$$T = D + W \sin\theta \tag{3.31}$$

And perpendicular to the flight path, we have

$$L = W \cos\theta \tag{3.32}$$

Note that the lift is now smaller than the weight.

$$TV_\infty = DV_\infty + WV_\infty \sin\theta$$

$$\frac{TV_\infty - DV_\infty}{W} = V_\infty \sin\theta \qquad (3.33)$$

$V_\infty \sin\theta$, is the airplane's vertical velocity. This vertical velocity is called the rate of climb R/C:

$$R/C = V_\infty \sin\theta \qquad (3.34)$$

$TV_\infty$is the power available. The second term is DV_∞, which for level flight is the power required. For climbing flight, however, DV_∞ isno longer precisely the power required, because power must be applied toovercome a component of weight as well as drag. Nevertheless, for small angles of climb, say 0 <200, it is reasonable to neglect this fact and to assume that theDV_∞term is given by the level-flightP_R. With this,

$$TV_\infty = DV_\infty = excess\ power$$

Where the excess power is the difference between power available and power required, for propeller-driven and jet-powered aircraft, respectively, we obtain

$$R/C = \frac{excesspower}{W} \qquad (3.36)$$

3.9: Gliding Flight

Consider an airplane in a power-off glide. The forces acting on this aircraft are lift, drag, and weight; the thrust is zero because the power is off. The glide flight path makes an angle θbelow the horizontal. For an equilibriumunaccelerated glide the sum of the forces must be zero.

Summing forces along the flight path, we have,

$$D = W \sin\theta \qquad (3.37)$$

and perpendicular to the flight path

$$L = W \cos\theta \qquad (3.38)$$

The equilibrium glide angle can be calculated by

$$\frac{\sin\theta}{\cos\theta} = \frac{D}{L}$$

or

$$\tan\theta = \frac{1}{L/D} \qquad (3.39)$$

Clearly, the glide angle is strictly a function of the lift-to-drag ratio; the higher the L/D, the shallower the glide angle. From this, the smallest equilibrium glide angle occurs at$(L/D)_{max}$, which corresponds to the maximum range for the glide.

3.10: Absolute and Service Ceilings

The effects of altitude on $P_A$and P_R. For the sake of discussion, consider a propeller-driven airplane: the results of this section will be qualitatively the same for a jet. As altitude increases, the maximum excess power Decreases. In turn, maximum R/C decreases.

There is some altitude high enough at which the P4 curve becomes tangent to the P_R curve. The velocity at this point is the only value at which steady, level flight is possible; moreover, there is zero excess power, hence

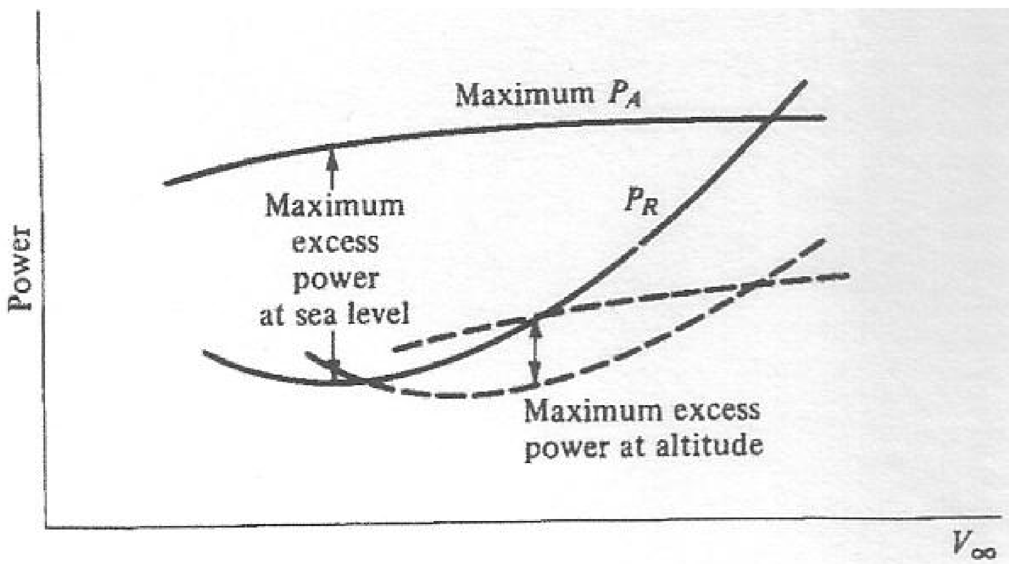


Figure 3.3 Variation of excess power with altitude.

zero maximum rate of climb, at this point. The altitude at which maximum $R/C = 0$ is defined as the absolute ceiling of the airplane. A more useful quantity is the service ceiling, defined as that altitude where the maximum R/C = 100 ft/min. The service ceiling represents the practical upper limit of steady, level flight.

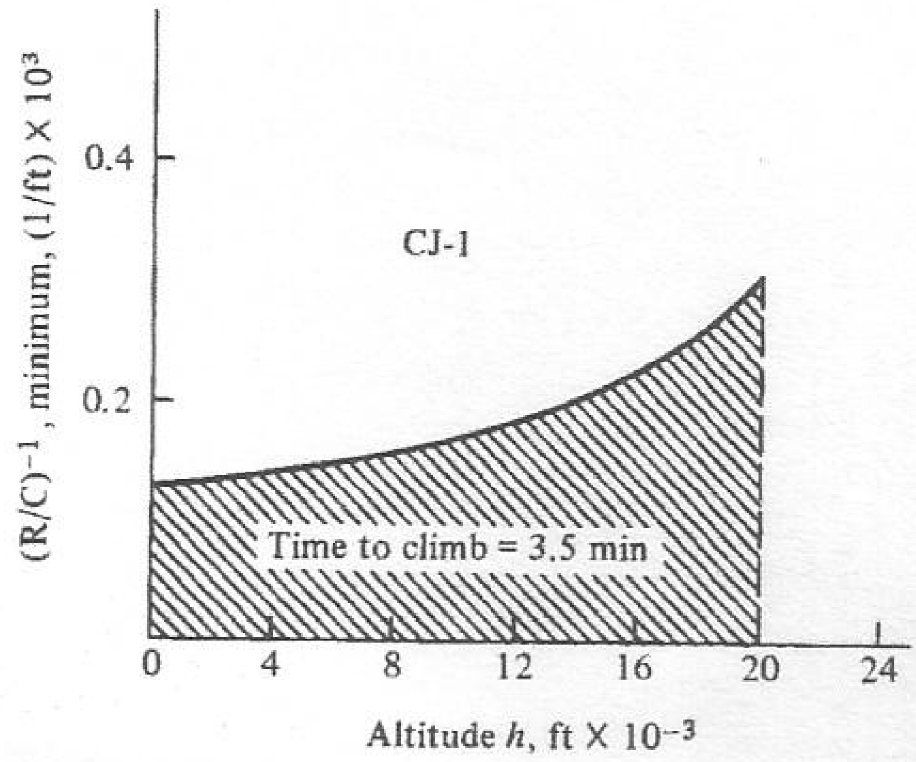


Figure 3.4 Determination of time to climb

Note that the CJ-1 climbs to 20,000 ft in one-eighth of the time required by the CP-1; this is to be expected for a high-performance executive jet transport in comparison to its propeller-driven piston engine counterpart.

3.11: Range and Endurance-Propeller-Driven Airplane

Range is technically defined as the total distance (measured with respect to the ground) traversed by the airplane on a tank of fuel. A related quantity is endurance, which is defined as the total time that an airplane stays in the air on a tank of fuel. In different applications, it may be desirable to maximize one or the other of these characteristics. The parameters which maximize range are different from those which maximize endurance; they are also different for propeller- and jet-powered aircraft. The purpose of this section is to discuss these variations for the case of a propeller-driven airplane; jet airplanes will be considered in the next section.

A Physical Consideration

One of the critical factors influencing range and endurance is the specific fuel consumption, a characteristic of the engine. For a reciprocating engine, specific fuel consumption (commonly abbreviated as SFC) is defined as the weight of fuel consumed per unit power per unit time. As mentioned earlier, reciprocating engines are rated in terms of horsepower, and the common units (although inconsistent) of specific fuel consumption are

$$SFC = \frac{Ib\ of\ fuel}{(bhp)(h)}$$

Where (bhp) signifies shaft brake horsepower.

First, consider endurance. On a qualitative basis, in order to stay in the air for the longest period of time, common sense says that we must use the minimum number of pounds of fuel per hour. On a dimensional basis this quantity is proportional to the horsepower required by the airplane and to the SFC:

$$\frac{Ib\ of\ fuel}{h} = (SFC)(hp_R)$$

Therefore, minimum pounds of fuel per hour is obtained with minimum hp_R. Since minimum pounds of fuel per hour gives maximum endurance, we quickly conclude that

Maximum endurance for a propeller-driven airplane occurs when the airplane is flying at minimum power required.

Maximum endurance for a propeller-driven airplane occurs when the airplane is flying at a velocity such that $C_L^{3/2}/C_D$ is maximum.

Now, consider range. In order to cover the longest distance (say, in miles), common sense says that we must use the minimum number of pounds of fuel per mile. On a dimensional basis, we can state the proportionality

$$\frac{Ib\ of\ fuel}{mi} \propto \frac{(SFC)(hp_R)}{V_\infty}$$

(Check the units yourself; assuming V is in miles per hour.) As a result, minimum pounds of fuel per mile are obtained with a minimumhp_R/V_∞. This minimum value of hp_R/V_∞ precisely corresponds to maximum L/D.

Maximum range for a propeller-driven airplane occurs when the airplane is flying at a velocity such that C_L/C_D is maximum.

B Quantitative Formulation

The important conclusions written above in italics were obtained from purely physical reasoning. We will develop quantitative formulas which substantiate these conclusions and which allow the direct calculation of range and endurance for given conditions.

In this development, the specific fuel consumption is couched in units that are consistent, i.e.,

$$\frac{lb\ of\ fuel}{\left(ft \times \frac{lb}{s}\right)(s)} \quad or \quad \frac{N\ of\ fuel}{(j/s)\ (s)}$$

For convenience and clarification, y will designate the specific fuel consumption with consistent units.

Consider the product *cPdt*, where P is engine power and *dt* is a small increment in time. The units of this product are (in the English engineering system)

$$cP\ dt = \frac{Ib\ of\ fuel}{(ft \times Ib/s)(s)} \frac{ft \times Ib}{s}(s) = Ibof\ fuel$$

Therefore, *cPdt* represents the differential change in the weight of the fuel due to consumption over the short time period *dt*. The total weight of the airplane, *W*, is the sum of the fixed structural and payload weights, along with the changing fuel weight. Hence, any change in *W*is assumed to be due to the change in fuel weight. Recall that *W* denotes the weight of the airplane at any instant. Also, letW_0= gross weight of the airplane (weight with full fuel and payload), *Wf* = weight of the fuel load, and W_1= weight of the airplane without fuel. With these considerations, we have

$$W_1 = W_0 - W_f$$

and $$dW_f = dW = -cP\ dt$$

or $$dt = -\frac{dW}{cP} \tag{3.40}$$

The minus sign in Eq. (3.40) is necessary because *dt* is physically positive (time cannot move backward, except in science fiction novels), while at the same time W is decreasing (hence *dW* is negative). Integrating Eq. (3.40) between time r = 0, where W = W_0, (fuel tanks full), and time t = E, where W = W_1 (fuel tanks empty), we find

$$\int_0^E dt = \int_{W_0}^{W_1} \frac{dW}{cP}$$

$$E = \int_{W_1}^{W_0} \frac{dW}{cP} \tag{3.41}$$

In Eq. (3.41), E is the endurance in seconds.

