DESIGN AND SIMULATION OF A RAMJET

By

Disha Ahluwalia (R180206017)

P. Moudgalya Kumar (R180206041)

Shikha Sharma (R180206053)

DEPARTMENT OF AEROSPACE ENGINEERING

COLLEGE OF ENGINEERING

UNIVERSITY OF PETROLEUM & ENERGY STUDIES, DEHRADUN

MAY 2010

Project Report

On

"Design and Simulation of Ramjet"

Submitted in Partial fulfilment of the Requirements for the degree of

B-Tech (Aerospace Engineering)

By

Disha Ahluwalia (R180206017)

Puligedda Moudgalya Kumar (R180206041)

Shikha Sharma (R180206053)

Under the Guidance of

Dr. Parag Mantri

Approved

....

Dean

Department of Aerospace Engineering

College of Engineering Studies

University of Petroleum & Energy Studies, Dehradun

May, 2010

CERTIFICATE

This is to certify that the work contained in this thesis titled " **Design and Simulation of Ramjet**" has been carried out by **Disha Ahluwalia (R180206017), P. Moudgalya Kumar (R180206041), Shikha Sharma (R180206053)** under my/our supervision and has not been submitted elsewhere for a degree.

Parag Mantri
Mentor

…orate Office :
…carbons Education & Research Society
…loor, PHD House,
…ri Institutional Area
…st Kranti Marg, New Delhi-110 016 India
…91-11-41730151-53 Fax: +91-11-41730154

Main Campus :
Energy Acres,
PO Bidholi Via Prem Nagar,
Dehradun - 248 007 (Uttarakhand), India
Ph. : +91-135-2261090-91, 2694201/203/208
Fax : +91-135-2694204

Regional Centre (NCR) :
SCO, 9-12, Sector-14,
Gurgaon 122 007
(Haryana), India
Ph +91-124-4540 300
Fax +91-124-4540 330

Regional Centre (Rajahmundry) :
GIET, NH 5, Velugubanda,
Rajahmundry - 533 294,
East Godavari Dist., (Andhra Pradesh), India
Tel: +91-883-2484811/55
Fax: +91-883-2484822

Email: info@upes.ac.in URL: www.upes.ac.in

ACKNOWLEDGEMENT

We take this opportunity to thank our mentor **Dr. Parag Mantri**, Adjunct Professor, **Mr. Raj Kishore Tripathi**, Course Coordinator, **Dr. Om Prakash**, Head of Department, Department of Aerospace, University of Petroleum & Energy Studies for their continual support and guidance throughout the project.

We are thankful to our Dean, Dr. Sri Hari for giving us this opportunity to perform the project and being a great inspiration for us throughout.

We would also like to thank our batch mate **Paras Kanwar**, B-Tech (Automotive Design Engineering) for his help and guidance in Computational Fluid Mechanics.

Finally we express a very special sense of appreciation to our parents and the entire University of Petroleum & Energy Studies family who benevolently supported us during our course.

Date: 13 May 2010

(Disha Ahluwalia)

(P. Moudgalya Kumar)

(Shikha Sharma)

Abstract

The work aims on studying the ramjet engine followed by its design and simulation The performance evaluation of the Ramjet has not been performed due to insufficient resources. The design has been done from historical data and references, followed by certain modifications based on various parameters. Ramjets are very efficient of engines which do not have any moving parts. The ramjet uses the high dynamic pressure of the intake air and converts it into static pressure through ramming effect. The intake compresses the air through a series of oblique shocks and hence its design is of crucial importance. The work mainly aims on the design of the inlet using modelling and analysis software. Solidworks 2005 has been used for designing and Ansys CFX has been used for the simulation of flow.

Contents

List of Figures............ viii

List of charts............ ix

Nomenclature............ x

1. Introduction............ 1

2. Literature survey............ 3

2.1 Shocks in supersonic Flow............ 3

2.1.1 Oblique shock............ 3

2.1.2 Relation between θ, β and M............ 3

2.1.3 Supersonic Compression............ 4

2.1.4 Simple and non-simple regions............ 4

2.1.5 Change of total pressure across shock............ 4

2.1.6 Mach Reflection............ 5

2.1.7 Converging & Diverging nozzle and Chocked Flow............ 6

2.2 Design............ 6

2.2.1Intake............ 7

2.2.2Combustor............ 8

2.2.3Nozzle............ 8

2.3 Use of computational fluid mechanics............ 9

2.3.1 Continuity equation............ 9

2.3.2 Momentum equation............ 10

2.3.3 Energy equation............ 10

2.4Simulation of Ramjet Engine............ 11

2.4.1 k-epsilon turbulence model............ 11

2.3.3 The Oswatitcsh Principle............ 12

2.4 Mathematical Calculations............ 13

2.4.1Thrust produced by a Ramjet............ 13

2.4.2 Ramjet Cycle Analysis............ 14

3.Theoretical calculations............ 17

3.1 Intake Cone............ 17

4. Modelling of Intake and Complete Ramjet.. 19

4.1 Intake Design.. 19

4.1.1 Case I... 19

4.1.2 Case II.. 20

4.1.3 Case III... 21

4.2 Entire ramjet.. 23

5. Domain Specification and Meshing.. 25

6. Analysis and Results.. 27

6.1 Intake Case Study... 27

6.1.1 Mach number variation.. 28

6.1.2 Static pressure variation... 30

6.2 Entire Ramjet Analysis.. 31

6.2.1 Mach number variation.. 32

6.2.2 Static pressure variation... 33

7. Difficulties Faced... 35

Conclusion & Future Scope.. 36

References.. 37

Appendix.. 38

List of figures

1.1	Schematic diagram of a Ramjet engine	2
2.1	Different types and conditions for intake	7
2.2	Shock angles b and flow turning angles d for a typical supersonic inlet	12
2.3	Schematic diagram of ramjet depicting the various stations for analysis	14
2.4	T-S diagram for the ramjet engine	14
3.1	Representation of the wedge and the shock angles to determine the distance at which the second δ so that both the shocks meet at one point that is the point of the start of the outer body	18
4.1	2-D view of the Ramjet intake module with dimensions	20
4.2	An isometric view of the Ramjet engine intake designed in solidworks 2005	20
4.3	2-D view of the modified intake module with intake cone diameter 2D	21
4.4	An isometric 3-D view of the intake designed in solid works 2005	21
4.5	2-D view of intake module with a modification in the cone diameter	22
4.6	Isometric 3-D view of the intake module (case 3) designed in solidworks 2005	22
4.7	2-D view of the entire Ramjet with intake cone diameter D= 1ft = 0.304m	23
4.8	Isometric 3-D view of the entire Ramjet designed using Solid works 2005	24
5.1	2-D view of the unstructured tetrahedral meshing of the intake module	25
5.2	3-D view of the meshed intake module. The different colours depict the different faces and curvatures	26
5.3	Meshing of the entire Ramjet as well as the domain	26
6.1	Yellow line depicting the direction and location where the parameters have been plotted	28
6.2	Mach number plot along the axis of the intake module	29
6.3	Static pressure changes through visualization along the intake module	30
6.4	Yellow line depicts the direction and line along which the parameters have been plotted	31

List of Graphs

Graph 6.1	Mach number v/s the axial length of the intake module..................	29
Graph 6.2	Static pressure variation along the length of the intake module, taking 1atm as reference..	31
Graph 6.3	Variation of the Mach number along the length of the Ramjet........	32
Graph 6.4	Variation of static pressure along the length of the entire Ramjet....	34

Nomenclature

$\dot{m}_e$	Mass flow of gases at exit of nozzle
$\dot{m}_0$	Mass flow of air at inlet
$\dot{m}_f$	Mass flow rate of fuel
f	Fuel to air ratio
U_e	Speed of air at the exit of nozzle
U_o	Speed of inlet air
p_e	Pressure at exit of nozzle
p_0	Pressure at inlet of engine
T_e	Temperature at the nozzle exit
T_o	Temperature at the inlet of engine
a_e	Speed of sound at nozzle exit
a_o	Speed of sound at inlet of engine
M_e	Mach number at nozzle exit
M_o	Mach number at the inlet
p_t	Total pressure
T_t	Total temperature
T_a-T_6	Temperatures at different stations
P_a-P_6	Pressures at different sections
τ	Total Temperature ratio across a component
θ	Total temperature to static temperature ratio
δ	Deflection angle
β	Shock angle
M_1	Inlet Mach number
M_2	Mach number after shock

Chapter 1

INTRODUCTION

Ramjet uses the engine's forward motion to compress incoming air, without a rotary compressor. Limitation being it cannot produce thrust at zero airspeed and hence cannot move an aircraft from a standstill.

Ramjet was first made by a French engineer; Rene Lorin in 1913. It is a steady combustion or continuous flow engine and has the simplest construction of all propulsion engines. It consists of a diffuser, a combustion chamber and an exit nozzle. As the ramjet has no compressor, the entire compression is ram compression.

Ramjets require considerable forward speed to operate well, and works most efficiently at speeds around Mach 3. It can operate up to speeds of at least Mach 5.

Ramjets can be very useful in applications requiring a small and simple engine for high speed use; such as missiles, while weapon designers are looking to use ramjet technology in artillery shells to give added range; it is expected that a 120-mm mortar shell, if assisted by a ramjet, could attain a range of 22 mi (35 km). They have also been used successfully, though inefficiently, as tip jets on helicopter rotors.

Ramjets are sometimes confused with pulsejets, which use an intermittent combustion, but ramjets employ a continuous combustion process, and are quite a distinct type of jet engine.

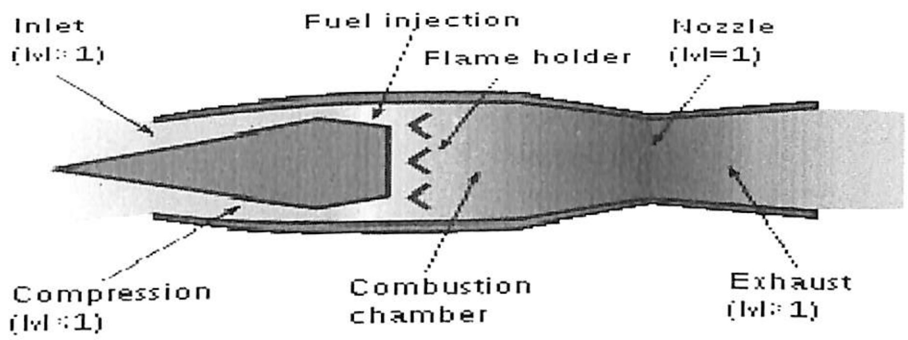


Fig 1.1: Schematic diagram of a Ramjet.

Chapter 2

Literature survey

2.1 Shocks in Supersonic flow

2.1.1 Oblique shocks

An oblique shock is a compression wave inclined at an angle to the flow. For steady subsonic flows, we normally do not think in terms of wave motion. It is much easier to view the motion from a frame of reference system in which the body is stationary and the fluid flows over it. When the relative speed is supersonic, the disturbance waves will not be able to propagate ahead of the immediate vicinity of the body, and the wave system travels with the body. Thus, in the reference system in which the body is stationary, the wave system is also stationary; then the connection between the wave system and the flow field is direct.

Oblique shocks usually occur when a supersonic flow is turned into itself. The opposite to this, i.e. when a supersonic flow is turned away from itself, it marks in the formation of an expansion fan.

