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Preliminary Design of a Ramjet Engine: An Analytical Approach

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Abstract: This work is based on preliminary design of Ramjet engine. Ramjet engine is simplest type of gas turbine engine used, which consists of non-moving parts for its operation. It is mainly used for power generation at supersonic speeds. This type of engine is mostly used in missiles, with a few applications in aircrafts. The present approach to preliminary design was based on mathematical equations considering ideal conditions. For the design, aero-thermodynamic equations were used starting with intake and followed by diffuser, combustor and nozzle. For the design purpose, initially Mach 2 and thrust of 10 kN was selected as desired condition. Subsequently, it was analyzed for the varying Mach numbers starting from Mach 1.5 to Mach 4 at desired thrust of 10 kN. Also, the design was analyzed for varying thrust from 6 kN to 22 kN at desired Mach number of 2. The results obtained have been reported and effects of variation of parameters have been represented graphically. The graphs were obtained using GNU Octave 6.4.0. The design achieved could be used for further steps of CAD model generation and subsequent analysis for the required purpose. It can serve as a base for detailed design of the engine.

Keywords: Ramjet, aerodynamics, thermodynamics, pressure, temperature

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1. Introduction

Ramjet engine is simplest type of gas-turbine engine used, which consists of non-moving parts for its operation. It is mainly used for power generation at supersonic speeds. This type of engine is mostly used in missiles, with a few applications in aircrafts.

Ramjet engines do not employ rotating turbomachinery to compress inlet air prior to fuel addition and combustion; rather, they compress the incoming air through supersonic inlets and diffusers, where compression is achieved by intricate shock structures in the flow. They are only able to fly at speeds beyond the local speed of sound because they only employ this type of compression. A supersonic combustion ramjet is called scramjet. The lure of ramjets and scramjets is that they offer high Mach number flight, but without the need to carry oxidizer onboard the flight vehicle as in rockets. This results in a higher specific impulse, Isp, which is a general measure of system efficiency.

In a typical ramjet engine (shown in Figure 1), supersonic air is ingested by the inlet diffuser and compressed through a series of oblique shocks followed by a normal shock, which, before the airflow reaches the combustor, reduces its speed to subsonic levels. A diverging duct is frequently used to further slow the flow down before it enters the combustor. Fuel is then injected in the combustor where it mixes with the subsonic air and burns to add energy to the airflow [1].

The burned air-fuel mixture is then ejected through a converging-diverging nozzle where it is accelerated to a supersonic velocity out the aft end of the engine (Heiser & Pratt, 1994) [2]. Figure 2 shows a typical ramjet engine designed by NACA [3]. The static thrust of these types of engine are zero, therefore it require assisted take-off for its operation like rocket assist.

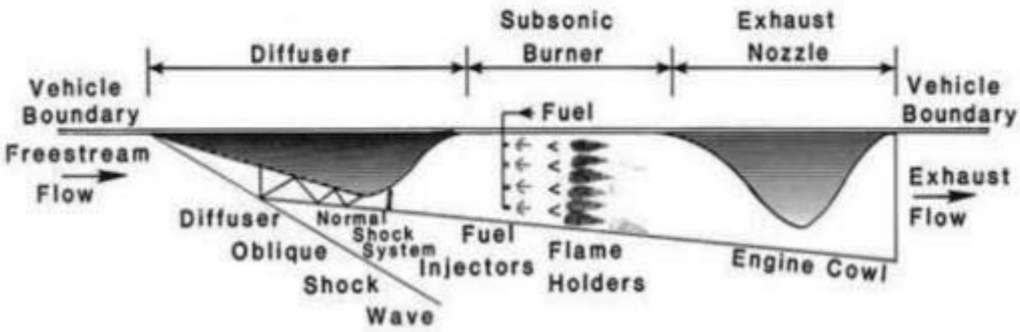


Figure. 1. A typical two-dimensional ramjet engine [1]

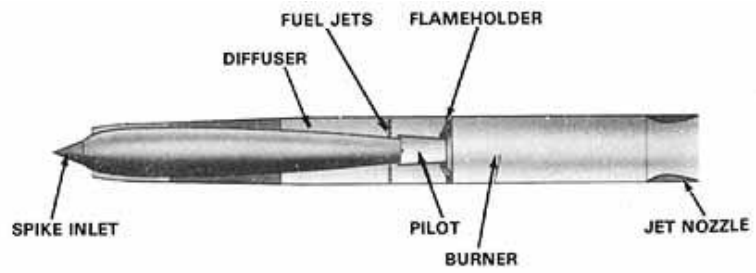


Figure. 2. A typical Ramjet engine (NACA) [3**]**

Falempin [4] did the study on Ramjet and dual mode operation. He studied the possibilty of combining air-breathing and rocket engines, for increasing the efficiency of supersonic flights. He tried to use this capability in space launching operations.