To obtain an analogous expression for range, multiply Eq. (3.40) by V_∞:

$$v_\infty . \mathrm{dt} = -\frac{V_\infty dW}{cP} \tag{3.42}$$

In Eq. (3.42), $V_\infty dt$ is the incremental distance *ds* covered in timedt.

$$dt = -\frac{V_\infty dW}{cP} \tag{3.43}$$

The total distance covered throughout the flight is equal to the integral of Eq.(3.43) from s = 0, where W = W_0 (full fuel tank), to s = R, where W = W_1(empty fuel tank):

$$\int_0^R ds = \int_{W_0}^{W_1} \frac{V_\infty dW}{cP}$$

Or

$$R = \int_{W_1}^{W_0} \frac{V_\infty dW}{cP}$$

C Breguet Formulas (Propeller-Driven Airplane)

For level, unaccelerated flight, $P_R = DV_\infty$. Also, to maintain steady conditions, the pilot has adjusted the throttle such that power available from the engine-propeller combination is just equal to the power required: $P_A = P_R = DV_\infty$

$$P = \frac{P_A}{\eta} = \frac{DV_\infty}{\eta} \tag{3.45}$$

$$R = \int_{W_1}^{W_0} \frac{V_\infty dW}{cP} = \int_{W_1}^{W_0} \frac{V_\infty \eta dW}{cDV_\infty} = \int_{W_1}^{W_0} \frac{\eta dW}{cD} \tag{3.46}$$

$$R = \int_{W_1}^{W_0} \frac{\eta}{cD}\frac{W}{W} dW = \int_{W_1}^{W_0} \frac{\eta}{c}\frac{L}{D}\frac{dW}{W} \tag{3.47}$$

of level, unaccelerated flight. However, for practical use, it will be further simplified by assuming that,$L/D = C_L/C_D$, and c are constant throughout the flight. This is a reasonable approximation for cruising flight conditions. Thus, Eq. (3.47) becomes

$$R = \frac{\eta}{c}\frac{C_L}{C_D}\int_{W_1}^{W_0} \frac{dW}{W}$$

$$\boxed{R = \frac{\eta}{c}\frac{C_L}{C_D}\ln\frac{W_0}{W_1}} \tag{3.48}$$

Equation (3.48) is a classic formula in aeronautical engineering; it is called the *Breguet range formula*, and it gives a quick, practical estimate for range which is generally accurate to within 10 to 20 percent.

3.12: NACA Cowling and the Fillet

The radial piston engine came into wide use in aviation during and after World War1, a radial engine has its pistons arranged in a circular fashion about the crankshaft, and the cylinders themselves are cooled by airflow over the outer finned surfaces. Until 1927, these cylinders were generally directly exposed to the main airstream of the airplane. As a result, the drag on the engine-fuselage combination was inordinately high. The problem was severe enough that a group of aircraft manufacturers met at Langley Field on May 24, 1927, to urge NACA to undertake an investigation of means to reduce this drag. Subsequently, under the direction of Fred E. Wick, an extensive series of tests was conducted in the Langley 20-ft propeller research tunnel using a Wright Whirlwind J-5 radial engine mounted to a conventional fuselage. In these tests, various types of aerodynamic surfaces, called cowlings, were used to cover, partly or completely, the engine cylinders, directly guiding part of the airflow over these cylinders for cooling purposes but at the same time not interfering with the smooth primary aerodynamic flow over the fuselage.

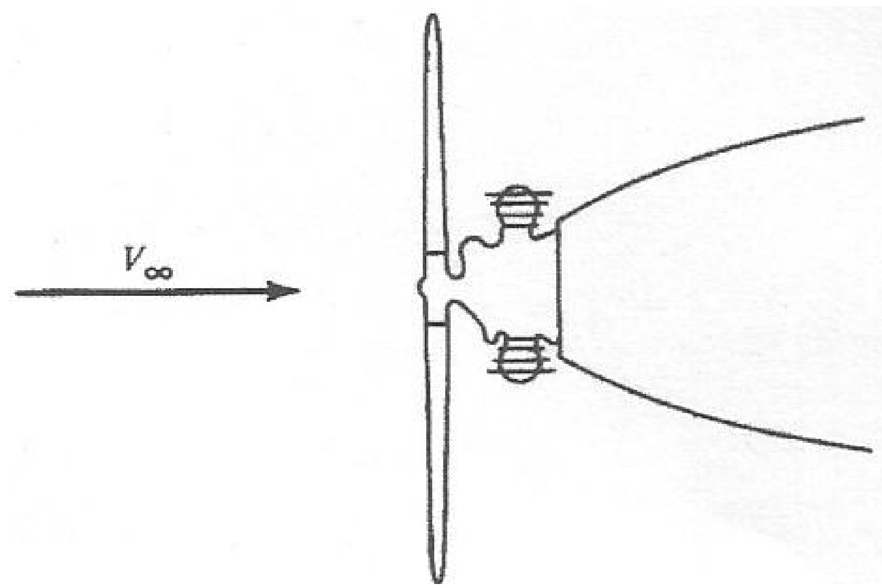


Figure 3.5 Engine mounted with no cowling.

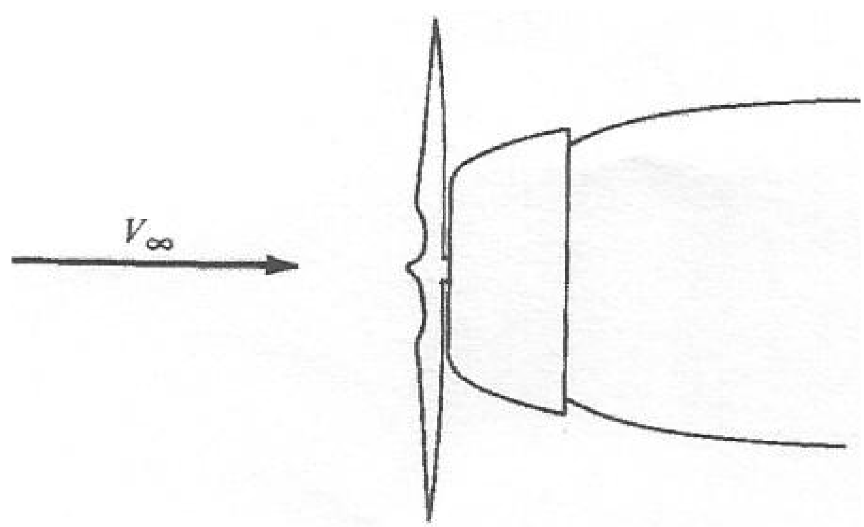


Figure 3.6 Engine mounted with full cowling.

Best cowling, completely covered the engine. The results were dramatic! Compared with the uncowled fuselage, a full cowling reduced the drag by a stunning 60 percent! Taken directly from Wick's report entitled "Drag and Cooling with Various Forms of Cowling for a Whirlwind Radial Air-Cooled Engine," NACA technical report no. 313, published in 1928. After this work, virtually all radial engine-equipped airplanes since 1928 have been designed with a full NACA cowling. The development of this cowling was one of the most important aerodynamic advancements of the1920s; it led the way to a major increase in aircraft speed and efficiency.

A few years later, a second major advancement was made, but by a completely different group and on a completely different part of the airplane. In the early 1930s, the California Institute of Technology at Pasadena, California, established a program in aeronautics under the direction of Theodore vonKarman. Von Karman, a student of Ludwig Prandil, became probably the leading aerodynamicist of the 1920-1960 time-periods. At Caltech, von Karman established an aeronautical laboratory of high quality, which included a large

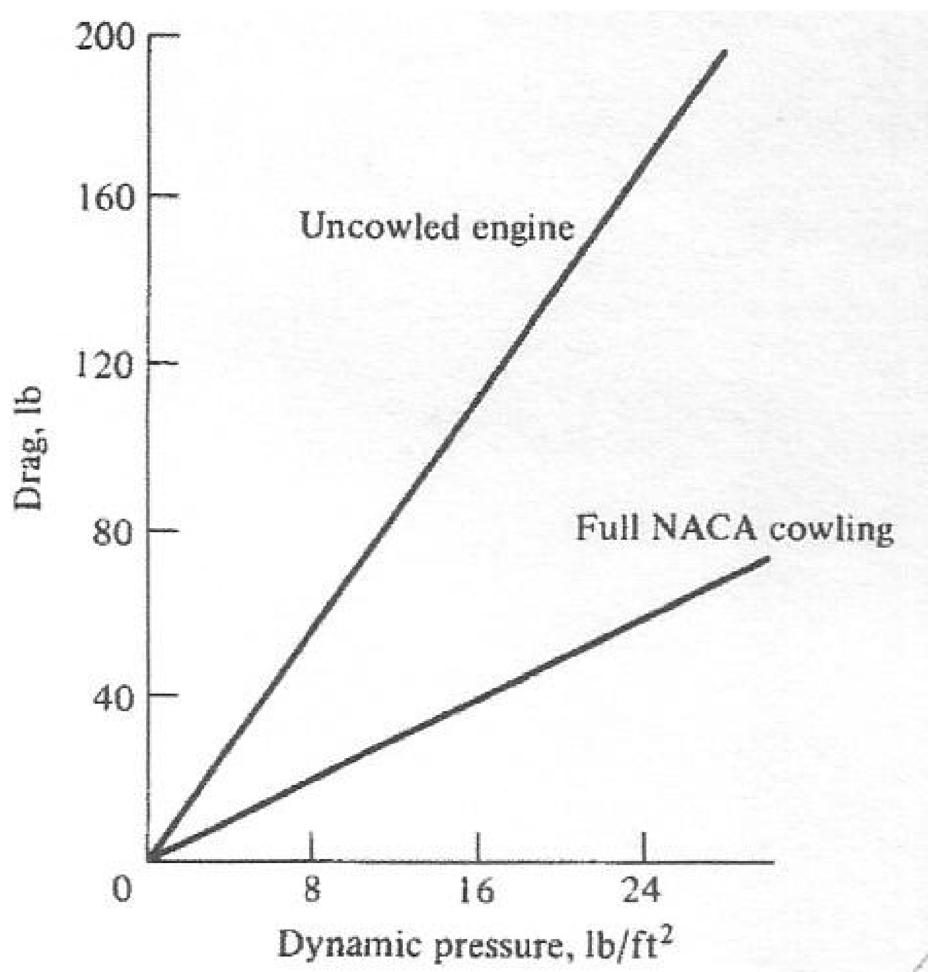


Figure 3.7 Reduction t» drag due to a full cowling.

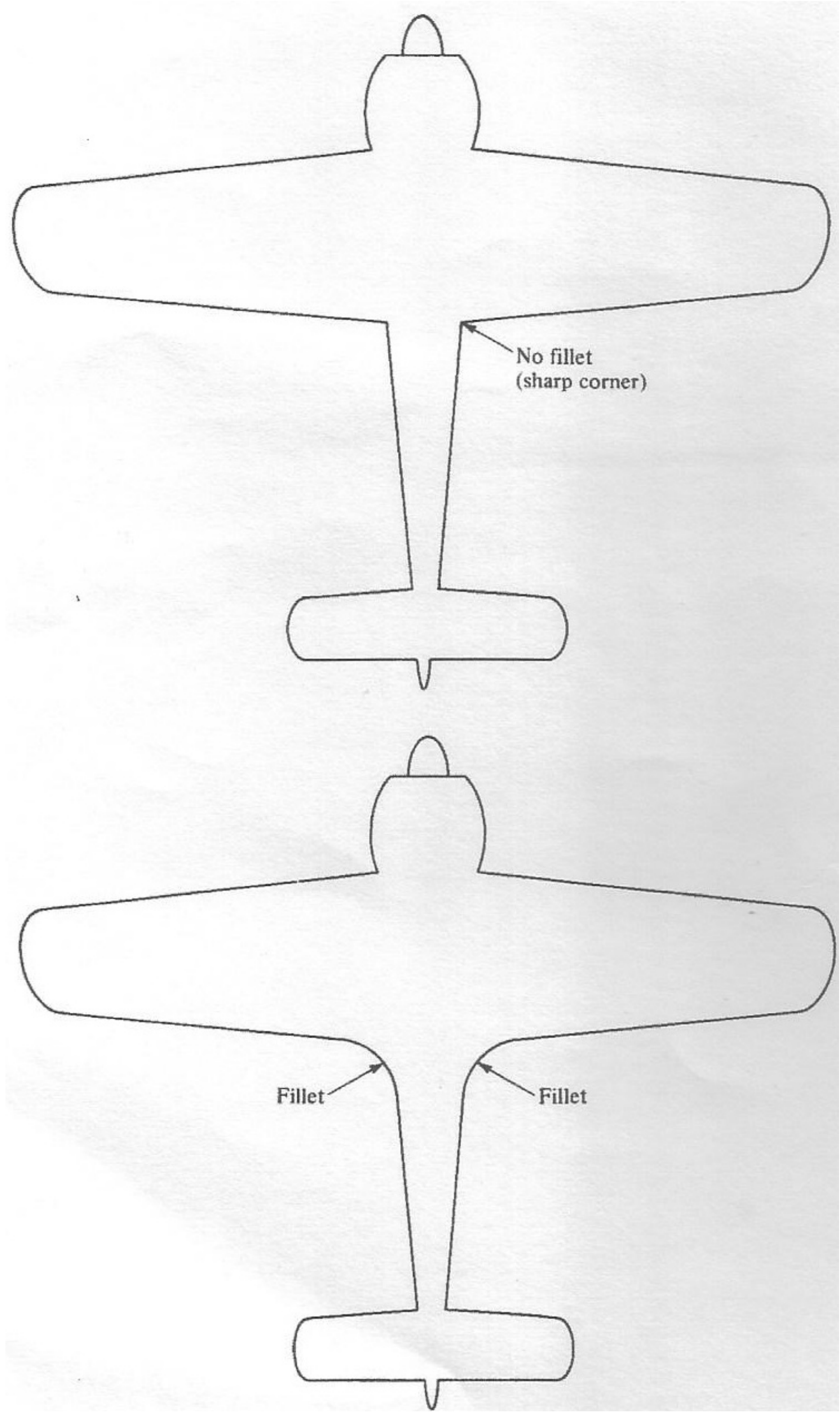


Figure 3.58 Illustration of the wing fillet

Subsonic wind tunnel funded by a grant from the Guggenheim Foundation. The first major experimental program in this tunnel was a commercial project for the Douglas Aircraft Company. Douglas was designing the DC-1, the forerunner of a series of highly successful transports (including the famous DC-3, which revolutionized commercial aviation in the 1930s). The DC-1 was plagued by unusual buffeting in the region where the wing joined the fuselage. The sharp corner at the juncture caused severe flow field separation, which resulted in high drag as well as shed vortices which buffeted the tail. The Caltech solution, which was new and pioneering, was to fair the trailing edge of the wing smoothly into the fuselage.