2.1.2 Relation between θ, β and M

It is seen that for determining M_2, the deflection angle θ must be known. Moreover, for each wave angle β, at a given M_1, there is a subsequent flow turning angle θ, that is □ can also be expressed as a exclusive function of M_1 and. Hence we can deduce a relationship between the 3 parameters, M_1, θ and β. β If M_1 and θ are known, then the corresponding value of β can be calculated by the use of the following equation.

$$\tan \theta = 2 \cot \beta \left((M_1{}^2 \sin^2 \beta - 1)/(M_1{}^2.(\gamma + \cos 2\beta) + 2)\right)$$

The software MATLAB 7 has been used in further calculations to iterate the values of equivalent β for a particular M_1 and θ.

2.1.3 Supersonic Compression

Compressions in supersonic flow are not typically isentropic. Generally, they take place through shock waves and are non-isentropic. But there are particular cases, for which the compression is isentropic. The weak oblique shocks divide the field near the wall into portions of uniform flow. Further away from the wall the weak shocks might coalesce and from a strong shock. The increase of entropy across a weak shock is of the order of third power of deflection angle.

Thus, if the compression is achieved via a large number of weak waves, the entropy increase can be reduced to a very large extent compared to a single shock giving the same net deflection.

2.1.4 Simple and Non-Simple Regions

The waves that cause isentropic expansion and compression are called simple waves. A simple wave is straight Mach line, with constant flow conditions and is governed by simple relations between flow direction and the Prandtl-Meyer function.

Supersonic expansion or compression with Mach lines which are straight is called simple region. The Mach lines which are straight in the simple region become curved after intersection with other Mach lines. These regions are called non-simple regions.

2.1.5 Change of Stagnation or Total Pressure across the shock

There is no heat added or taken away from the flow as it moves across the shock wave i.e. the flow process across the shock wave is adiabatic. Therefore, the total temperature remains the same aft and before the wave, i.e.

$$T_{02} = T_{01}$$

Now, it is important to note that the above equation valid for a perfect gas, is a special case of the more broad result that the total enthalpy is constant across a normal shock. For a stationary normal shock, the total enthalpy is at all times constant across the wave which, for calorically or thermally perfect gases, translates into a constant total pressure across the shock. However, for a chemically reacting gas, the total temperature is not constant across the shock. Also, for a shock wave which is not stationary, neither the total enthalpy nor the total temperature is constant across the shock wave. For an adiabatic process of a perfect gas, we have

$$S_{02} - S_{01} = R \ln(p_{01}/p_{02})$$

In the above equation, all the quantities are expressed as stagnation quantities. It is evident from the equation that the entropy varies only when there are losses in pressure. It is not dependent on velocity, and hence there is nothing like stagnation entropy. Therefore

$$S_2 - S_1 = R \ln(p_{01}/p_{02})$$

The exact expression for the ratio of total pressures may be obtained as

$$P_{02}/P_{01} = (1+(2\gamma/\gamma+1)(M_1^2 - 1))^{-1/(\gamma-1)} ((\gamma+1)M_1^2/(\gamma-1)M_1^2+2))^{\gamma/(\gamma-1)}$$

The above equation is a significant and useful equation since it connects the stagnation pressures on either side of a normal shock to the flow Mach number ahead of the shock. Also, we can see the utility of the equation from the application aspect. When a pitot probe is placed in a supersonic flow in front of the flow, there is a detached shock standing ahead of probe nose and therefore the probe measures the total pressure behind that normal shock. Having the stagnation pressure ahead of the shock, which is the pressure in the reservoir, for isentropic flow up to the shock, we can determine the flow Mach number ahead of the shock.

2.1.6 Mach Reflections

A look at the detached shock field will show that the difficulties are because of the appearance of subsonic regions in the flow. Similar problems leading to a condition where no solution with simple oblique shock waves is possible will arise in a flow field with shock reflections, when the Mach number M_2 after the incident shock is lower than the disconnection Mach number for a given θ. These flow configurations are called Mach reflections.

Intersection of normal and right- running oblique shock produces a reflected left-running oblique shock, in order to bring the flow into the original direction.

The left-running shock must have a reduced amount of strength compared to the right-running shock because of the deflection θ involved in the process, but M1>M2. It may so happen that M_2 is less than the detachment M for the wall deflection required; in such a case, the whole picture of the flow field changes, all the shocks become curved, and the flow behind the shock system need not be parallel to the wall. Some other phenomenon might take place afterwards to bring the flow parallel to the wall.

A similar phenomenon also takes place when two oblique shocks of opposite family cross with a normal shock with a normal shock bridging them.

From the discussions of oblique shock we observe that, in case of oblique shocks, the strong shock solution is physically impossible. But the detached shocks are a part of the strong shock solution.

2.1.7 Converging-Diverging Nozzle and hocked Flow

It is important to comprehend that the statement Supersonic Velocity can be reached only if $p_e/p_o < 0.528$ is the essential requirement for the nozzle to choke at the throat(i.e. to have $M^* = 1$ at the throat). Thus, in the exacting sense it should be stated that, a nozzle will have $M_{th} = 1$ at throat only if $p_{th}/p_o <= 0.528$, and the flow in the divergent of the section of the nozzle will accelerate to increasing supersonic Mach numbers only if $p/p_{th} <1$, where p is the local static pressure in the divergent portion of the nozzle. The flow after choking at the nozzle throat carry on to accelerating to increasingly higher supersonic Mach numbers, in the divergent portion downstream of the throat only if $p_e/p_{th} < 1$.Consequently, for the flow to accelerate, a favourable pressure gradient should exist. Thus, for a convergent-divergent nozzle to experience supersonic flow from downstream, of the throat up to the exit, the limiting pressure ratio p_{th}/p_o required across the nozzle is dictated by the presence of normal shock at the exit. Therefore, p_e/p_o required to choke the flow at the nozzle can also be greater than the isentropic limiting pressure ratio of 0.528.

2.2 Design

A ramjet is designed around its inlet. An object moving at high speed through air produces a high pressure region in front and a low pressure region to the rear. A ramjet uses this high pressure in front of the engine to force air through the tube, where it is heated by combustion with fuel. It is then passed through a nozzle and accelerated to supersonic speeds. This acceleration provides the ramjet forward thrust.

A ramjet is also referred to as a 'flying stovepipe', a very simple device comprising an air intake, a combustor, and a nozzle. The only moving parts are those within the turbopump, which pumps the fuel to the combustor in a liquid-fuel ramjet. Solid-fuel ramjets are simpler.

By way of contrast, a turbojet uses a gas turbine driven fan to compress the air. This gives higher efficiency and far more power at low speeds, where the ram effect is weak, but is also more complex, heavier and more expensive, and the temperature limits of the turbine section, limits the top speed and thrust at high speed.

2.2.1 Inlet

In Ramjets high dynamic pressure within the air approaching the intake lip is exploited. An efficient intake should recover much of the freestream stagnation pressure, which is used to support the combustion and expansion process in the nozzle.

Most ramjets operate at supersonic flight speeds and use either one or a number of conical (or oblique) shock waves, terminated by a strong normal shock, to slow down the airflow to a subsonic velocity at the exit of the intake. Further diffusion is then required to get the air velocity down to a level suitable for the combustion process.

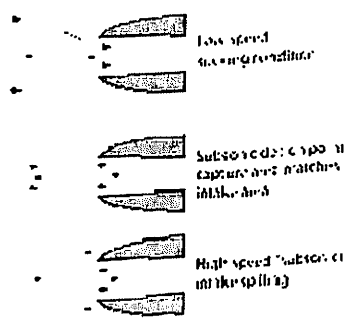


Fig 2.1 Different types and conditions for intake

Subsonic ramjets don't need such a complicated inlet since the airflow is already subsonic and a simple hole is usually used. This would also work at slightly supersonic speeds, but as the air starts to choke at the inlet, this becomes inefficient.

2.2.2 Combuster

As with other jet engines the combustor's job here is also to generate hot air. It does this by burning fuel with the air at constant pressure. The airflow through the jet engine is generally quite high, so sheltered combustion zones are required and produced by using flame holders that stop the flames blowing out.

Due to the absence of a downstream turbine, a ramjet combustor can safely operate at stoichiometric fuel-air ratios, which produces a combustor exit stagnation temperature of the order of 2400 K for kerosene. The combustor must be capable of operating over a wide range of throttle settings, for a range of flight speeds/altitudes. Usually a sheltered pilot region enables combustion to continue when the vehicle intake undergoes high yaw/pitch, during turns. Other flame stabilization techniques make use of flame holders, which vary in design from combustor cans to simple flat plates, to shelter the flame and provide for better fuel mixing. Overfuelling the combustor can result in the normal shock within a supersonic intake system being pushed forward beyond the intake lip, resulting in a substantial drop in engine airflow and net thrust.

2.2.3 Nozzles

The exit nozzle is a critical part of a ramjet design, since it accelerates exhaust flow to produce thrust.

In case of subsonic flight Mach number, exhaust flow is accelerated through a converging nozzle. For a ramjet operating at supersonic flight Mach number, acceleration is typically achieved via a convergent-divergent nozzle.

Ramjets generally give little or no thrust below the airspeed of about half the speed of sound, and they are inefficient (less than 600 seconds due to low compression ratios) until the speed exceeds 1000 km/h (600 mph). Even above this minimum speed a wide flight envelope (range of flight conditions), consisting of low to high speeds and low to high altitudes, can force significant design compromises, and they tend to work best when optimised for one designed speed and altitude (point designs). However, ramjets generally outperform gas turbine based jet engines at supersonic speeds (mach 2-4). Although inefficient at the slower speeds they are more fuel-efficient than rockets over their useful working range.

2.3 Use of Computational Fluid Mechanics:

In our simulation, using numerical methods, finite volume method, a full model of a formula body was developed and simulated. A hybrid mesh for **turbulent, inviscid, steady and transient state, 3d simulation** was used. Surface as well as volumetric meshes were created.

Here a finite computational fluid domain was made in which the vehicle bluff body was incorporated. The surface mesh of the body was created and then the domain was discretized into fine elements using tetrahedral elements. Element control was given by varying element size, radius of influence, inflations, maximum and minimum spacing.

The next step was to give the boundary conditions after defining regions. The boundary conditions remained same for all the analyzed bodies. The various equations used are:

2.3.1 Continuity equation

A continuity equation is a differential equation that describes the transport of some kind of conserved quantity. Since mass, energy, momentum, electric charge and other natural quantities are conserved, a vast variety of physics may be described with continuity equations.

The general form for a continuity equation is

$$\frac{\partial \varphi}{\partial t} + \nabla . v = s$$

Where φ is some quantity, v is a vector function describing the flux, and s describes the generation (or removal) rate of φ. This equation may be derived by considering the fluxes into an infinitesimal box. If φ is a conserved quantity, the generation or removal rate is zero:

$$\frac{\partial \varphi}{\partial t} + \nabla . v = 0$$

In fluid dynamics, the continuity equation is a mathematical statement that, in any steady state process, the rate at which mass enters a system is equal to the rate at which mass leaves the system. The differential form of the continuity equation is:

$$\frac{\partial \rho}{\partial t} + \nabla . (\rho u) = 0$$

Where ρ is fluid density, t is time, and u is fluid velocity. If density (ρ) is a constant, as in the case of incompressible flow, the mass continuity equation simplifies to a volume continuity equation:

$$\nabla . u = 0$$

this means that the divergence of velocity field is zero everywhere.

2.3.2 Momentum equation

The momentum equation is based on Newton's second law of motion that states that the net force on the fluid element equals its mass times the acceleration of the element.