In the past, a large number of studies were performed [1, 2] on the design of different parts of a Ramjet engine. Many studies have been performed to improve the design of intake of a ramjet. Cain [5] has done the study of Ramjet intake. He did the study and understood the fundamentals and limitations of ram compression. He discussed the historic supersonic intakes and tried to point out the problems occured in the engines designed in past and possible solutions to overcome those problems. For the purpose, he did study of one and two dimensional flow. Watanabe et al [6] discussed the significance of ramp for controlling the flow at supersonic intake. For the purpose, they carried out the wind tunnel testing of supersonic inlet. They studied the concept to control buzz at the inlet. Understanding the buzz phenomena can be helpful to design an efficient intake [7]. Alekhya et al [8] did the design and performance analysis of intakes using CFD.They performed the analysis on both the cone inlet and ramp inlet and delineated the differences. Humphrey [9] did the study on design and fabrication of a Ramjet inlet as his bachelors thesis. For the design he used single ramp and single sided three shock internal compression. He used the inlet to decelerate the airflow from supersonic Mach 3.3 to subsonic Mach 0.54 using shock waves. He found the optimum ramp angle for the intake wedge by making a initial guess and then performing iterations to match the thermodynamic parameters with desired value. Idris et al [10] did the cold-flow study to analyze experimentally the inlet of a scramjet engine at mach number 5. They used pressure-sensitive paint (PSP) to investigate the flow characteristics on the compression ramp.They used luminescent measurement system to capture various shock-shock interaction, corner and shoulder separation regions and shock trains to analyze the flow behaviour. They showed the improved performance of scramjet inlet-isolator when operated in a modest angle of attack. Alekhya et al [11] did the study for the performance comparison of conic and ramp inlet using CFD. The total pressure recovery was used as the main parameter to compare the performace of two types of inlets. As per their investigation ramp inlet was found as the best total pressure recovery index. Sarout et al [12] highlighted the significant points for the performance enhancement of supersonic air intake model through the implementation of pressure feedback system. They also studied the hypersonic behaviour at Mach 5. They investigated the performance improvement of the intake system using three different pressure feedback configurations. Improvement in total pressure was accomplished by removing the separation bubble to great extent. Improvement in total pressure recovery depicted the better performance which happened due to implementation of pressure feedback system.

Some studies were also performed on the design of combustion chamber of a ramjet. Paul Cameron Stone [13] did the study on Ramjet combustion chamber as his bachelor thesis. He designed the engine combustion chamber using analytical approach and made a physical model. Subsequently, he studied the baseline performance of Ramjet engine. He studied the ramjet operation for the engine operating at Mach 3.3. Ebrahimi [14] performed CFD study using three-dimensional up-wind-differenced finite volume solver to evaluate combustor and nozzle flows. He used two kinetic models in which first was single-step reaction model and second was involving eight chemical reactions. Cohen and Guile [15, 16] performed a classical experiment for free jet with mixing combustion. Micka et al [17] performed study on combustion characteristics of a dual-mode scramjet combustor with cavity flame holder.

Similarly, there have been many studies performed on the design of nozzle of a Ramjet engine. Khan et al [18] presented the idea of theory of characteristics for the design of a two-dimensional supersonic nozzle. They calculated the minimum length of supersonic nozzle for the optimum mach number by developing a MATLAB program. The flow holds the consistency in converging section, so design was focused on diverging section. The design approach for the convergent-divergent nozzle was advantageous to avoid shock formation in the diverging section corresponding to supersonic mach number. Karthik et al [19] presented the design of a ramjet engine with modified nozzle and flame stabilizer working under subsonic condition. They also designed the associated systems like liquid fuel injector and ignition system.The air inlet was enabled by free jet facility.To compare the performance of Ramjet, they operated the engine with two types of fuel, Biofuel and Jetfuel. They performed the experiments for varying inlet velocities starting from ~13 m/s to a maximum of 140 m/s. The thrust obtained by Aviation Turbine Fuel (ATF) was higher only at air inlet velocity of ~19 m/s , during all other conditions biofuel performance was found to be better. Valli et al [20] did the design and thermal analysis of Ramjet engine for supersonic speeds. They designed the ramjet engine‘s convergent-divergent nozzle for different supersonic speeds of Mach 1.6, 1.8 and 2.0. From their analysis, it was found that as the velocity and mass flow rate of air increases before the diffuser, the speed of jet also increases.