These fairings, called fillets, were empirically designed and were modeled in clay on the DC-1 wind-tunnel models. The best shape was found by trial and error. The addition of a fillet solved the buffeting problem by smoothing out the separated flow and hence also reduced the interference drag. Since that time, fillets have become a standard airplane design feature. Moreover, the fillet is an excellent example of how university laboratory research in the 1930s contributed directly to the advancement of practical airplane design.

CHAPTER FOUR

RESULTS AND DISCUSSION

4.1-Results

4.1.1: view on general CFD main steps

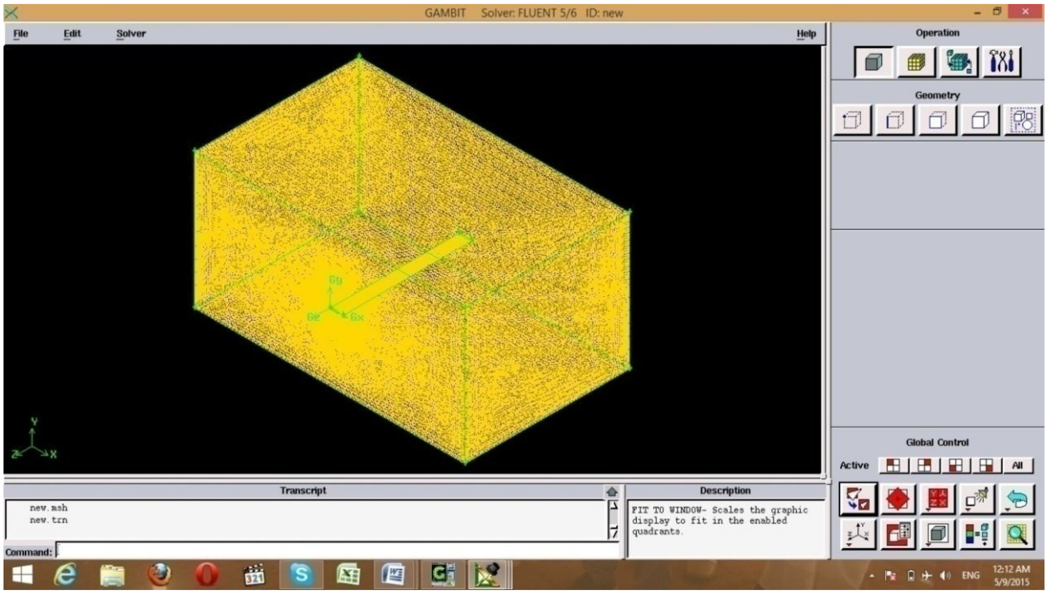


Fig: 4.1: The wing and wind wind-tunnel final view in GAMBIT

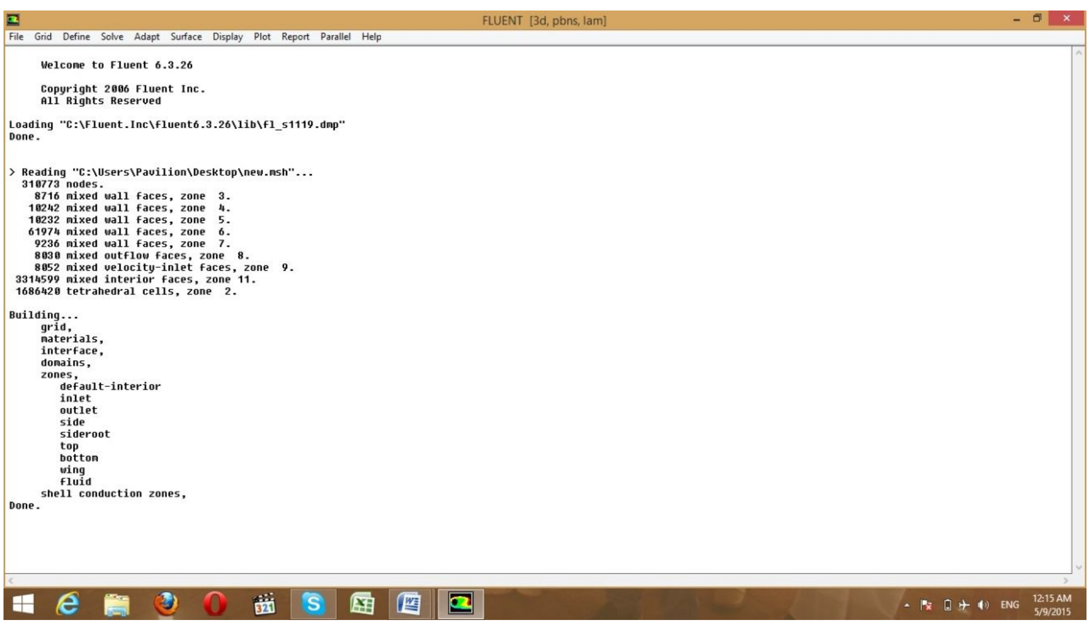


Fig: 4.2: The case reading in FLUENT

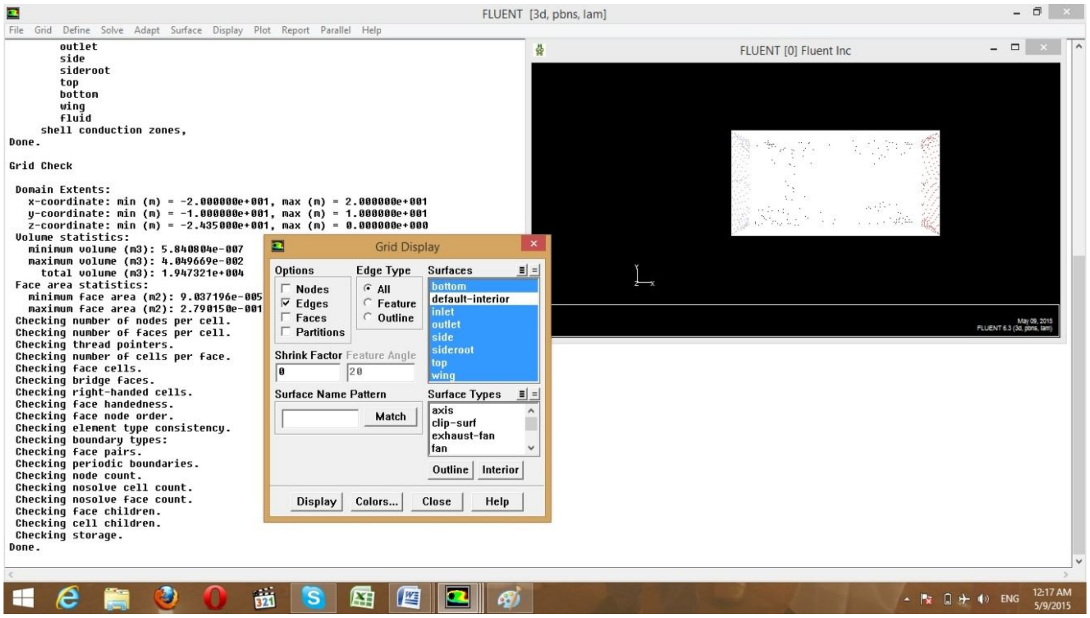
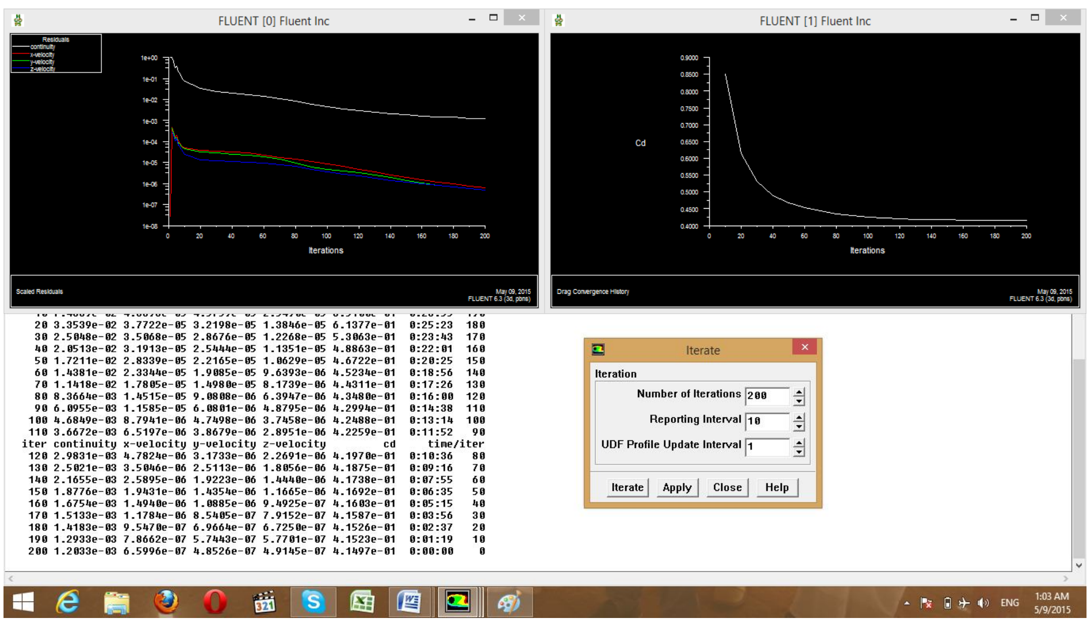