Thus the x component of the momentum equation is given by:

$$\frac{\partial (\rho u)}{\partial t} + \nabla . (\rho u V) = -\frac{\partial p}{\partial x} + \frac{\partial \tau_{xx}}{\partial x} + \frac{\partial \tau_{yx}}{\partial y} + \frac{\partial \tau_{zx}}{\partial z} + \rho f_x$$

Similarly the y and z components are given by:

$$\frac{\partial (\rho v)}{\partial t} + \nabla . (\rho v V) = -\frac{\partial p}{\partial y} + \frac{\partial \tau_{yy}}{\partial y} + \frac{\partial \tau_{xy}}{\partial x} + \frac{\partial \tau_{zy}}{\partial z} + \rho f_y$$

$$\frac{\partial (\rho w)}{\partial t} + \nabla . (\rho w V) = -\frac{\partial p}{\partial z} + \frac{\partial \tau_{xz}}{\partial x} + \frac{\partial \tau_{yz}}{\partial y} + \frac{\partial \tau_{zz}}{\partial z} + \rho f_z$$

2.3.3. Energy equation

The energy equation is based on the physical principle that energy is conserved. It can neither be created nor destroyed. It can only be transformed from one form to another.Thus the energy equation is given by:

$$\frac{\partial}{\partial t}\left[\rho\left(e+\frac{V^2}{2}\right)\right]+\nabla\left[\rho\left(e+\frac{V^2}{2}\right)V\right]=\rho\dot{q}+\frac{\partial}{\partial x}\left(k\frac{\partial T}{\partial x}\right)+\frac{\partial}{\partial y}\left(k\frac{\partial T}{\partial y}\right)+\frac{\partial}{\partial z}\left(k\frac{\partial T}{\partial z}\right)-\frac{\partial(up)}{\partial x}-\frac{\partial(vp)}{\partial y}-\frac{\partial(wp)}{\partial z}$$
$$+\frac{\partial(u\tau_{xx})}{\partial x}+\frac{\partial(u\tau_{yx})}{\partial y}+\frac{\partial(u\tau_{zx})}{\partial z}+\frac{\partial(v\tau_{xy})}{\partial x}+\frac{\partial(v\tau_{yy})}{\partial y}+\frac{\partial(v\tau_{zy})}{\partial z}+\frac{\partial(w\tau_{xz})}{\partial x}+\frac{\partial(w\tau_{yz})}{\partial y}+\frac{\partial(w\tau_{zz})}{\partial z}+\rho f.V.$$

2.4 Simulation of Ramjet Engine

The performance of a ramjet depends on the combination of the efficient performance of both, the intake and the combustion chamber. Since the performance depends on their interdependence, there is a requirement for the simulation of a complete ramjet engine to analyze its realistic performance characteristics. With the analysis of the engine in totality we can better choose the proper location and size of air inlets as well as combustor geometry to obtain the nest results. At this time deep emphasis must be laid on the interaction of intake, combustor and nozzle. For example, if sub-critical conditions exist, an unattached shock may lead to air spillage resulting in poor combustion and loss of thrust. Whereas in super-critical condition as the shock train moves into the diffuser, there maybe poor pressure recovery again resulting in lower propulsive efficiency. The geometries of the intake and the heat release in the combustor affect the locations of the shock as well as the shock reflection. The extent of flow acceleration in the nozzle is dependent on both the stagnation pressure and stagnation temperature conditions achieved in the combustor section. The complete engine simulation is useful in understanding internal flows including flame holding and spreading, inlet/combustor coupling, shockwave/boundary layer interactions and combustion dynamics.

2.4.1 k-epsilon Model

Two equation turbulence models are the most common turbulence models. Models like the k-epsilon have become the industry standard models and are commonly used for most types of engineering problems

By definition, a two equation model has two extra transport equations to represent the turbulent properties of the flow. This allows a two equation model to account for history effects like convection and diffusion of turbulent energy.

Generally one of the transported variables is the turbulent kinetic energy, k. Common choices for second transported variable are the turbulent dissipation, ϵ or the specific dissipation, ω. The second variable is the variable that determines the scale of the turbulence (length-scale or time-scale), whereas the first variable, k determines the energy in the turbulence.

In K-Epsilon turbulence model the two transport equations which are solved are for the turbulent kinetic energy **k** and its dissipation rate ϵ.

2.4.2 The Oswatitcsh Principle

The principle states that for a two-dimensional supersonic duct, the maximum shock pressure-recovery can be obtained if the strength of the train of oblique shocks are equal[1] .That is, the components of Mach numbers in directions normal to the shocks are equal

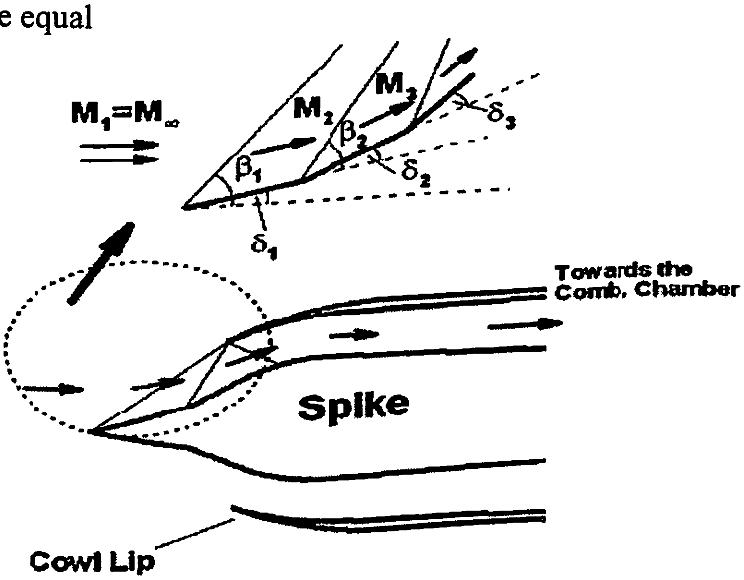


Fig2.2: Shock angles b and flow turning angles d for a typical supersonic inlet

$M_1\sin\beta_1 = M_2\sin\beta_2 = M_3\sin\beta_3 = \dots\dots\dots$

M_1, M_2, M_3 = Mach number corresponding to various shocks.

β_1 β_2 β_3 = Shock angles corresponding to various deflection angles δ_1, δ_2, δ_3

[1] Akbarzadeh.M and Kermani.J.M , NUMERICAL SIMULATIONS OF INVISCID AIRFLOWS IN RAMJET INLETS.

2.4 Mathematical Calculations

2.4.1 Thrust Produced by a Ramjet

All aircraft engines are heat engines, in that they use the thermal energy derived from combustion of fossil fuels to produce mechanical energy in the form of kinetic energy of an exhaust jet. The excess momentum of the exhaust jet over that of the incoming airflow results in thrust, which is used to propel the aircraft. The thrust of a ramjet engine can be calculated using the formula.

$$T = \dot{m}_e U_e - \dot{m}_o U_o + (p_e - p_o)A_e$$

$$\dot{m}_e = (1+f)\dot{m}_o \qquad\qquad f = \frac{\dot{m}_f}{\dot{m}_o}$$

This equation1 may be written in a convenient dimensionless form as shown below:

$$\frac{T}{\dot{m}_o U_o} = (1+f)\frac{U_e}{U_o} - 1 + \frac{p_o A_e}{\dot{m}_o U_o}\left(\frac{p_e}{p_o} - 1\right)$$

In the modelling of aircraft engines, if we assume that the exhaust pressure is equal to the ambient pressure $p_e = p_o$, and that $f \ll 1$, then the expression becomes:

$$\frac{T}{\dot{m}_o a_o} = M_o\left(\frac{U_e}{U_o} - 1\right)$$

In this expression we have introduced the non-dimensional Mach number, $M_0 = U_0/a_0$, where a_0 is the local speed of sound. We can write the ratio of exhaust to inlet velocity ratio as:

$$\frac{U_e}{U_o} = \frac{M_e\sqrt{\gamma R T_e}}{M_o\sqrt{\gamma R T_o}} = \frac{M_e}{M_o}\sqrt{\frac{T_e}{T_o}}$$

It is most efficient to find the exit Mach number, M_e, and temperature, T_e, by keeping track of the stagnation temperatures and pressures through several components. In general, it is the stagnation properties that most conveniently represent the effect of the components on the fluid as it flows through the engine. The relations for stagnation temperature and pressure are given below

$$\frac{T_t}{T} = 1 + \frac{\gamma - 1}{2} M^2 \qquad \frac{p_t}{p} = \left(1 + \frac{\gamma - 1}{2} M^2\right)^{\frac{\gamma}{\gamma - 1}}$$

2.4.2 Ramjet Cycle Analysis

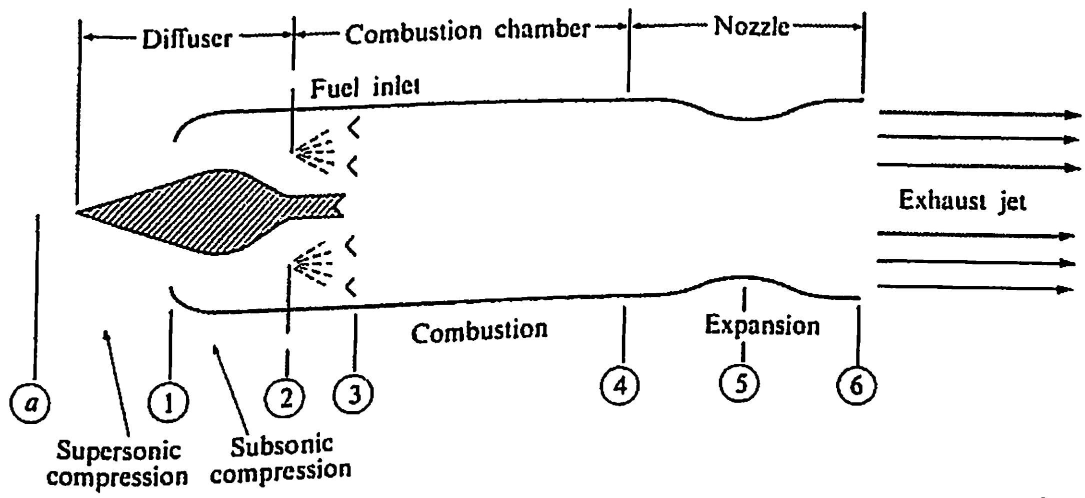


Fig 2.3: Schematic diagram of ramjet depicting the various stations for analysis[2]

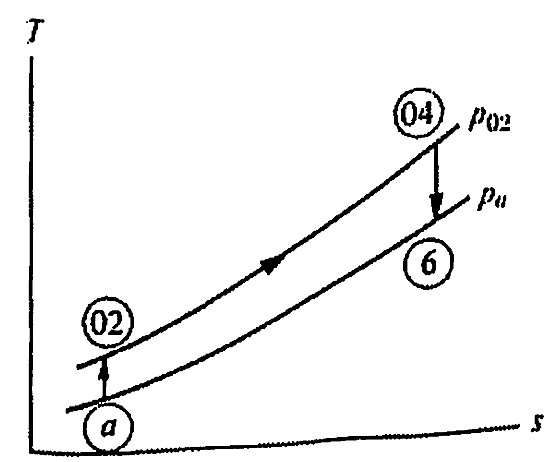


Fig 2.4: T-S diagram for the ramjet engine

2

The reason that the combustor has been divided into two parts is to keep the numbering convention consistent with the turbojet and turbofan engines, which are far more common propulsion devices than the ramjet. In the turbojet and turbofan, the diffuser exit / compressor inlet is always station 2, the compressor exit / combustor inlet is always station 3, and the combustor exit / turbine inlet is always station 4. Station 2 is kept as the exit of the diffuser, but since there is no rotating machinery compressor, station 3 is kept as the inlet to the combustor.