However, there is a paucity of literatures giving the insight on analytical approach to the design of a Ramjet engine for its all parts collectively. Moreover, although the literatures available for design of the parts of Ramjet engine have demonstrated the method to calculate aero-thermodynamic parameters in different parts but way to calculate dimensions of the parts have not been clearly delineated. There had been many design of Ramjet engines in the past, but their design features was not disclosed due to security or industrial reasons [21]. The present study was focused on analytical approach for the design of Ramjet engine using aero-thermodynamic equations and subsequently the dimensions of different parts of the engine have been calculated.

2. Materials and Method

The primary aim of this study was to obtain a preliminary design of a Ram- jet engine. For the design, directly approaching towards experimental techniques may be a tedious task, time consuming and subsequently it may lead to high design cost. So, for the present approach design was based on aero-thermodynamic effects.

The preliminary design of main components of ramjet engine i.e. intake, diffuser, combustor and nozzle were obtained using aero-thermodynamic equations (Fig. 3). For the design of Ramjet engine in the present study, optimal expansion condition was assumed in convergent-divergent nozzle.

For the design, atmospheric conditions were assumed as sea-level conditions. So, atmospheric parameters at inlet were as follows:

Temperature, T_1 = 288.15 K
Pressure, P_1 = 1 atm = 101325 Pa
Density, ρ_1 = 1.225 kg/m^3
Gas constant, R = 287 J/kg-K
Specific heat at constant pressure, C_p = 1005 J/kg-K
Specific heat at constant volume, C_v = 718 J/kg-K
Isentropic expansion factor, γ = 1.4

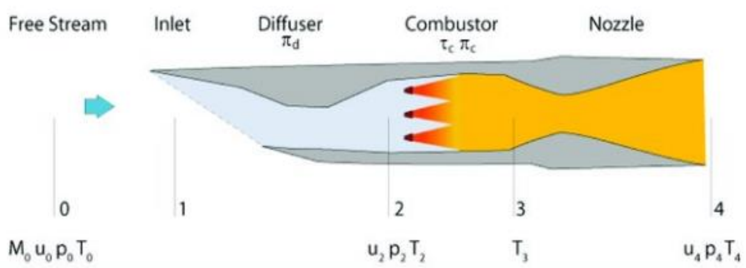


Figure 3. Thermodynamic stations of Ramjet [22]

The stagnation pressure and stagnation temperature at inlet for free-stream conditions could be obtained by using isentropic relations, for the free stream Mach number. In general, for an adiabatic flow field, the stagnation temperature is defined by the relationship

$$T_0/T = 1 + (\gamma-1)/2 \times M^2 \quad (1)$$

where, T_0 = Total temperature or stagnation temperature
T = Static temperature
M = Mach number
The stagnation pressure could be found from the above relation as follows :

$$\left(\frac{P_0}{P}\right) = \left(\frac{T_0}{T}\right)^{\frac{\gamma}{\gamma-1}} \tag{2}$$

Where, P_0 = Stagnation pressure
P = Static pressure

2.1 Design of Intake

In the intake, reduction of velocity was obtained by shock waves. A single normal shock can lead to a large pressure loss. So, it is always advantageous to get the required deceleration by using oblique shocks followed by terminal normal shock. From the "Elements of Gas dynamics" by Liepmann and Roshko [23], if the compression is obtained by a large number of weak waves, the entropy increase can be reduced significantly, compared to a single shock producing the same net deflection. The entropy increase is inversely proportional to square of no of deflections. Thus, for infinite no of compressions, entropy increase will be negligible and the compression will be isentropic. The pressure change is directly proportional to angle of deflection while entropy is proportional to cube of angle of deflection [23].

$$\Delta p \sim \Delta\theta$$
$$\Delta s \sim (\Delta\theta)^3 \tag{3}$$

If there are 'n' segments in the complete turn,

$$\theta = n\Delta\theta \tag{4}$$

Then, the overall pressure and entropy changes are

$$p_k - p_1 \sim n\Delta\theta \sim \theta$$
$$s_k - s_1 \sim n(\Delta\theta)^3 \sim n(\Delta\theta)(\Delta\theta)^2 \sim \theta(\Delta\theta)^2 \tag{5}$$

So, no. of reflections could be determined by taking the consideration of changes in pressure and entropy. But designing an intake with large number of variations of angles could be a cumbersome task. So, most of the literatures have suggested to use two oblique shocks [9]. So, for the current approach two oblique shocks followed by a terminal normal shock was considered (Figure 4).