Fig: 4.3: The grid display in FLUENT



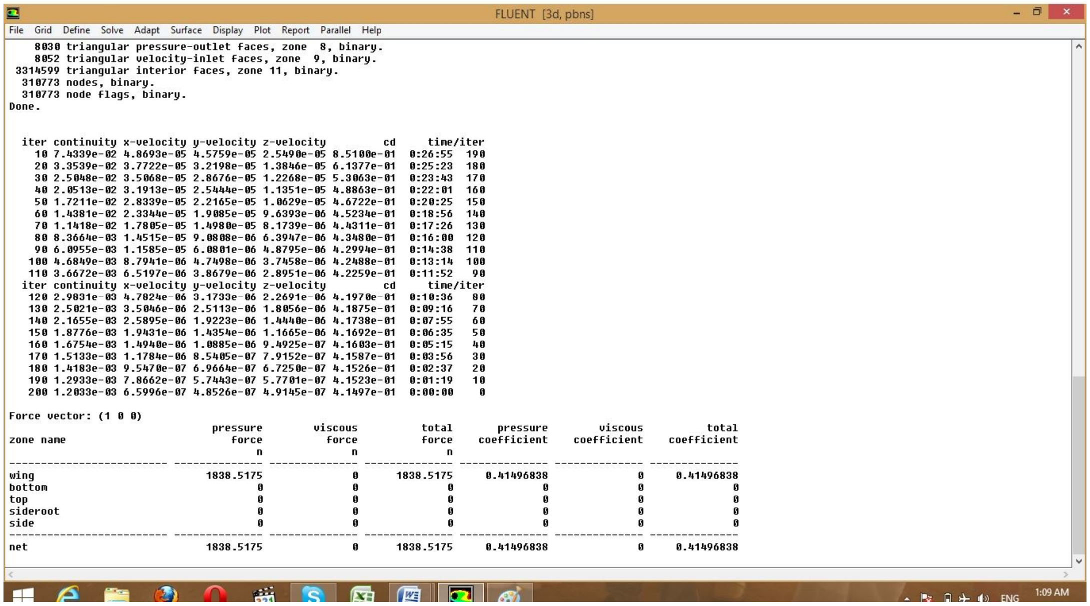


Fig: 4.5: The final forces results in FLUENT

4.1.2: Results Obtained With EXCEL

No.	H (m)	T (K)	ρ (Kg/M³)	M∞	a (m/s)	V∞ (m/s)	CL	CD□	CD	TR (N)	PR (W)	PA (W)
1	0.0	288.150	1.225	0.200	340.314	68.063	0.648	0.012	0.026	2530.516	172234.244	520000.000
2	0.0	288.150	1.225	0.250	340.314	85.079	0.415	0.012	0.018	2692.345	229060.938	520000.000
3	0.0	288.150	1.225	0.300	340.314	102.094	0.288	0.012	0.015	3248.075	331610.031	520000.000
4	0.0	288.150	1.225	0.350	340.314	119.110	0.212	0.013	0.014	4274.908	509184.516	520000.000
5	0.0	288.150	1.225	0.400	340.314	136.126	0.162	0.013	0.014	5371.461	731194.316	520000.000
6	0.0	288.150	1.225	0.450	340.314	153.141	0.128	0.013	0.014	6703.191	1026536.707	520000.000
7	0.0	288.150	1.225	0.500	340.314	170.157	0.104	0.013	0.014	8299.319	1412188.967	520000.000
8	0.0	288.150	1.225	0.550	340.314	187.173	0.086	0.014	0.014	10265.420	1921408.780	520000.000
9	0.0	288.150	1.225	0.600	340.314	204.189	0.072	0.014	0.014	12344.845	2520677.253	520000.000
10	0.0	288.150	1.225	0.650	340.314	221.204	0.061	0.014	0.014	14664.004	3243742.019	520000.000
11	0.0	288.150	1.225	0.700	340.314	238.220	0.053	0.015	0.015	17430.910	4152393.073	520000.000
12	0.0	288.150	1.225	0.750	340.314	255.236	0.046	0.015	0.015	20374.008	5200176.817	520000.000
13	0.0	288.150	1.225	0.800	340.314	272.252	0.040	0.015	0.015	23357.287	6359057.452	520000.000

No.	ΔP (w)	RC m/s	V_Tr.min (m/s)	at (m/s²)	1/at (s²/m)	R (N)	Dt (m/s²)	1/Dt (s²/m)	Da (m/s²)	R2 (N)	D.total (m/s²)	1/D.total (s²/m)
1	347765.756	5.539	30.530	0.798	1.253	-2530.516	-0.395	-2.529	11627.833	-14158.349	-2.212	-0.452
2	290939.062	4.634	30.492	0.534	1.872	-2692.345	-0.421	-2.377	18168.488	-20860.833	-3.260	-0.307
3	188389.969	3.001	30.409	0.288	3.468	-3248.075	-0.508	-1.970	26162.623	-29410.698	-4.595	-0.218
4	10815.484	0.172	29.926	0.014	70.483	-4274.908	-0.668	-1.497	35610.237	-39885.145	-6.232	-0.160
5	-211194.316	-3.364	29.880	-0.242	-4.125	-5371.461	-0.839	-1.191	46511.330	-51882.792	-8.107	-0.123
6	-506536.707	-8.068	29.800	-0.517	-1.935	-6703.191	-1.047	-0.955	58865.902	-65569.094	-10.245	-0.098
7	-892188.967	-14.210	29.670	-0.819	-1.221	-8299.319	-1.297	-0.771	72673.954	-80973.272	-12.652	-0.079
8	-1401408.78	-22.321	29.441	-1.170	-0.855	-10265.42	-1.604	-0.623	87935.484	-98200.904	-15.344	-0.065
9	-2000677.25	-31.866	29.325	-1.531	-0.653	-12344.85	-1.929	-0.518	104650.493	-116995.34	-18.281	-0.055
10	-2723742.02	-43.383	29.211	-1.924	-0.520	-14664.00	-2.291	-0.436	122818.982	-137482.99	-21.482	-0.047
11	-3632393.08	-57.855	29.013	-2.383	-0.420	-17430.91	-2.724	-0.367	142440.949	-159871.86	-24.980	-0.040
12	-4680176.82	-74.544	28.871	-2.865	-0.349	-20374.01	-3.183	-0.314	163516.395	-183890.40	-28.733	-0.035
13	-5839057.45	-93.002	28.808	-3.351	-0.298	-23357.29	-3.650	-0.274	186045.321	-209402.60	-32.719	-0.031

Table: 4.1 performance characteristic at sea level

No.	H (m)	T (K)	ρ (Kg/M³)	M∞	a (m/s)	V∞ (m/s)	CL	CD□	CD	TR (N)	PR (W)	PA (W)
1	3098.0	268.020	0.900	0.200	328.212	65.642	0.948	0.009	0.039	2611.687	171437.505	400000.000
2	3098.0	268.020	0.900	0.250	328.212	82.053	0.607	0.009	0.021	2221.214	182257.390	400000.000
3	3098.0	268.020	0.900	0.300	328.212	98.464	0.421	0.009	0.015	2236.713	220234.944	400000.000
4	3098.0	268.020	0.900	0.350	328.212	114.874	0.310	0.009	0.012	2483.210	285256.862	400000.000
5	3098.0	268.020	0.900	0.400	328.212	131.285	0.237	0.009	0.011	2887.615	379100.126	400000.000
6	3098.0	268.020	0.900	0.450	328.212	147.695	0.187	0.010	0.011	3683.176	543988.330	400000.000
7	3098.0	268.020	0.900	0.500	328.212	164.106	0.152	0.010	0.011	4378.077	718469.074	400000.000
8	3098.0	268.020	0.900	0.550	328.212	180.517	0.125	0.010	0.010	5173.769	933951.536	400000.000
9	3098.0	268.020	0.900	0.600	328.212	196.927	0.105	0.010	0.010	6063.972	1194161.647	400000.000
10	3098.0	268.020	0.900	0.650	328.212	213.338	0.090	0.010	0.010	7044.729	1502907.593	400000.000
11	3098.0	268.020	0.900	0.700	328.212	229.749	0.077	0.011	0.011	9086.772	2087672.234	400000.000
12	3098.0	268.020	0.900	0.750	328.212	246.159	0.067	0.011	0.011	10385.687	2556531.510	400000.000
13	3098.0	268.020	0.900	0.800	328.212	262.570	0.059	0.011	0.011	11779.496	3092939.064	400000.000

No.	ΔP (w)	RC m/s	V_Tr.min (m/s)	at (m/s^2)	1/at (s^2/m)	R (N)	Dt (m/s^2)	1/Dt (s^2/m)	Da (m/s^2)	R2 (N)	D.total (m/s^2)	1/D.total (s^2/m)
1	228562.495	3.640	38.196	0.544	1.838	-2611.687	-0.408	-2.451	7945.566	-10557.253	-1.650	-0.606
2	217742.610	3.468	38.196	0.415	2.412	-2221.214	-0.347	-2.881	12414.946	-14636.161	-2.287	-0.437
3	179765.056	2.863	38.196	0.285	3.506	-2236.713	-0.349	-2.861	17877.523	-20114.236	-3.143	-0.318
4	114743.138	1.828	38.196	0.156	6.407	-2483.210	-0.388	-2.577	24333.295	-26816.504	-4.190	-0.239
5	20899.874	0.333	38.196	0.025	40.202	-2887.615	-0.451	-2.216	31782.262	-34669.877	-5.417	-0.185
6	-143988.330	-2.293	37.391	-0.152	-6.565	-3683.176	-0.575	-1.738	40224.426	-43907.601	-6.861	-0.146
7	-318469.074	-5.072	37.391	-0.303	-3.298	-4378.077	-0.684	-1.462	49659.785	-54037.862	-8.443	-0.118
8	-533951.536	-8.505	37.391	-0.462	-2.164	-5173.769	-0.808	-1.237	60088.340	-65262.108	-10.197	-0.098
9	-794161.647	-12.649	37.391	-0.630	-1.587	-6063.972	-0.947	-1.055	71510.090	-77574.062	-12.121	-0.083
10	-1102907.6	-17.567	37.391	-0.808	-1.238	-7044.729	-1.101	-0.908	83925.036	-90969.765	-14.214	-0.070
11	-1687672.23	-26.881	36.327	-1.148	-0.871	-9086.772	-1.420	-0.704	97333.178	-106419.95	-16.628	-0.060
12	-2156531.51	-34.348	36.327	-1.369	-0.731	-10385.69	-1.623	-0.616	111734.516	-122120.20	-19.081	-0.052
13	-2692939.06	-42.892	36.327	-1.603	-0.624	-11779.496	-1.841	-0.543	127129.049	-138908.55	-21.704	-0.046

Table: 4.2 performance characteristics at 3km

No.	H (m)	T (K)	ρ (Kg/M^3)	M∞	a (m/s)	V∞ (m/s)	CL	CD□	CD	TR (N)	PR (W)	PA (W)
1	6094.0	248.550	0.653	0.200	316.066	63.213	1.410	0.008	0.075	3352.797	211941.084	300000.000
2	6094.0	248.550	0.653	0.250	316.066	79.017	0.902	0.008	0.036	2474.516	195527.626	300000.000
3	6094.0	248.550	0.653	0.300	316.066	94.820	0.626	0.008	0.021	2133.527	202300.718	300000.000
4	6094.0	248.550	0.653	0.350	316.066	110.623	0.460	0.008	0.015	2069.732	228960.254	300000.000
5	6094.0	248.550	0.653	0.400	316.066	126.426	0.352	0.009	0.013	2352.653	297437.604	300000.000
6	6094.0	248.550	0.653	0.450	316.066	142.230	0.278	0.009	0.012	2621.372	372837.121	300000.000
7	6094.0	248.550	0.653	0.500	316.066	158.033	0.226	0.009	0.011	2984.964	471722.918	300000.000
8	6094.0	248.550	0.653	0.550	316.066	173.836	0.186	0.009	0.010	3427.918	595896.748	300000.000
9	6094.0	248.550	0.653	0.600	316.066	189.640	0.157	0.010	0.011	4341.789	823375.378	300000.000
10	6094.0	248.550	0.653	0.650	316.066	205.443	0.133	0.010	0.011	4988.518	1024856.005	300000.000
11	6094.0	248.550	0.653	0.700	316.066	221.246	0.115	0.010	0.010	5701.098	1261346.709	300000.000
12	6094.0	248.550	0.653	0.750	316.066	237.050	0.100	0.010	0.010	6476.908	1535348.435	300000.000
13	6094.0	248.550	0.653	0.800	316.066	252.853	0.088	0.010	0.010	7314.122	1849396.821	300000.000