Examining the T-S diagram, we can make several modelling approximations:

1. The compression and expansion processes are taken to be isentropic, i.e. the process is reversible and there is no heat transfer (adiabatic). On the T-S diagram this corresponds to legs a-02 and 04-6. For all isentropic processes the total temperature and total pressure is a constant.
2. The combustion process, between 02 and 04 (or neglecting the fuel injectors, between 03 and 04) is done at low speed (M<0.3), and is modelled as constant pressure heat addition. Thus the stagnation pressure remains constant, although the stagnation temperature increases due to the heat addition (combustion).

Items (1) and (2) imply that the stagnation pressure remains constant throughout the ramjet engine. We will make use of this observation in modelling the ramjet engine.

$$P_{t0} = P_{t6}$$

$$\frac{P_{t0}}{P_0} = \left(1 + \frac{\gamma - 1}{2} M_0^2\right)^{\gamma/\gamma-1}$$

$$\frac{P_{t6}}{P_6} = \frac{P_{t6}}{P_e} = \left(1 + \frac{\gamma - 1}{2} M_e^2\right)^{\gamma/\gamma-1}$$

Where M_0 is the flight Mach number and M_e is the exit plane Mach number. If we assume that the nozzle is ideally expanded, then $P_e = P_0$, and we can write:

$$\frac{P_{t0}}{P_0} = \frac{P_{t6}}{P_e}$$

$$M_e = M_0$$

This implies that:

$$\frac{U_e}{U_o} = \frac{a_e}{a_0} = \frac{\sqrt{T_e}}{\sqrt{T_o}} = \sqrt{\frac{T_{t6}}{T_{t0}}} = \sqrt{\frac{T_{t4}}{T_{t3}}}$$

Now substitute this result into the thrust equation below:

$$\frac{T}{\dot{m}_0 a_o} = M_o\left(\frac{U_e}{U_o} - 1\right)$$

$$\frac{T}{\dot{m}_0 a_o} = M_o\left(\sqrt{\frac{T_{t4}}{T_{t3}}} - 1\right)$$

The ratio T_{t4}/T_{t3} is the total temperature ratio across the combustor, which can be written in shorthand as τ_b. So, the thrust equation becomes:

$$\frac{T}{\dot{m}_0 a_o} = M_o\left(\sqrt{\tau_b} - 1\right)$$

This equation may also be written as:

$$\frac{T}{\dot{m}_0 a_o} = M_o\left(\sqrt{\frac{\theta_4}{\theta_0}} - 1\right)$$

These equations point out some interesting aspects of the ramjet engine:

1. Ramjets (or scramjets) develop no static thrust; they must be moving to develop thrust. This will be in direct contrast to turbojets and turbofans.
2. The device relies on 'ram' compression of the air, and has no moving parts (no spinning compressor to compress the air prior to combustion). To have efficient compression of the air, the ramjet requires high flight speeds.
3. The performance of the device relies in the stagnation temperature rise across the burner.

Chapter 3

Theoretical calculations

3.1 Intake cone:

The half angle of the intake cone is taken as = 14.03 deg.

Diameter of the intake cone considered = 0.304m = 1 ft

Intake cone part = 2D = 0.608m

Constant duct area = 3D = 0.912m

Angle of intake cone = $\tan^{-1}(0.5D/2D)$ = 14.03 deg.

In Case 2, when the cone maximum diameter is made 2D = 0.608m

Angle of intake cone = $\tan^{-1}(D/2D)$ = 26.56 deg.

Design of Case 3 intake by the application of Oswatwitch principle.

Initial angle of intake cone = 14.03 deg

Intake Mach number = 2.5

Using MATLAB code *(appendix-I)*, the shock angle β_1 = 35.919 deg.

From Ostwatitch principle

$M_1 \sin\beta_1 = 2.5 * \sin(35.919) = 1.41$

We know from M_1 and M_2 relations that for inlet mach number 2.5 and theta 14.03 degree, the M_2 value is 1.917

Hence the value of β_2 can be determined as $\beta_2 = 49.91$

Hence by considering the angles of the triangle, the value of delta and the distance at which the deflection is to be given can be calculated.

As shown in the figure below, considering the data already specified the angles to be calculated can be obtained using simple relations.

From calculations it is fount that the slant distance at which an angle of 15,45 degrees is to be given is 0.41m. the axial distance of 0.41m.

Chapter 4

Modeling of Intake and the complete Ramjet

The intake and the complete Ramjet engine has been designed using historical data as well as the operating conditions.

In the design of the intake, three cases have been considered.

Case 1. From historical data and references.

Case 2. Diameter of intake cone changed.

Case 3. Intake cone geometry modified by the application of Ostwatitsch principle.

The design of the entire Ramjet has been done considering historical data amd the fact that there should be sufficient space defined for the instalment of a solid propellant for taking the missile from rest to supersonic speeds.

4.1 Intake Design

The design of intake has been done from historical data as well as the Mach number at which it will be operating. As shown in the figure, the total intake, consisting of the intake, constant duct, the subsonic diffuser, and the diameter, length of the outer tube has been specified.

The point of the start of the outer body as compared to the inlet cone has been calculated using the relation between Mach number M_1, β and θ, using a MATLAB code shown in *appendix-I.* The design taken here is tested in CFX and modifications are brought about to optimize the intake.

4.1.1 Case 1

1. D = 0.304m
2. Total length of intake = 2.74m
3. Length of intake cone part = 2.432m
4. Length of outer body (tube) = 2.148m

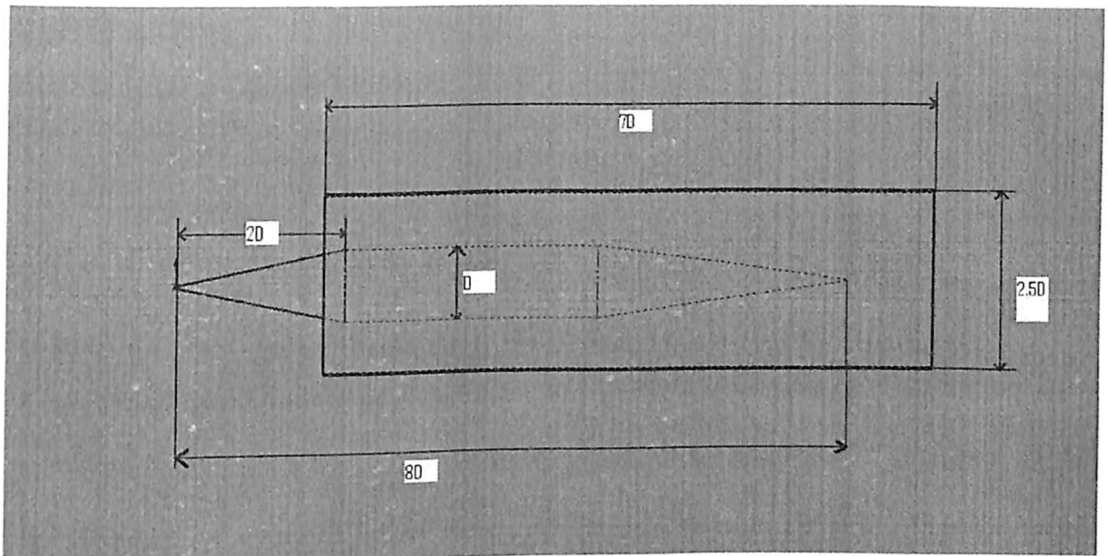


Fig 4.1: 2-D view of the Ramjet intake module with dimensions.

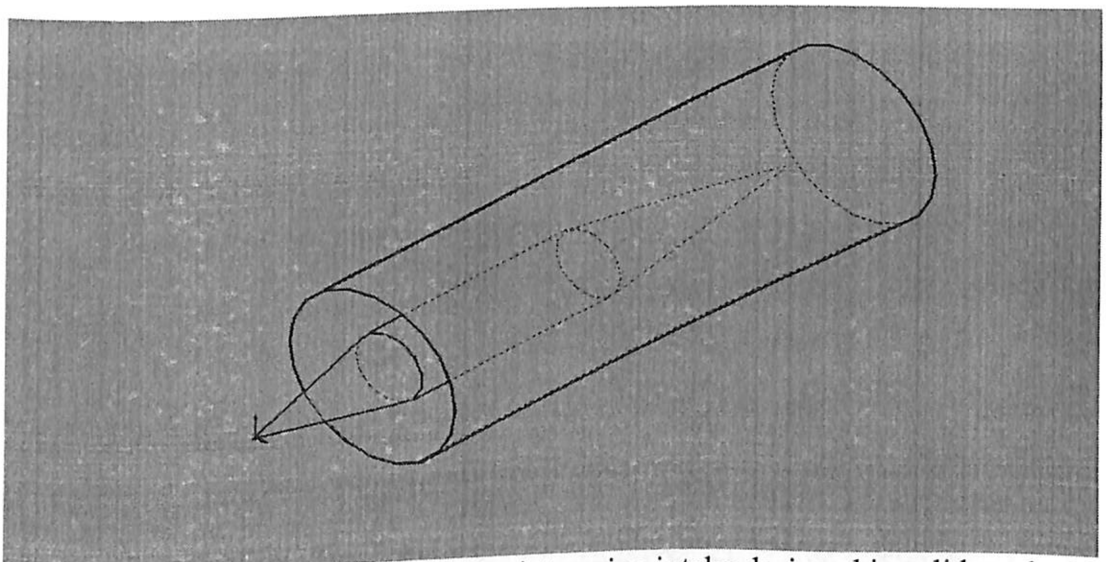


Fig 4.2: An isometric view of the Ramjet engine intake designed in solid works 2005.