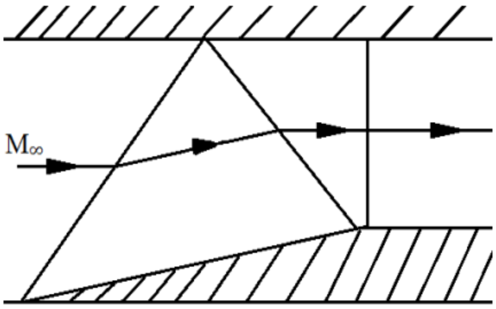


Figure 4. Three shock compression [9]

To get the required shocks, optimum ramp angles are required to be provided at the inlet face of the engine. The optimum ramp angle should be selected in a way that all shocks need to be of equal strength.

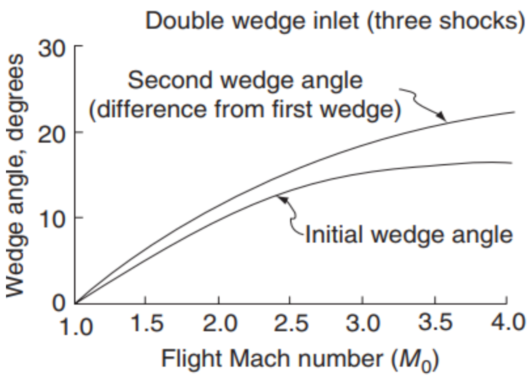


Figure 5. Optimum ramp angle for double oblique shock compression [24]

Figure 5 shows the optimum ramp angle for first and second ramp for double oblique shock compression, corresponding to different Mach numbers. For double ramp intake, at Mach 2, first wedge angle will be 10 degrees and second wedge angle will be 11.7 degrees (Figure 5). The two wedges will lead to production of two oblique shocks followed by a normal shock. The thermodynamic parameters like temperature, pressure and area relations could be determined with help of normal shock and oblique shock equations. To determine the properties normal shock tables, oblique shock tables or equations could be used. In the current approach post shock parameters were determined with the help of shock equations and calculations were performed with the help of GNU Octave 6.4.0. Wave angle β, could be find from the oblique shock relation of wave angle β, deflection angle θ and initial Mach number M_1 as follows:

$$tan\theta = 2\cot(\beta)\frac{M_1^2\sin^2(\beta)-1}{M_1^2(\Upsilon+\cos(2\beta))+2} \tag{6}$$

The desired wave angle was found from the above equation using MATLAB script through GNU Octave 6.4.0. This equation assumed a weak oblique shock, since the flow was still supersonic. To solve for air properties after the oblique shock, the normal component of the initial Mach number was required. It was calculated with the following equation,

$$M_{n,1} = M_1\sin(\beta) \tag{7}$$

Where, $M_{n,1}$ is the normal component of the initial Mach number. Using this value, the air properties after the oblique shock were calculated as follows,

$$M_{n,2}^2 = \frac{1+\left(\frac{\Upsilon-1}{2}\right)M_{n,1}^2}{\Upsilon M_{n,1}^2-\left(\frac{\Upsilon-1}{2}\right)} \tag{8}$$

$$\frac{\rho_2}{\rho_1} = \frac{(\Upsilon+1)M_{n,1}^2}{2+(\Upsilon-1)M_{n,1}^2} \tag{9}$$

$$\frac{p_2}{p_1} = 1+\frac{2\Upsilon}{\Upsilon+1}\left(M_{n,1}^2-1\right) \tag{10}$$

Where $M_{n,2}$ is the normal component of the Mach number after the oblique shock, and ρ_2, p_2 are the density and pressure after the oblique shock respectively. Using the outputs from eqs. (8), (9) and (10), the Mach number and temperature ratio after the oblique shock were calculated.

$$M_2 = \frac{M_{n,2}}{\sin(\beta-\theta)} \tag{11}$$

$$\frac{T_2}{T_1} = \frac{p_2}{p_1}\frac{\rho_1}{\rho_2} \tag{12}$$

Where M_2 is the Mach number after the oblique shock and T_2 is the temperature after the oblique shock.