No.	ΔP (w)	RC m/s	V_Tr.min (m/s)	at (m/s^2)	1/at (s^2/m)	R (N)	Dt (m/s^2)	1/Dt (s^2/m)	Da (m/s^2)	R2 (N)	D.total (m/s^2)	1/D.total (s^2/m)
1	88058.916	1.403	46.186	0.218	4.594	-3352.797	-0.524	-1.909	5345.126	-8697.922	-1.359	-0.736
2	104472.374	1.664	46.186	0.207	4.841	-2474.516	-0.387	-2.586	8351.759	-10826.275	-1.692	-0.591
3	97699.282	1.556	46.186	0.161	6.211	-2133.527	-0.333	-3.000	12026.533	-14160.060	-2.213	-0.452
4	71039.746	1.131	46.186	0.100	9.966	-2069.732	-0.323	-3.092	16369.448	-18439.180	-2.881	-0.347
5	2562.396	0.041	44.846	0.003	315.771	-2352.653	-0.368	-2.720	21380.503	-23733.156	-3.708	-0.270
6	-72837.121	-1.160	44.846	-0.080	-12.497	-2621.372	-0.410	-2.441	27059.699	-29681.071	-4.638	-0.216
7	-171722.918	-2.735	44.846	-0.170	-5.890	-2984.964	-0.466	-2.144	33407.036	-36392.000	-5.686	-0.176
8	-295896.748	-4.713	44.846	-0.266	-3.760	-3427.918	-0.536	-1.867	40422.514	-43850.431	-6.852	-0.146
9	-523375.378	-8.336	43.680	-0.431	-2.319	-4341.789	-0.678	-1.474	48106.132	-52447.921	-8.195	-0.122
10	-724856.005	-11.545	43.680	-0.551	-1.814	-4988.518	-0.779	-1.283	56457.891	-61446.409	-9.601	-0.104
11	-961346.709	-15.312	43.680	-0.679	-1.473	-5701.098	-0.891	-1.123	65477.791	-71178.889	-11.122	-0.090
12	-1235348.435	-19.676	43.680	-0.814	-1.228	-6476.908	-1.012	-0.988	75165.831	-81642.739	-12.757	-0.078
13	-1549396.821	-24.678	43.680	-0.957	-1.044	-7314.122	-1.143	-0.875	85522.012	-92836.134	-14.506	-0.069