4.1.2 Case 2

1. Diameter of intake cone = 0.608m
2. Length of intake cone part = 2.432m
3. Length of outer body (tube) = 2.128m

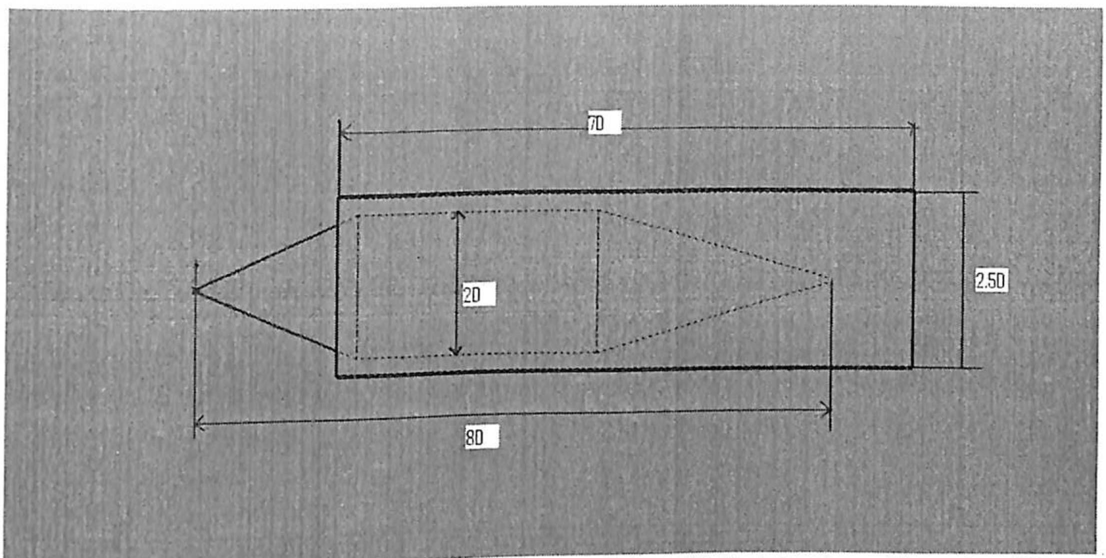


Fig 4.3: 2-D view of the modified intake module with intake cone diameter 2D

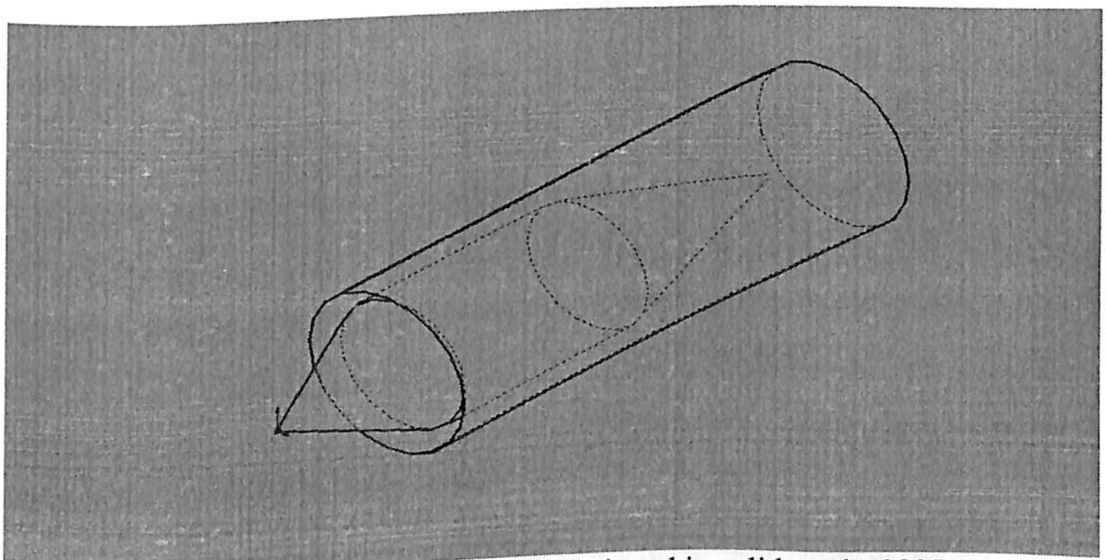


Fig 4.4: Isometric 3-D view of the intake designed in solid works 2005.

4.1.3 Case 3

The major modification made in the intake module in case 3 is the use of Oswatwitch principle. The principle states that for a supersonic flow to undergo a series of isentropic compression, the following relation should be satisfied.

$M_1 \sin\beta_1 = M_2 \sin\beta_2 = M_3 \sin\beta_3 = \ldots\ldots$

M_1, M_2, M_3, are mach consecutive mach numbers after a series of shocks.

β_1 β_2 β_3 are the consecutive shock angles corresponding to the above mach numbers.

1. D = 0.304m
2. Diameter of intake cone = 0.608m
3. Length of intake cone part = 2.432m
4. Length of outer body (tube) = 2.128m

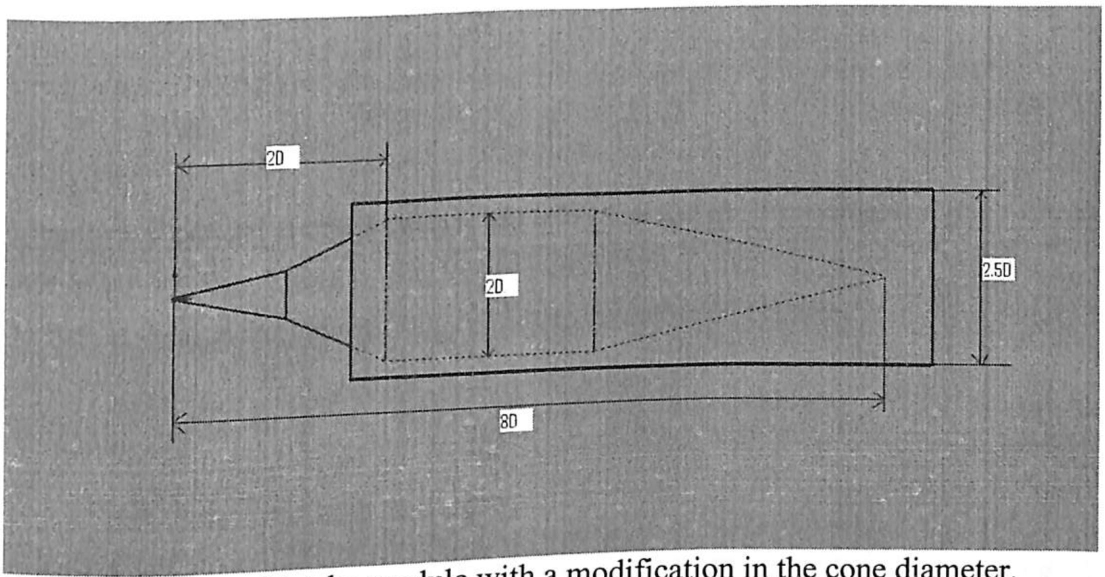


Fig4.5: 2-D view of intake module with a modification in the cone diameter.

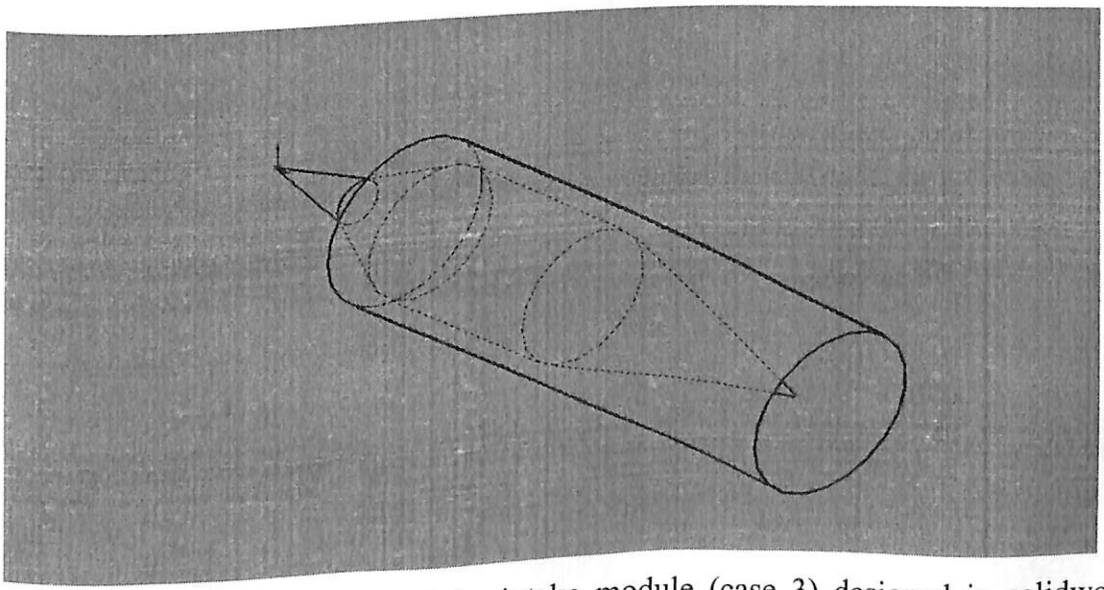


Fig4.6: Isometric 3-D view of the intake module (case 3) designed in solidworks 2005.

4.2 Entire Ramjet

The design of the entire Ramjet has been done from the reference paper and taking the reference diameter 'D' to be 1 feet. Since the dimensions are in meters, the dimension is taken as 0.304m.

The following are the main aspects that have been considered while designing the entire body

1. A dump type combustion chamber is made by providing a sudden expansion of 0.3D as shown in figure.
2. Sufficient length has been provided for the inclusion of a solid propellant incase the missile is fired from the ground.
3. The dump zone for the combustion chamber has been covered by a streamline curvature to prevent wave drag.

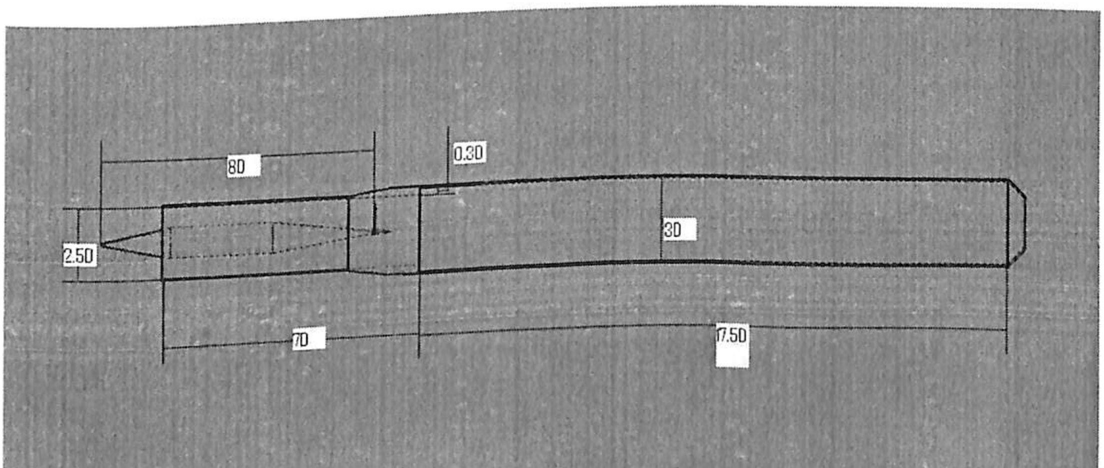


Fig4.7: 2-D view of the entire Ramjet with intake cone diameter D= 1ft = 0.304m.

1. D = 0.304m
2. Length of intake cone part = 2.432m
3. Length of the outer body of intake (tube) = 2.432m
4. Dump zone length (one side) (0.3D) = 0.0912m
5. Length of combustion chamber = 5.32m
6. Total length of the missile (without nozzle) = 8.08m

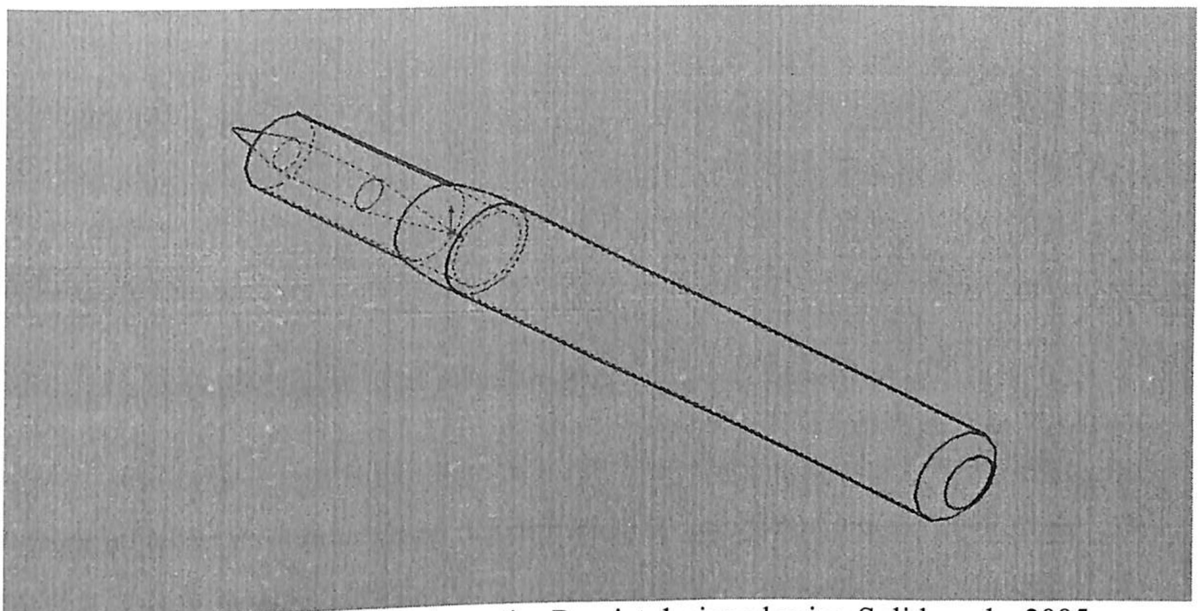


Fig 4.8: Isometric 3-D view of the entire Ramjet designed using Solid works 2005.

Chapter 5

Domain specification and Meshing

The model of the design is exported in IGES format and is loaded in CFX. Using the modelling tools, a cuboid shaped domain space is constructed around the Ramjet. The body is then subtracted from the domain space using Boolean operations to embed the body in the domain. The faces of the ramjet engine are then labeled including the various other domain locations such as inlet, outlet, boundary walls etc.

The meshing done here consists of two parts, surface mesh and volumetric mesh. For surface meshing, a triangular mesh is used because the memory of the computer would not be capable to solve for square or hexagonal mesh. The meshing on the intake cone part is made more fine since the flow is to be properly studied in this region. Once the surface meshing is done, volumetric meshing is begun.

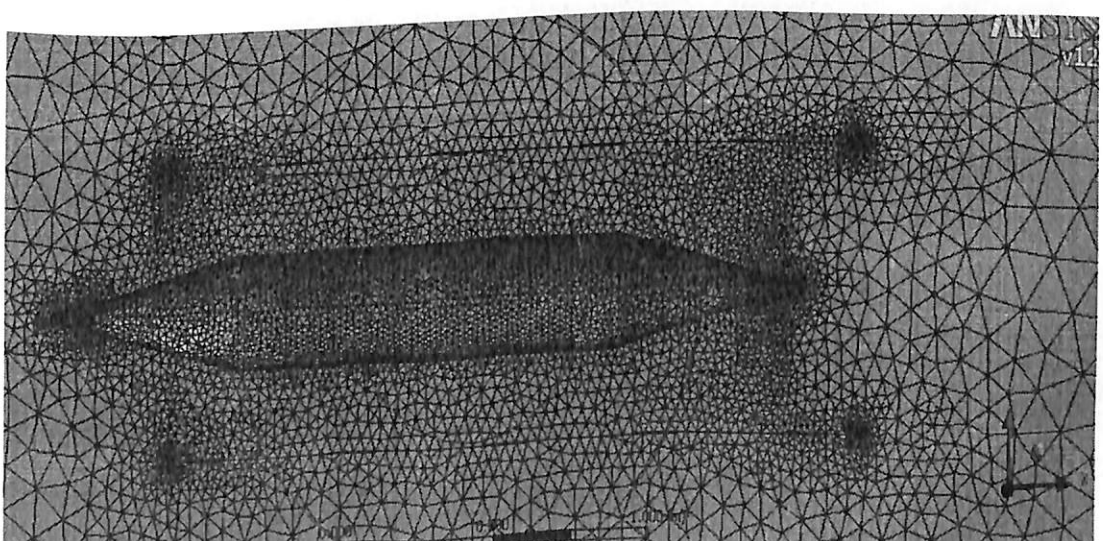


Fig 5.1: 2-D view of the unstructured tetrahedral meshing of the intake module.

The type of volumetric meshing done here is tetrahedral shape. The size of the elements inside outer body has been made more fine in order to obtain accurate results. The meshing in the remaining domain has been made coarser in order to reduce the calculation time. CFX has an option of automatically changing the mesh

size according to the priorities of the user. Advancing front method has been used here wherein the volumetric meshing of the domain advances from the surface mesh of the intake cone. The growth rate and maximum size of the mesh is specified.
The above figure depicts the meshing of the entire domain and the figure below shows the surface meshing of the body with the different colours showing different curvatures. The iterations will be done at each node.
Figure 5.3. Shows the snapshot of the entire Ramjet meshed using CFX mesh. Similar triangular and tetrahedral meshes are formed. Since this case is studied with no heat addition, the walls of the engine are kept thin and without thickness.

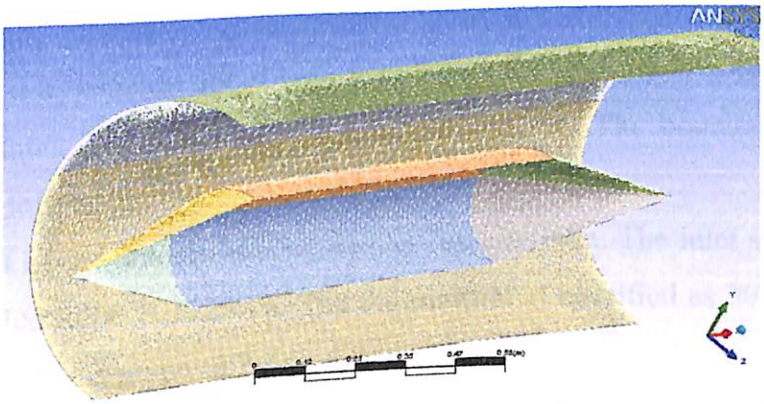


Fig 5.2: 3-D view of the meshed intake module. The different colours depict different faces and curvatures

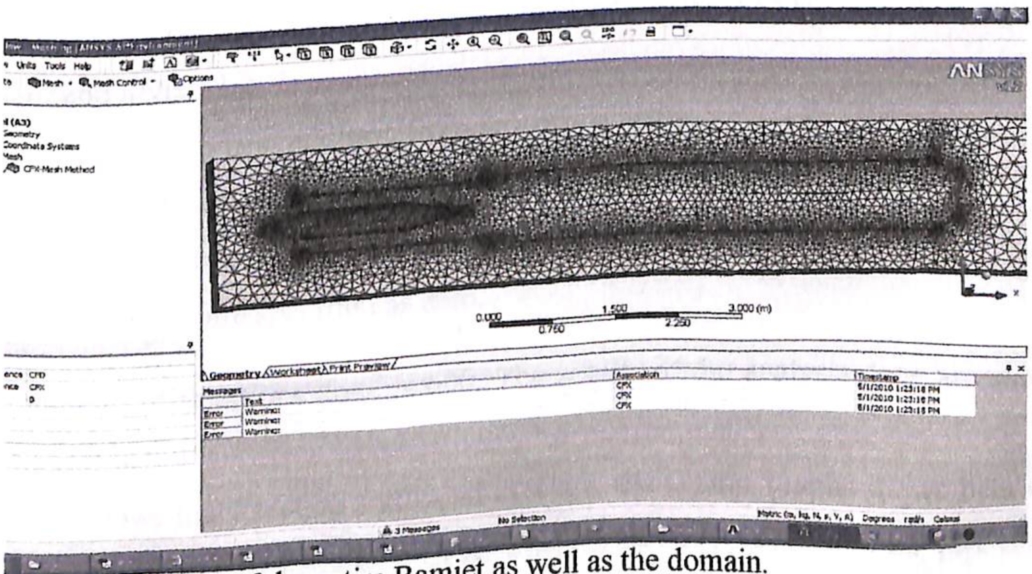


Fig 5.3: Meshing of the entire Ramjet as well as the domain.

Chapter 6

Analysis and Results

6.1 Intake Case Study

Since the body is symmetric about both the planes, the analysis is done only on one half of the Ramjet. The analysis is also done in CFX. The boundary conditions specified are as follows.

1. The inlet of the domain is specified as velocity inlet. The inlet static pressure is specified to be 101325 Pa. The temperature is specified as 300K. The inlet Mach number is given as 2.3.
2. The walls of the boundary as well as the body are no slip walls and that the fluid will interact with the boundaries. They are also specified to be adiabatic.
3. The outlet flow conditions are preset in the case of supersonic flow.
4. The working fluid is specified as ideal air and is considered to be compressible and inviscid.
5. There is no heat addition from any of the walls.
6. The turbulence model used in the analysis is K-€ model.
7. The radial velocities and radial derivatives of all the variables along the axis of symmetry are specified as zero.

The number of iterations done is 600. The results of the analysis done are shown in the figures below.

The line shows the direction and location where the results, graphs shown below have been plotted. This is a feature of CFX, wherein the specification of the plot line will provide a report on the required plots such as Mach number, static pressure, total pressure etc. The parameters studied here are static pressure and Mach number.

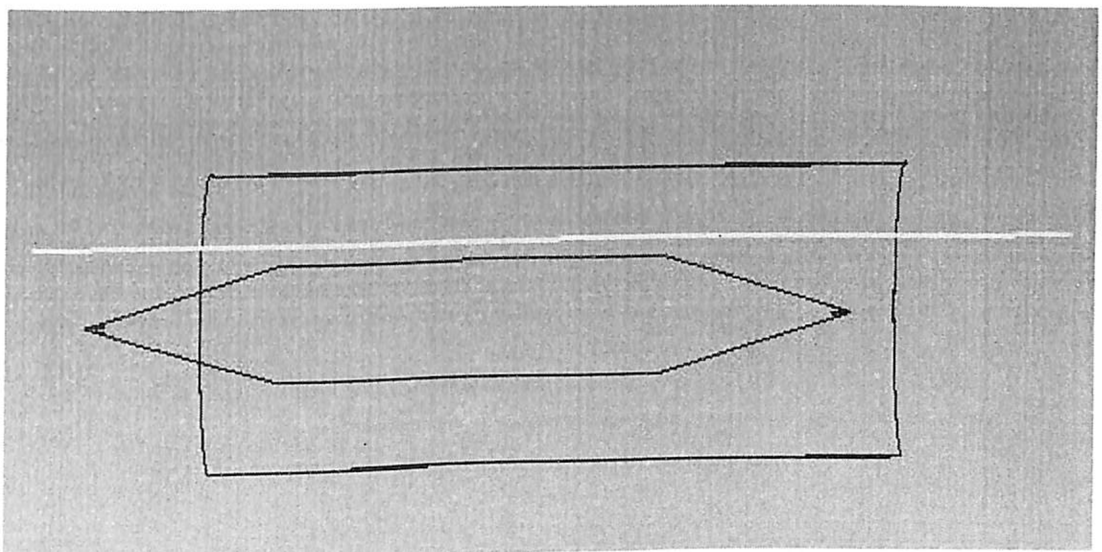


Fig 6.1: Yellow line depicting the direction and location where the parameters have been plotted.

6.1.1 Mach number variation:

The formation of shock waves can be observed by the abrupt change in parameters. As the flow passes through the shock waves, the Mach number decreases. The other figure depicts the plot of Mach number along the length of the intake.

The following are the observations and conclusions based on the visualization and the charts obtained.