Table: 4.3 performance characteristics at 6km

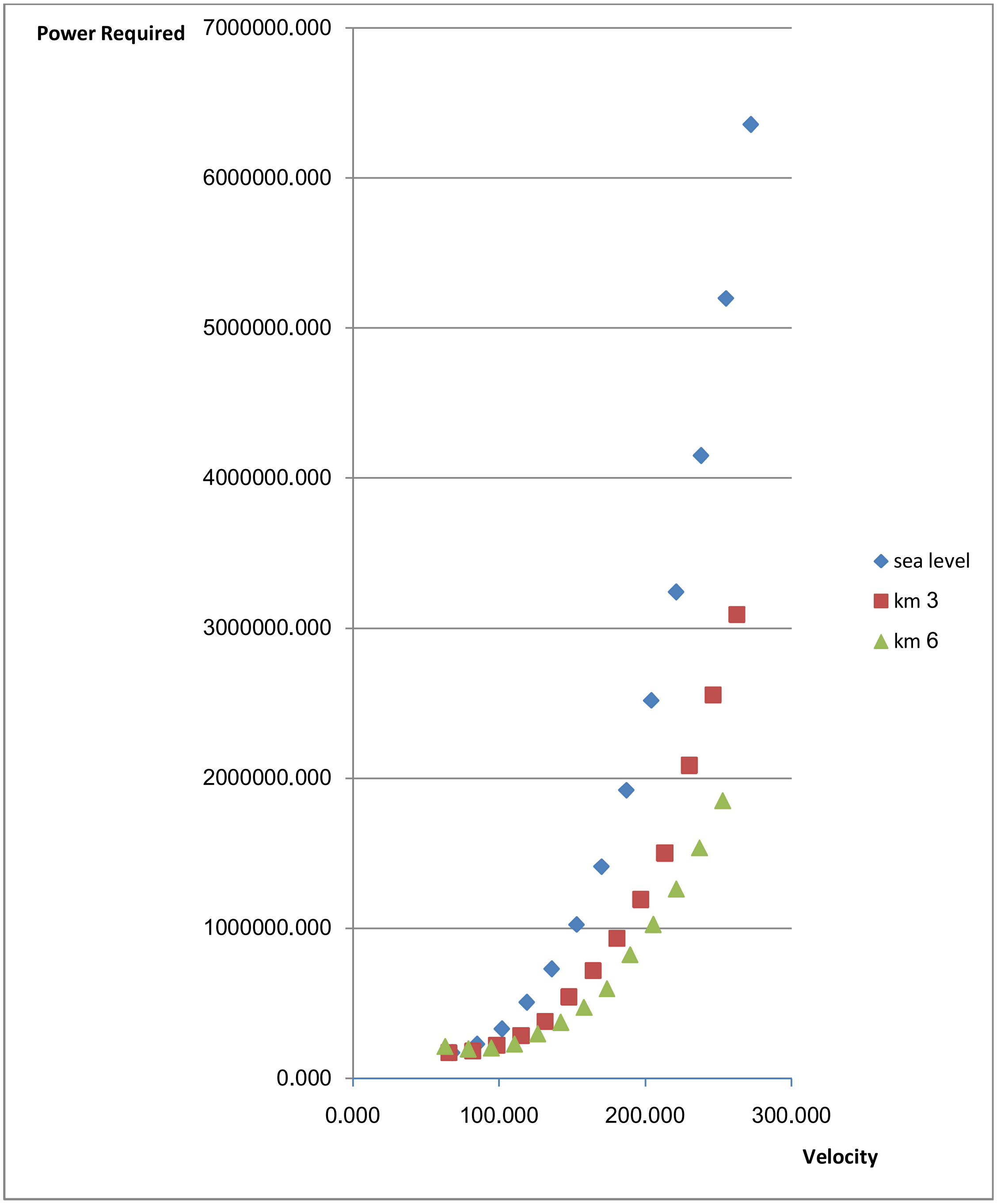


Fig: 4.6: Velocity versus Power Required

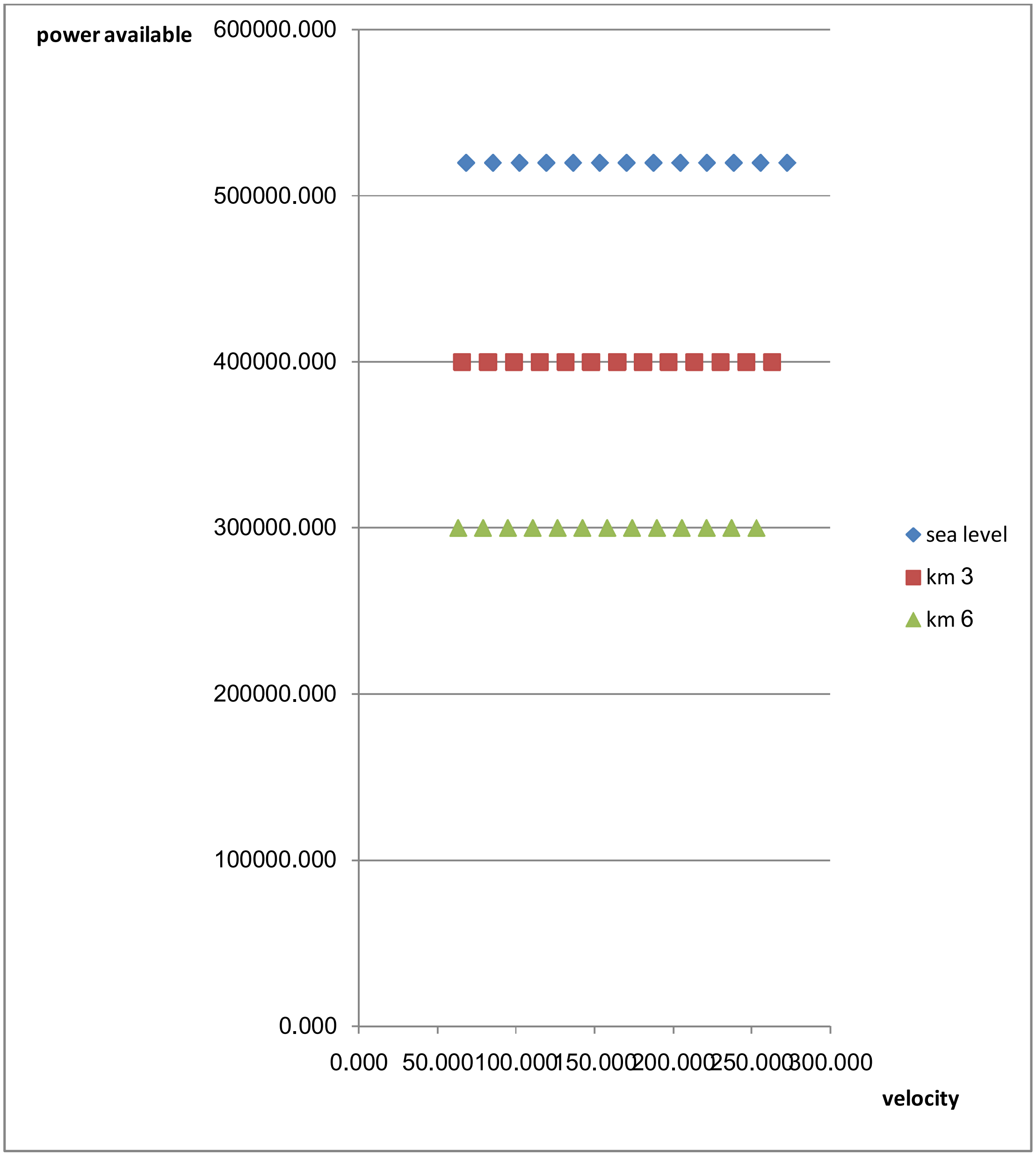


Fig: 4.7: Velocity Versus Power Available

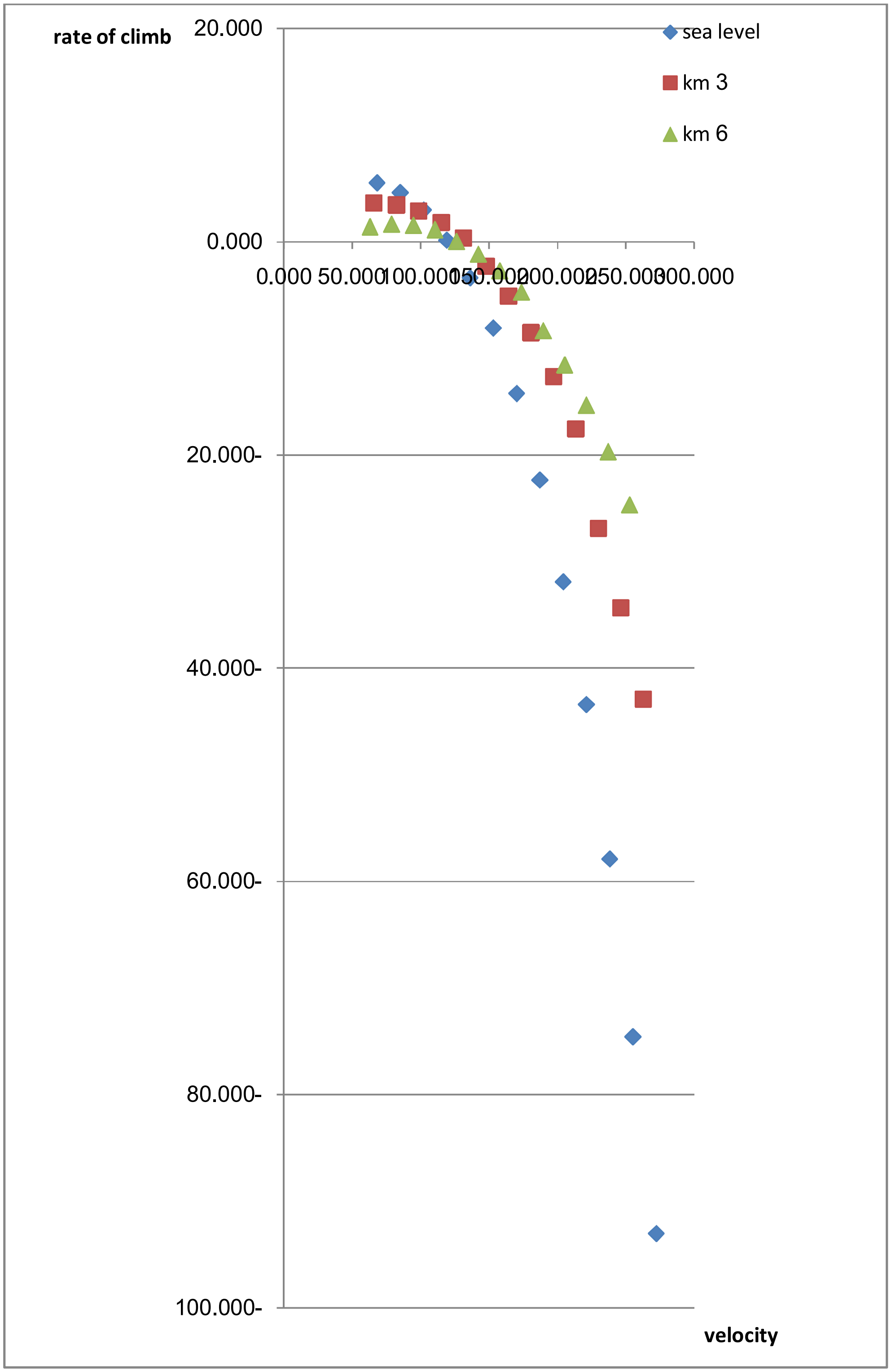


Fig: 4.8: Velocity Versus Rate of climb

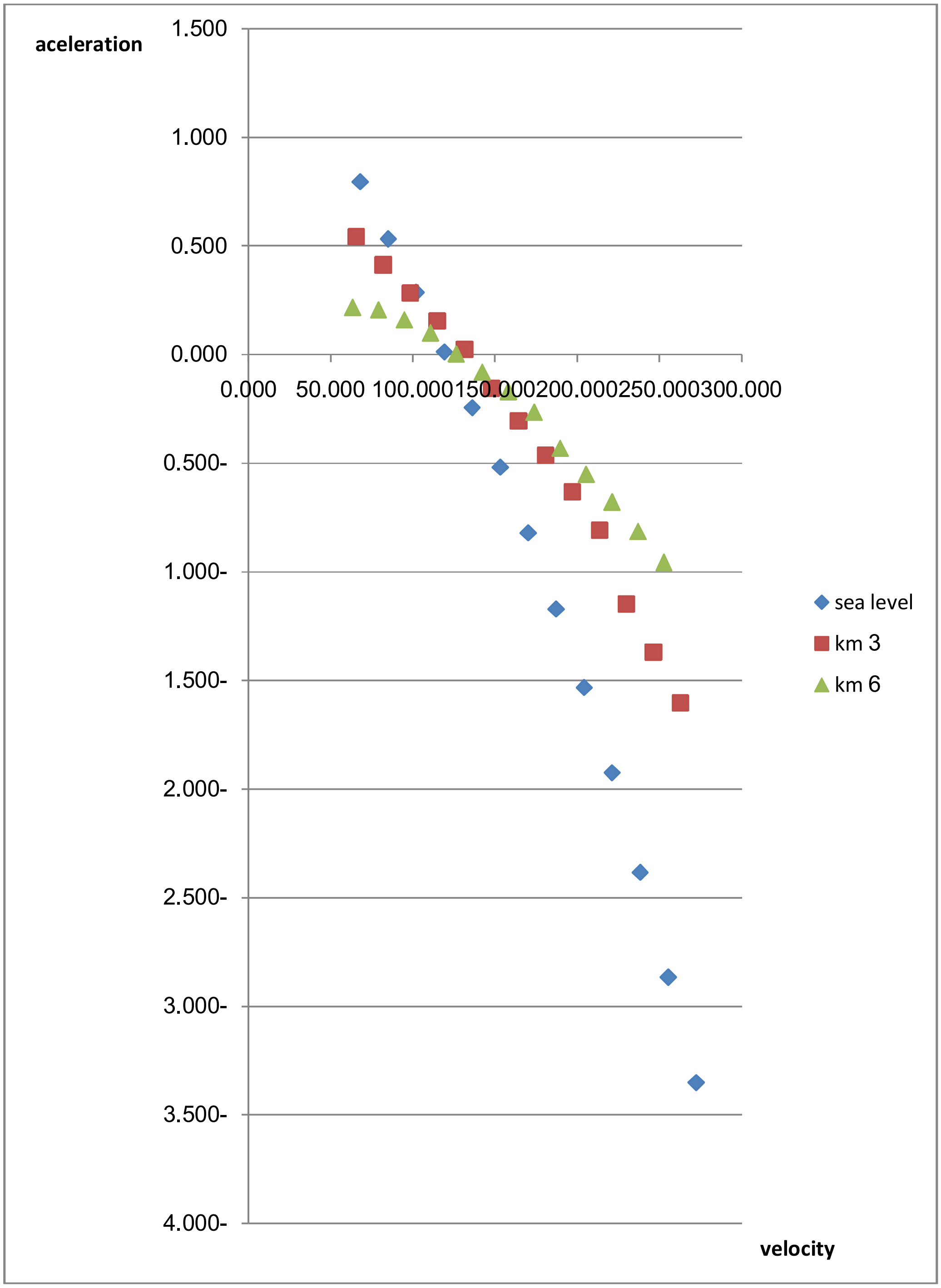


Fig: 4.9: Velocity Versus Acceleration

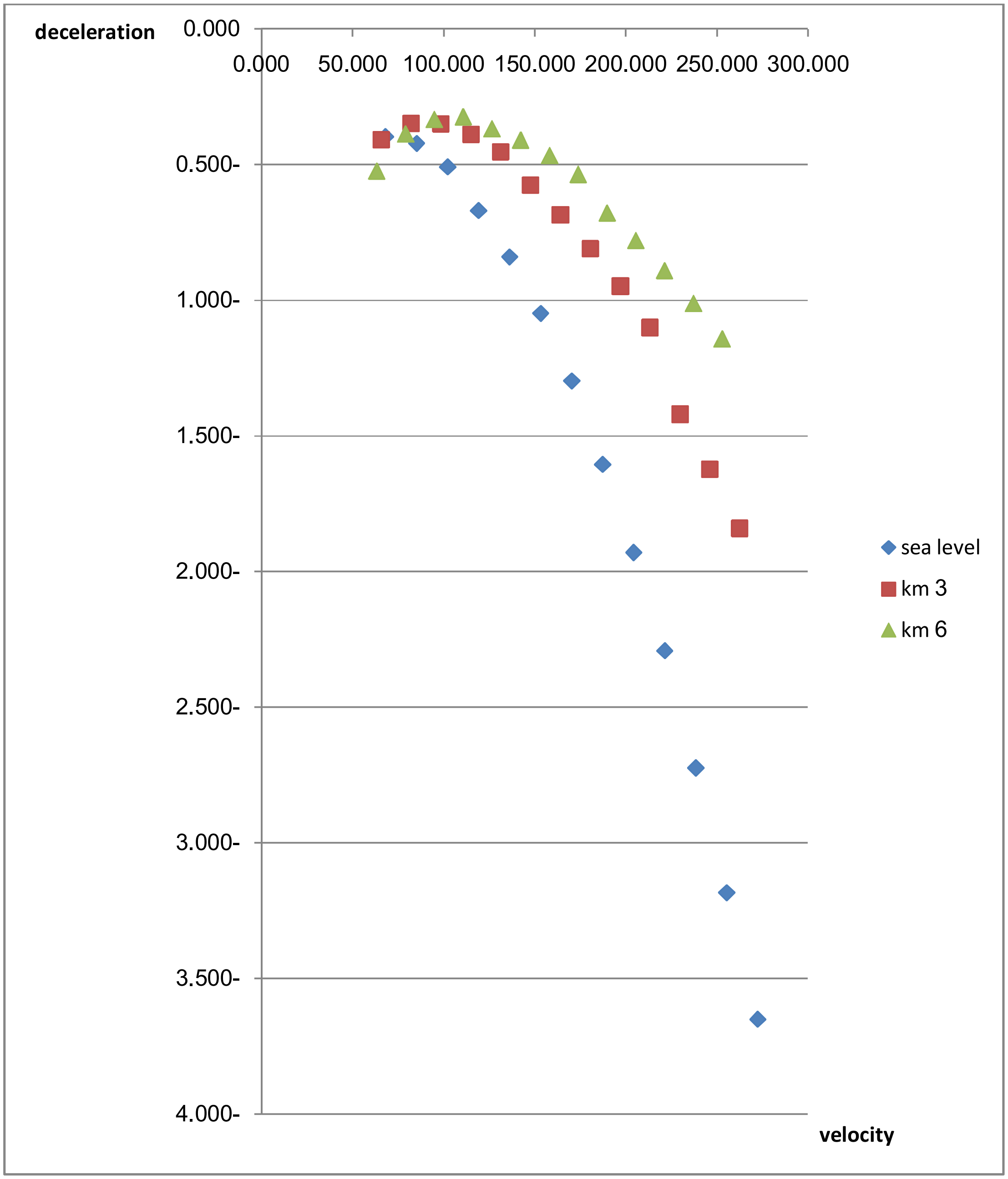


Fig: 4.10: Velocity Versus Deceleration without Aerodynamic Brake

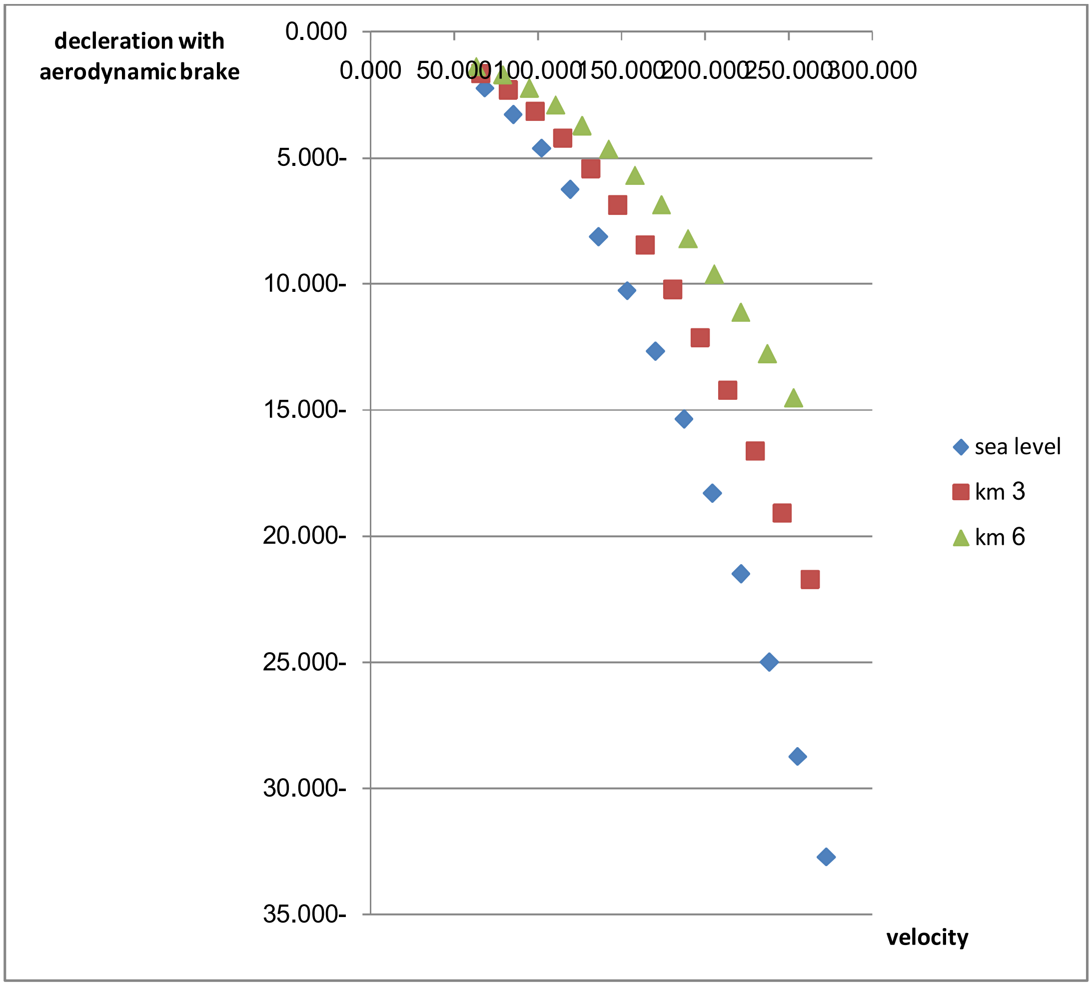


Fig: 4.11: Velocity Versus Deceleration with Aerodynamic Brake

4.1.3: Results obtained with MATLAB

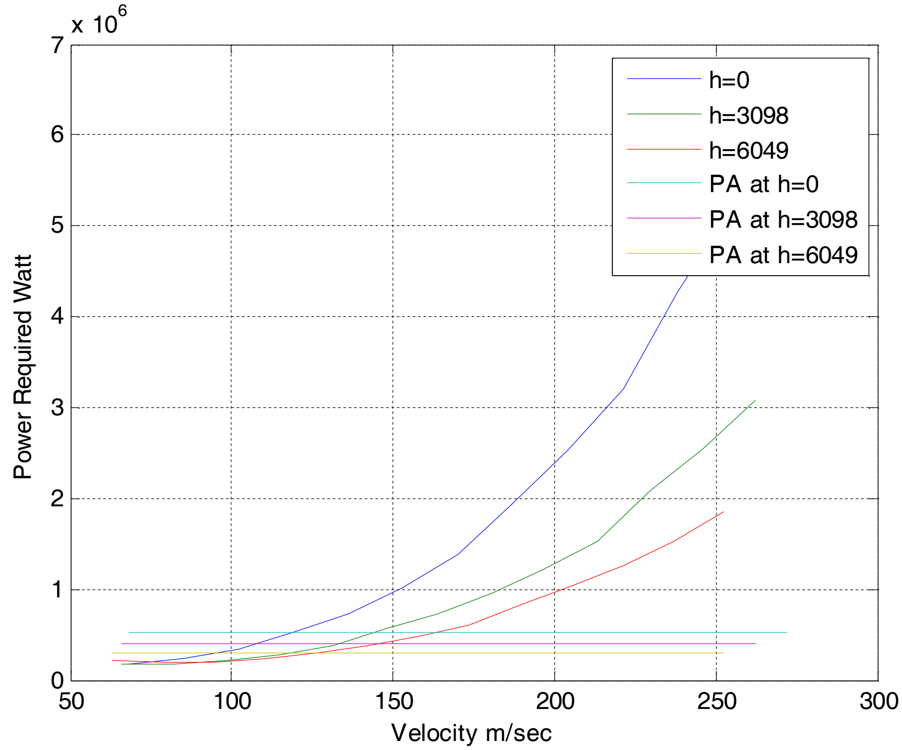


Fig: 4.12: velocity versus power required- power available

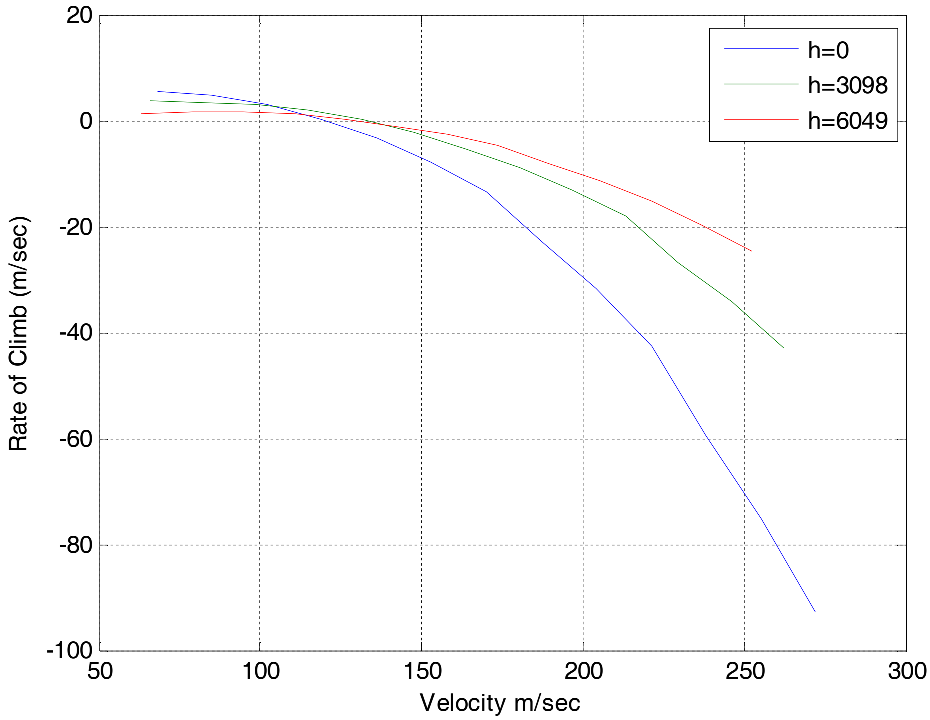


Fig: 4.13: velocity versus rate of climb

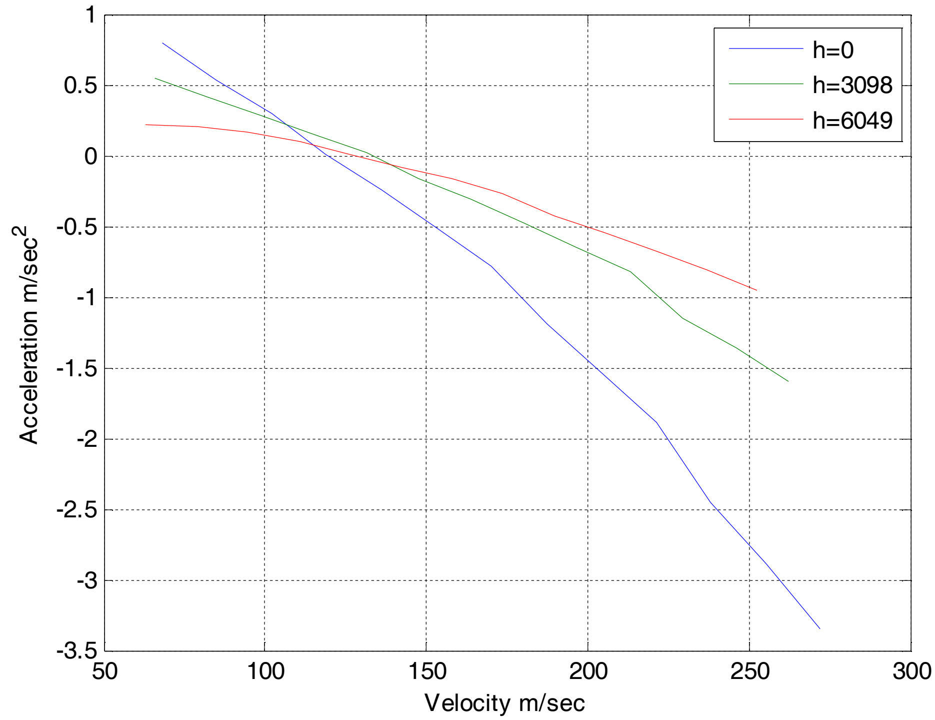


Fig: 4.14: velocity versus acceleration

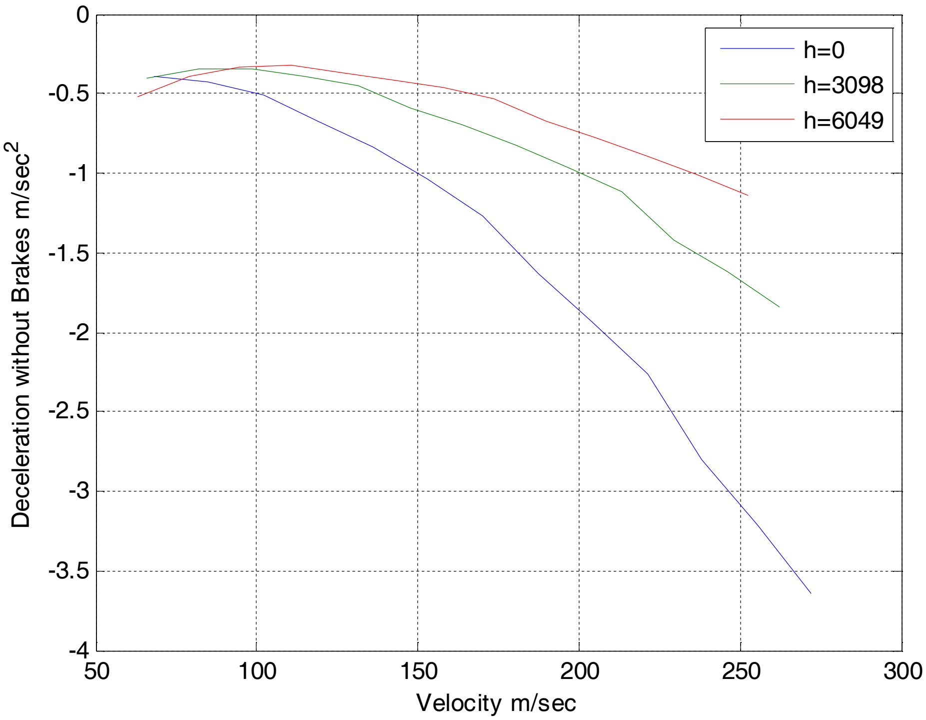


Fig: 4.15: velocity versus deceleration without aerodynamic brake

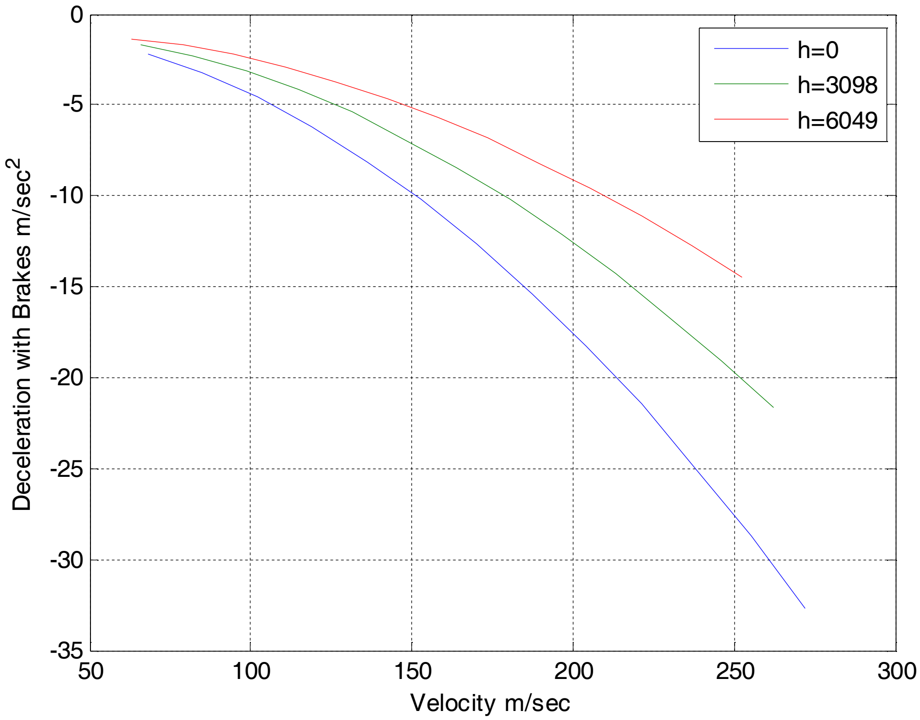


Fig: 4.16: deceleration with aerodynamic brake

3.13: Discussion

For every aerodynamic body there is a relation between C_D and C_L that can be expressed as an equation or plotted on a graph. Both the equation and the graph are called the drag polar. Virtually the aerodynamic information about an airplane necessary for a performance analysis is wrapped up in the drag polar. We have examined this matter and construct a suitable expression for the profile drag polar for the L-410turbolet airplane. The profile drag is a combination of two types of drag: skin-friction drag and pressure drag due to flow separation.

Skin-friction drag: is due to the frictional shear stress acting on the surface of the wing. Pressure drag due to flow separation is caused by the imbalance of the pressure distribution in the drag direction when the boundary layer separates from the wing surface. The recommended formula developed by White and Christophe in which the skin-friction drag is more easily calculated in an explicit manner from:

$$c_f = \frac{0.42}{\ln^2(0.056\,\mathrm{Re})} \qquad (3.49)$$

$$\mathrm{Re}_{\mathrm{trans}} = \frac{\rho_\infty V_\infty x_{\mathrm{tr}}}{\mu_\infty} \qquad (3.50)$$

We can recognize that the zero lift drag increases as the velocity increases; and this increment is due to the increment of both Reynolds number, and the adverse pressure gradient.

In Airplane performance and design, drag is perhaps the most important aerodynamic quantity. It is also important to recognize that the analytical predication of drag is much harder and more tenuous than that of lift. Closed form analytical expressions for drag exist only in special cases. How ever faced with this situations people responsible for airplane design and analyses have assimilated many empirical data on drag, and have synthesized various methodologies for drag predication.