6.1.1.1 Observations

1. The formation of oblique shock train can be observed by the sudden change in mach numbers.
2. The shock gets reflected from the outer tube and undergoes a series of reflection.
3. The normal shock is not created before the subsonic diffuser due to which the flow again undergoes expansion and becomes supersonic at the outlet.
4. The formation of oblique shock train can be observed by the sudden change in mach numbers.
5. The shock gets reflected from the outer tube and undergoes a series of reflection.
6. The normal shock is not created before the subsonic diffuser due to which the flow again undergoes expansion and becomes supersonic at the outlet.

7. The Mach number goes on decreasing in the graph until it reaches the diffuser, where it expands.

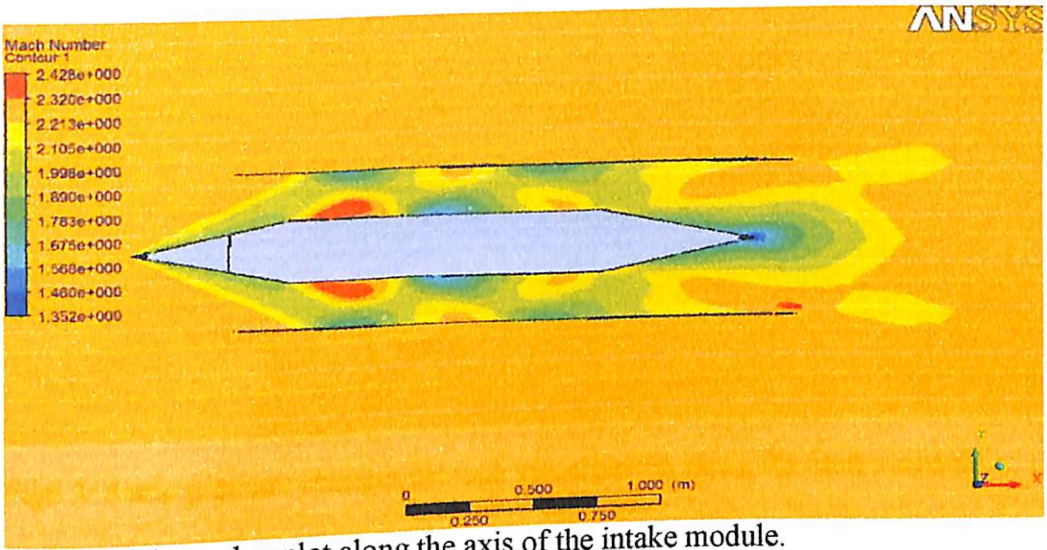
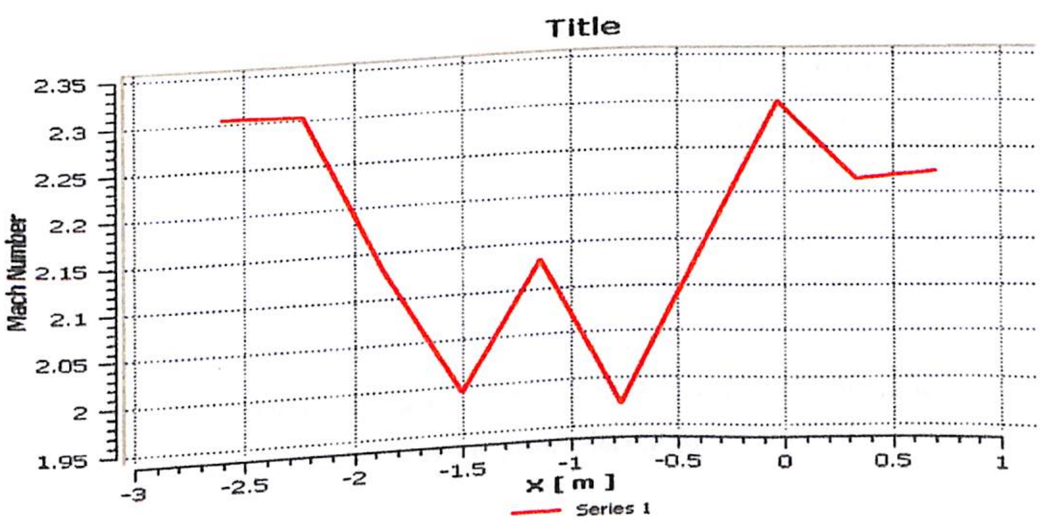


Fig6.2 Mach number plot along the axis of the intake module.



Graph 6.1: Mach number v/s the axial length of the intake module.

6.1.1.2 Conclusions:

1. The flow undergoes supersonic compression as it passes through a series of oblique shocks.
2. The absence of a back pressure at the subsonic inlet is the reason for the normal shock not being formed and the flow being supersonic again.

6.1.2 Pressure variation along the intake:

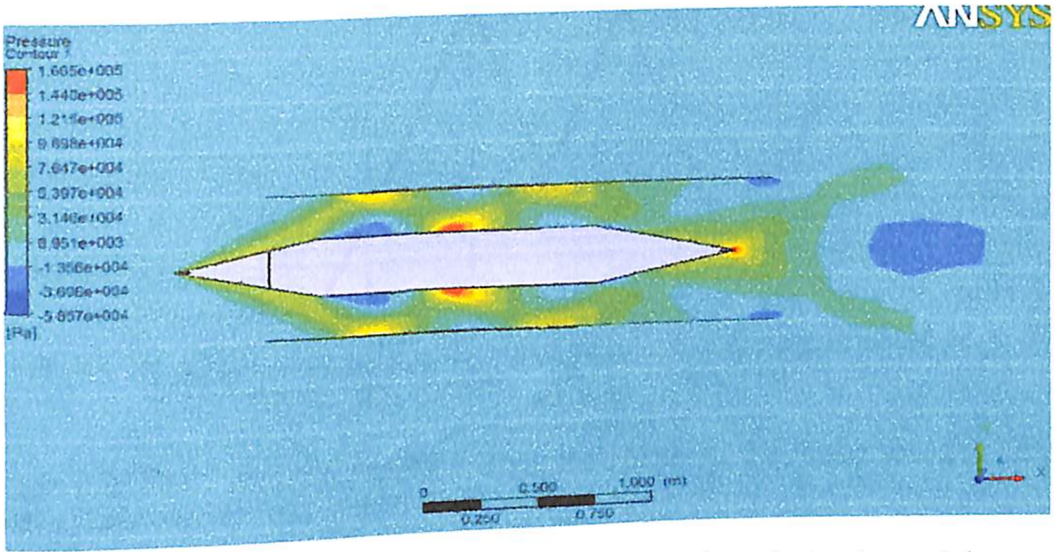


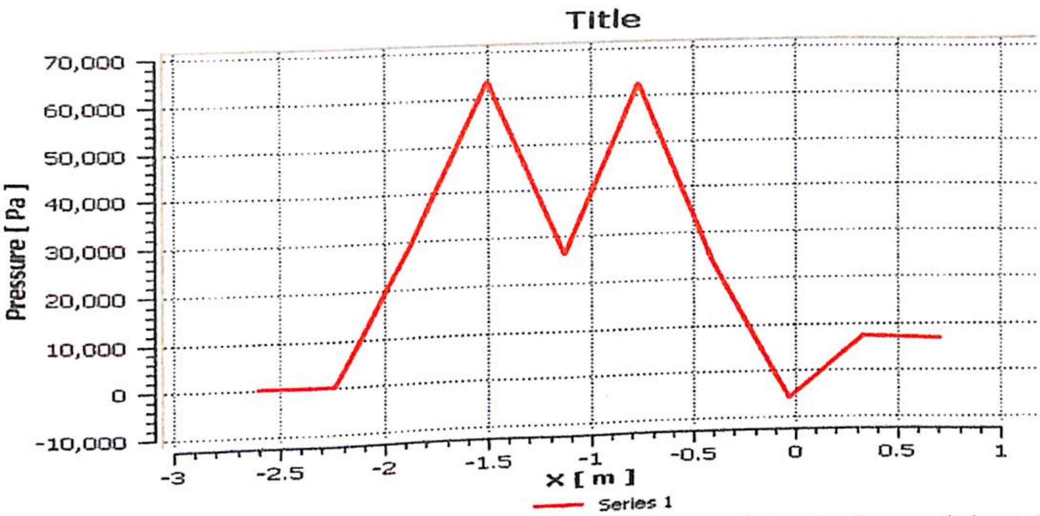
Fig6.3: Static pressure changes through visualization along the intake module.

6.1.2.1 Observations

1. The static pressure increases across the oblique shock.
2. Since Mach number is decreasing, the static pressure is increasing.
3. The pressure increases and decreases along the length of the intake.

6.1.2.2 Conclusions

1. The dynamic pressure of the flow gets converted into static pressure as it passes through an oblique shock.
2. Due to the reflection of shocks there are certain regions in the flow where the flow expands due to which the pressure decreases.



Graph 6.2: Static pressure variation along the length of the intake module, taking 1atm as reference.

6.1.2 Entire Ramjet analysis

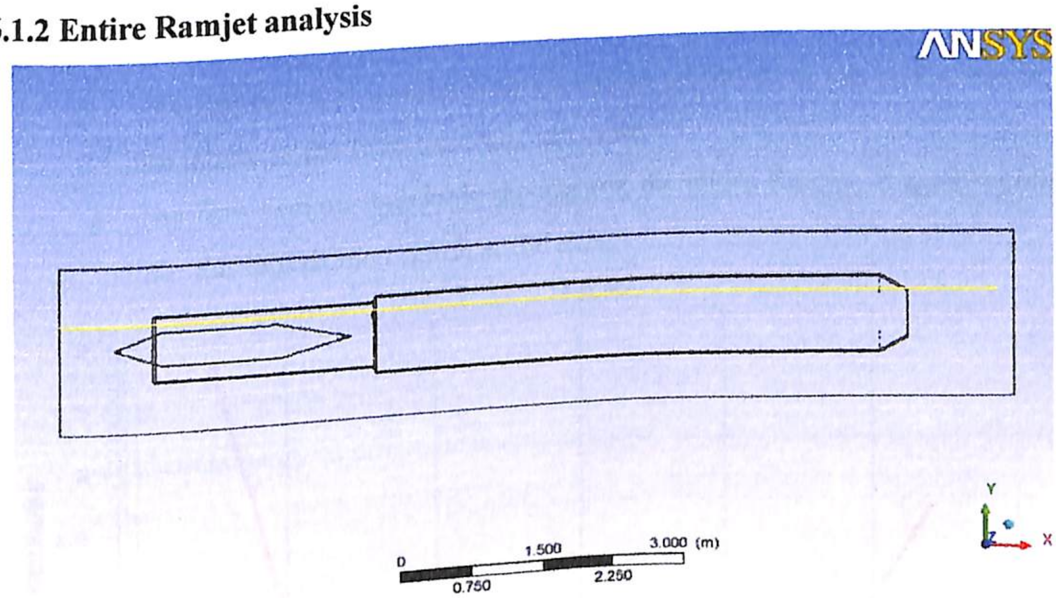


Fig 6.4: Yellow line depicts the direction and line along which the parameters have been plotted.

The numerical analysis of the entire ramjet has also been done. Heat addition was not considered since only the study of flow had to be done. The yellow line in the above shows the direction and the location where the parameters have been in the graphs below.

6.1.2.1 Mach number variation

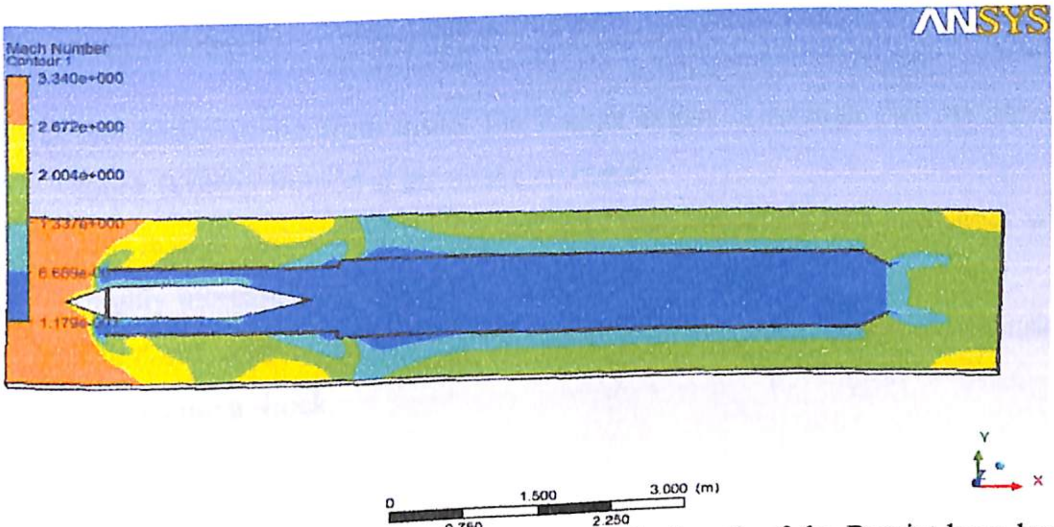
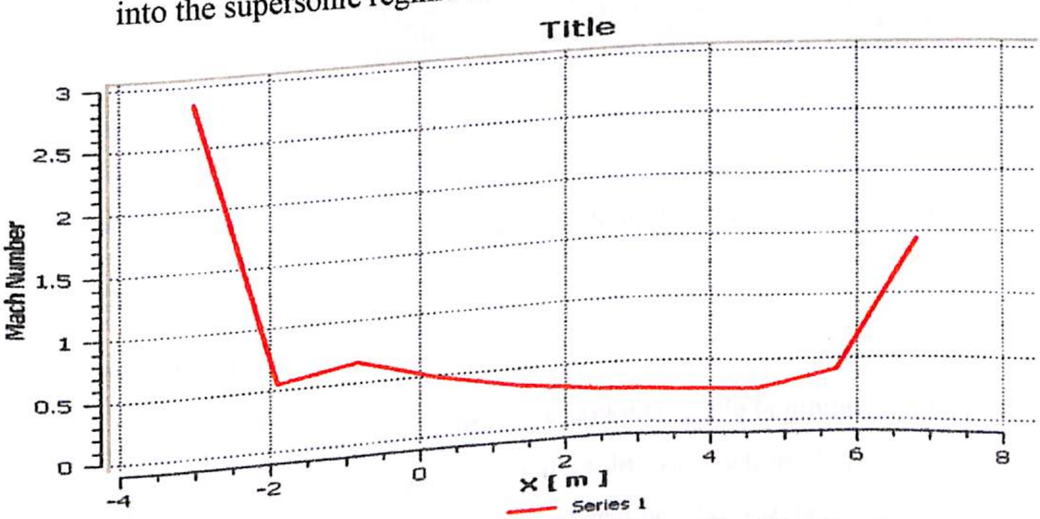


Fig 6.7: The variation of mach numbers along the length of the Ramjet have been depicted by the means of different colours

6.2.1.1Observations:

1. The formation of a normal shock can be observed at the very entrance of the intake as the flow becomes subsonic.
2. The flow remains subsonic throughout the entire Ramjet. It against comes into the supersonic regime at the exit.



Graph 6.3 Variation of the Mach number along the length of the Ramjet.

3. It can be seen from the chart that the flow becomes subsonic at the very beginning of the intake which validate the formation of normal shock at the beginning of the intake.

6.2.1.2Conclusions:

1. The backpressure from inside the Ramjet engine is so high that the normal shock is being formed at the intake entrance.
2. This is undesirable since the formation of what appears to be a bow shock will highly increase the force acting upon the entire engine.
3. The dump zone in the combustion chamber on not being cowled from outside is creating a shock.

6.2.2 Pressure variation:

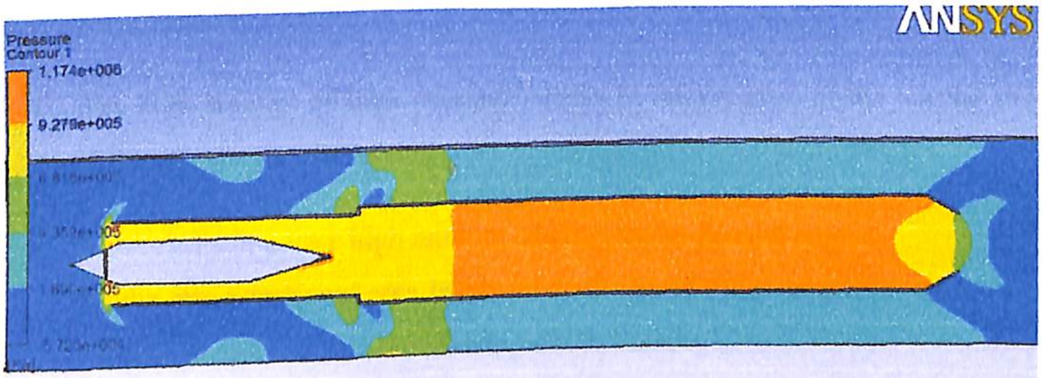
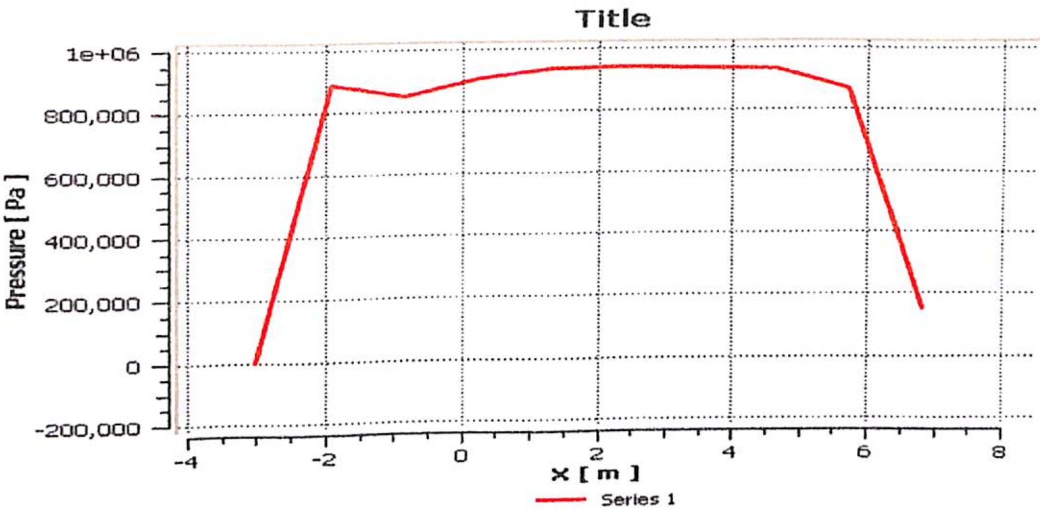


Fig 6.9: variation of static pressure along the length of the ramjet depicted through colour contours

6.2.2.1 Observations:

1. There is a sudden increase in the static pressure at the beginning of the intake.
2. The pressure further increases as it enters the combustion chamber.
3. There is a decrease in pressure as the air reaches the nozzle exit.



Graph 6.4 Variation of static pressure along the length of the entire Ramjet.

6.2.2.2 Conclusions

1. The high amount of back pressure results in the creation of the normal shock at the entrance of the intake.
2. This results in a very high amount of drag on the Ramjet engine.
3. As the cross sectional area increases in the combustion chamber, the pressure further increases due to the slowing of the flow.
4. Even though there is a large increase in the static pressure at the beginning, the immense amount of drag being created on the engine will make it inefficient.
5. Hence, the role of back pressure is very important in the case of ramjet engines as they control the formation of the terminal shock.

Chapter 7

Difficulties faced

7.1 Specification of back pressure

The flow has to become subsonic which is only possible by the presence of a normal shock. CFX does not provide a method to specify the backpressure at the outflow region in a supersonic flow. Due to this the flow undergoes a centered expansion process and becomes supersonic again in the subsonic diffuser part of the intake (as shown in the analysis in previous chapter).

The formation of normal shock in the supersonic intake plays a vital role in the compression process. It plays a very important role not only in the formation of normal shock but also in its location. If the back pressure is too high, the normal shock will be pushed out hence increasing the total pressure head loss. On the other hand if the backpressure is too less, the normal shock will be pulled down into the diffuser region which also ends

The back pressure in the intake is controlled by the amount of heat addition in the combustion chamber. For a general heat addition of 10MW, there is a creation of a backpressure of the order of 300KPa.

7.2 Insuffient resources

The calculations and simulations were done on a 1GB RAM computer. This amount of memory was insufficient for the calculation with complex boundary conditions. Hence the second phase of the project that is the complete simulation with heat addition and its performance evaluation could not be achieved in this project.

Conclusion& Future scope

The intake of the Ramjet engine plays a very vital role in the performance of the entire part. The basic idea of the design is to obtain maximum pressure recovery. The position of terminal shock plays a very important role in the compression process.

The backpressure created in the combustion chamber is the main parameter controlling the position of the normal shock. Hence it has to be maintained properly using various methods. The heat addition in the combustion chamber controls the backpressure.

The further phase of the project which consisted of the simulation of the complete Ramjet and the performance evaluation could not be performed due to insufficient resources such as processing capabilities. The use of mathematical processors would ease the process and hence increase the scope of work.

The further analysis of the flow could be done by analysis along with consideration of heat addition, which would determine the optimum conditions for the position of the normal shock. Hence the accurate design and performance evaluation can be done.

References

Akbarzadeh.M and Kermani. M. J, Numerical Simulation of Inviscid Airflows in Ramjet Inlets.

Francois Falempin, MBDA France, "Propulsion system for hypersonic flight".

Paul J. Waltrup, Michael E. White, "History of Ramjet and Scramjet Propulsion Development for U.S navy missiles".

Ratakrishnan. E, Gas Dynamics.

Thangadurai G.R.S, Subhash Chandran B.S., Babu. Vand Sundararajan.T, Numerical Analysis of Integrated Liquid Ramjet Engine.

Yadav R, "Steam and Gas turbines and Powerplant Engineering", Central Publishing House, Allahbad.

www.adl.gatech.edu/classes/propulsion/prop6.html, Ideal Ramjet Cycle.

www.aerospaceweb.org

Appendix-I

Matlab code for the calculation of shock angle beta for a particular M_1 and θ:

```
clc
clear all

%-------------- taking inputs-----------------

M1 = input ('enter inlet mach number:');      % inlet mach
number
A = input ('enter wedge angle(half):');      % wedge angle
Beta = input ('enter nearby value:');          % minimum
beta value

%--------------- loop to find beta---------------

for (i=0:0.00001:1000)
    theta = (2*pi*(A/360));                       % conversion to
radians
    beta =(2*pi*(Beta/360));
    a = tan(theta)
    B=
2*cot(beta)*(((M1^2)*((sin(beta))^2)1)/(M1^2*(1.4+cos(2*b
eta))+2))
    Beta = Beta+i
        if B-a >= 0.00001;
        break
    end
end
```