Hence in the previous section we have provide a graphical and an analytical formulas for only aspects of drag for the L-410turbolet aircraft, and hopefully we give some idea of what can be done by predict drag for purposes of preliminary performance analyses and conceptual design of airplane.

The induced flow effects due to wing-tip vortices result in an extra component of drag on three dimensional lifting body. This extra drag is called ***induced drag***. Induced drag is purely a pressure drag caused by the wing-tip vortices which generate an induced perturbing flow field over the wing which in turn perturbs the pressure distribution over the wing surface in such a way the integrated pressure distribution yields an increase in drag. The induced drag for a high-aspect-ratio straight wing, prandtl's lifting line theory shows that the induced drag coefficient defined by:

$$C_{D_i} = \frac{C_L^2}{\pi e \mathrm{AR}} \qquad (3.51)$$

Where "e" is the span efficiency factor given by:

$$e = \frac{1}{1+\delta} \qquad (3.52)$$

"□" is calculated from lifting line theory. It is a function of aspect ratio and taper ratio.

For the subsonic transport the elements labeled wing, body, empennage, engine installations, interference, leaks, under carriage, and flaps are the contributors to the zero-lift parasite drag, that is, they stem from friction drag and pressure drag (due to flow separation). The element labeled lift-dependent drag (drag due to lift) stems from the increment of parasite drag associated with the change of angle of attack from the zero lift value, and the induced drag. Note that the most of the drag at cruise is parasite drag, where as the most of drag at takeoff is lift-dependent drag, which in this case is mostly induced drag, associated with high lift coefficient at takeoff.

Propellers are driven by reciprocating piston engines or by gas turbine (turboprop). The engines in both these cases are rated in terms of power not thrust.

We can clearly from figure (4.6) see that the power required increase as the free stream velocity increase (at constant altitude) and this for two reasons; first in this region of flight velocity (region of velocity stability) , the zero-lift drag increases as the square of free stream velocity. At high velocity the total drag is dominated by the zero-lift drag. Hence, as the velocity of the airplane increases, and the thrust required increase. Also the power required is the product of thrust required and free stream velocity which will result in further increase of power required as the free stream velocity increase. From other point of view we find that the power required at constant velocity decrease with increase of altitude and this because the decrease in density with increasing altitude which we lead to a decrease in the value of drag and hence the thrust required which in turn will decrease the power required value.

We can note from figure (4.7) that the power available (at constant altitude) is constant with respect to the free stream velocity, and this is understandable when we know that the power available for propeller driven aircraft is the product of the shaft power (which is a constant value with respect to the flight velocity) and the propeller efficiency (which changes by a neglect able value with the flight velocity), and this can make us understand why propeller driven aircrafts are expressed in term of power instead of thrust terms (as in turbojet and turbofan aircrafts).Also we can note that the power available decrease with increasing altitude and this occurs as the efficiency of the fuel combustion process decrease as the air density decrease with increasing altitude.

From figure (4.8) it is clear that both the net power and the rate of climb (at constant altitude) decrease with increasing the free stream velocity and this can be simply understood because the power required increase while the power available is constant. And physically this is because the propeller encounters problems with increasing velocity which may include tip separation or at more high speeds shock wave formation at the tip of propeller blades.

And from the same reasons discussed in the previous paragraph we can see from figure (4.9) that the acceleration of the aircraft decrease with increasing free stream velocity (at constant altitude) and vice versa. And at increasing the altitude the (at constant velocity) the acceleration will decrease.

In contrast we find from fig (4.10) and fig (4.11) as the free stream velocity (at constant altitude) the value of both deceleration without aerodynamics brake and deceleration with aerodynamic brake will increase, but the deceleration with aerodynamic brake will increase more than the deceleration without aerodynamic brake because it is composed of two components; the drag (thrust required), and the aerodynamic brake and both components increase with increasing the flight velocity. When the altitude increase (for a given flight velocity) both deceleration with aerodynamic brake and without aerodynamic brake will decrease as a result to the decreased air density.

4.3 comparisons between L-410turbolet and ANTONOV an-2 in main performance parameters

Performance parameter	ANTONOV an-2	L-410turbolet
Max speed	258 km/hr	388 km/hr
Cruising speed	190 km/hr	365 km/hr
Service ceiling	14750 ft	24300 ft
Initial rate of climb	700 ft/min	1378 ft/min
Empty weight	3300 kg	3925 kg
Maximum take-off weight	5500 kg	6400 kg

Table 4.4

CHAPTER FIVE

CONCLUSION AND RECOMMENDATIONS

5.1 Conclusion

The drag polar of L-410turbolet has been estimated using computational fluid dynamics techniques (CFD), then the performance characteristics of the aircraft have been calculated at different speeds of flight and different altitudes using EXCEL software, and their variation has been plotted in curves. Also a MATLAB code has been made in order to ease the usage of these calculations for any other air craft by modifying the inputted data simply. Finally a comparison has been made between the main performance parameters of L-410turbolet and ANTONOV an-2 to show the development after modification.

5.2 Recommendations

- To study the stability and control characteristics of L-410turbolet and to design a program to compute these characteristics at various conditions.
- To study the advantages and disadvantages of employing an auto throttle system or a fuel control system in order to control the acceleration and deceleration process of the L-410turbolet according to the computed variation.
- To study the effects of re designing the L-410turbolet wings' with a fillet in order to improve the aerodynamic efficiency and the resulted improvement in the mechanics of flight characteristics.
- To study the performance characteristics of the L-410turbolet using the analytical approach and then using the graphical approach and to compare between the results obtained by two methods.

References:

- John D.Anderson, Jr, "INTRODUCTION TO FLIGHT", McGraw-Hill Book Company, 1989.
- John D.Anderson, Jr, "AIRCRAFT PERFORMANCE AND DESIGN", McGraw-Hill Book Company, 1999.
- Bandu N. Pamadi, "PERFORMANCE, STABILITY, DYNAMICS AND CONTROL OF AIRPLANE", American Institute of Aeronautics and Astronautics, Reston, Virginia, 1998.

Appendix

Wing's vertex data:

Tip airfoil:

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(Stations and ordinates given

in per cent of airfoil chord)

Upper Surface		Lower Surface	
Station	Ordinate	Station	Ordinate
0.0000	0.0000	0.0000	0.0000
0.5000	1.2675	0.5000	-0.7126
0.7500	1.4974	0.7500	-0.8852
1.2500	1.8861	1.2500	-1.1361
2.5000	2.6280	2.5000	-1.5647
5.0000	3.7194	5.0000	-2.1048
7.5000	4.5629	7.5000	-2.4761
10.0000	5.2586	10.0000	-2.7576
15.0000	6.3594	15.0000	-3.1571
20.0000	7.1819	20.0000	-3.4100
25.0000	7.7886	25.0000	-3.5529
30.0000	8.2104	30.0000	-3.6027
35.0000	8.4477	35.0000	-3.5504
40.0000	8.5152	40.0000	-3.4042
45.0000	8.4194	45.0000	-3.1679
50.0000	8.1775	50.0000	-2.8591
55.0000	7.8036	55.0000	-2.4943
60.0000	7.3104	60.0000	-2.0912
65.0000	6.7074	65.0000	-1.6673
70.0000	6.0031	70.0000	-1.2434

75.0000	5.2068	75.0000	-0.8511
80.0000	4.3221	80.0000	-0.5417
85.0000	3.2964	85.0000	-0.3697
90.0000	2.2249	90.0000	-0.2345
95.0000	1.1248	95.0000	-0.1291
100.0000	-0.0000	100.0000	0.0000

L.E. radius = 0.990 percent c

slope of mean line at LE = 0.1902

NACA 63A412

(Stations and ordinates given

in per cent of airfoil chord)

Upper Surface		Lower Surface	
Station	Ordinate	Station	Ordinate
0.0000	0.0000	0.0000	0.0000
0.3184	1.0676	0.6816	-0.8427
0.5467	1.3129	0.9533	-0.9966
1.0170	1.7148	1.4830	-1.2323
2.2248	2.4818	2.7752	-1.6380
4.6874	3.6000	5.3126	-2.1576
7.1737	4.4625	7.8263	-2.5172
9.6716	5.1738	10.3284	-2.7897
14.6864	6.2995	15.3136	-3.1767
19.7158	7.1413	20.2842	-3.4209
24.7538	7.7632	25.2462	-3.5575
29.7972	8.1969	30.2028	-3.6026
34.8442	8.4429	35.1558	-3.5472
39.8926	8.5155	40.1074	-3.4000

44.9407	8.4214	45.0593	-3.1647
49.9866	8.1784	50.0134	-2.8582
55.0289	7.8011	54.9711	-2.4965
60.0665	7.3031	59.9335	-2.0967
65.0985	6.6945	64.9015	-1.6757
70.1246	5.9843	69.8754	-1.2538
75.1458	5.1824	74.8542	-0.8618
80.1745	4.2886	79.8255	-0.5499
85.1703	3.2602	84.8297	-0.3746
90.1195	2.1990	89.8805	-0.2373
95.0609	1.1113	94.9391	-0.1304
100.0000	-0.0000	100.0000	0.0000

L.E. radius = 0.990 percent c

slope of mean line at LE = 0.1902

Root airfoil:

NACA Section 63A418

(Stations and ordinates given

in per cent of airfoil chord)

Upper Surface		Lower Surface	
Station	Ordinate	Station	Ordinate
0.0000	0.0000	0.0000	0.0000
0.5000	1.8226	0.5000	-0.9511
0.7500	2.1423	0.7500	-1.2135
1.2500	2.6849	1.2500	-1.6245
2.5000	3.7246	2.5000	-2.3843
5.0000	5.2327	5.0000	-3.3969
7.5000	6.3836	7.5000	-4.1143
10.0000	7.3239	10.0000	-4.6720
15.0000	8.7963	15.0000	-5.4916
20.0000	9.8805	20.0000	-6.0428
25.0000	10.6655	25.0000	-6.3920
30.0000	11.1883	30.0000	-6.5651
35.0000	11.4513	35.0000	-6.5536
40.0000	11.4750	40.0000	-6.3710
45.0000	11.2712	45.0000	-6.0272
50.0000	10.8698	50.0000	-5.5536
55.0000	10.2950	55.0000	-4.9795
60.0000	9.5663	60.0000	-4.3309
65.0000	8.7008	65.0000	-3.6344
70.0000	7.7168	70.0000	-2.9225
75.0000	6.6378	75.0000	-2.2415

80.0000	5.4752	80.0000	-1.6469
85.0000	4.1704	85.0000	-1.1960
90.0000	2.8056	90.0000	-0.7828
95.0000	1.4317	95.0000	-0.4192
100.0000	-0.0000	100.0000	0.0000

L.E. radius = 1.794 percent c

slope of mean line at LE = 0.1902

NACA 63A418

(Stations and ordinates given

in per cent of airfoil chord)

Upper Surface		Lower Surface	
Station	Ordinate	Station	Ordinate
0.0000	0.0000	0.0000	0.0000
0.2474	1.4409	0.7526	-1.2160
0.4654	1.7753	1.0346	-1.4591
0.9185	2.3380	1.5815	-1.8554
2.0990	3.4230	2.9010	-2.5793
4.5367	4.9877	5.4633	-3.5453
7.0132	6.1788	7.9868	-4.2335
9.5085	7.1519	10.4915	-4.7678
14.5291	8.6762	15.4709	-5.5534
19.5728	9.8003	20.4272	-6.0799
24.6297	10.6164	25.3703	-6.4108
29.6952	11.1640	30.3048	-6.5697
34.7662	11.4444	35.2338	-6.5488
39.8392	11.4778	40.1608	-6.3623
44.9115	11.2766	45.0885	-6.0198

49.9800	10.8717	50.0200	-5.5515
55.0429	10.2894	54.9571	-4.9848
60.0983	9.5506	59.9017	-4.3442
65.1451	8.6738	64.8549	-3.6550
70.1830	7.6788	69.8170	-2.9483
75.2137	6.5901	74.7863	-2.2695
80.2556	5.4120	79.7444	-1.6733
85.2493	4.1029	84.7507	-1.2173
90.1743	2.7579	89.8257	-0.7962
95.0899	1.4067	94.9101	-0.4258
100.0000	-0.0000	100.0000	0.0000

L.E. radius = 1.794 percent c

slope of mean line at LE = 0.1902