To solve for the stagnation properties, the change in entropy across the shock was required. The equation below was used to do this.

$$s_2 - s_1 = c_p ln\frac{T_2}{T_1} - Rln\frac{p_2}{p_1} \tag{13}$$

Where s is entropy, R is the gas constant, and c_p is specific heat at constant pressure.

The change in entropy from eq. (13) was used to find the ratio of the stagnation pressures before and after the oblique shock.

$$\frac{p_{02}}{p_{01}} = e^{-\Delta sR} \tag{14}$$

The previous equations (6) to (14), for calculating air properties after the first oblique shock were used again for the second oblique shock reflected from the second wedge.

For the third and final shock, Eqs. (8), (9), (10), and (12) were used once again with the most recent normal Mach number for the inputted normal Mach number. The wave angle β would not exist in this case since the shock was completely normal. In this final shock, the air flow was made subsonic and the final values for the pressure, temperature, and Mach number were calculated.

But the Mach number achieved after the normal shock, still may be high for combustion to take place. For better combustion, a Mach number at entry of combustion chamber should be in the range of Mach 0.2 - 0.4. So, a diffuser is required to be placed to reduce the velocity up to desired Mach number. In the present approach, velocity was reduced up to Mach 0.2 at diffuser exit or at entry of combustion chamber.

2.2 Design of Diffuser

For the design of diffuser, after the intake, area was increased in the upstream flow direction to reduce the velocity up to Mach 0.2 at combustor entry. The diffuser was assumed as operating in the ideal condition. So, isentropic conditions were assumed in the diffuser. For calculating the area and thermodynamic properties, isentropic relations were used. For calculating the properties isentropic flow tables or equations could be used. In the present study, calculations were performed with the help of equations and for the purpose GNU Octave was used.

The area ratio used for isentropic condition was as follows:

$$\frac{A}{A^*} = \left(\frac{\Upsilon+1}{2}\right)^{\frac{-(\Upsilon+1)}{2(\Upsilon-1)}} \frac{\left(1+\frac{\Upsilon-1}{2}M^2\right)^{\frac{\Upsilon+1}{2(\Upsilon-1)}}}{M} \tag{15}$$

Stagnation pressure and stagnation temperature remained constant throughout the diffuser due to isentropic conditions. The static temperature and static pressure at exit of diffuser was found from eqs. (1) and (2). Here, the diffuser was of diverging section.

2.3 Design of combustion chamber

In the present study, the combustor was assumed to be operating in the ideal condition. The Mach number at the entry of combustion chamber was taken as Mach 0.2. In the combustion chamber, fuel is get added to the flowing air and ignition takes place which increases the temperature. It is the highest temperature region of the engine.

The mass flow rate and temperature at exit of combustion chamber could be found from the mass and energy balance equations (Figure 6). The mass flow rate and temperature may vary depending on the type of fuel being used. In the present study, the fuel used for the analysis was Octane, C_8H_{18}, which is the commonly used hydrocarbon fuel.

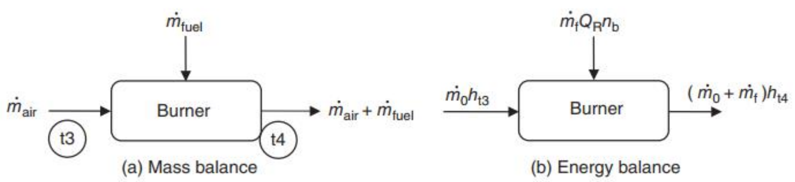


Figure 6. Block diagram of combustor with mass and energy balance

The combustion equation of an arbitrary hydrocarbon with simplified air can be represented as

$$C_xH_y + a\,(O_2 + 3.76\,N_2) \longrightarrow b\,CO_2 + cCO + dH_2O + eH_2 + fO_2 + 3.76\,aN_2 \tag{16}$$

Here, the simplified air could be defined as the combination having 21 percent of oxygen and 79 percent of nitrogen. For lean or stoichiometric conditions, the above equation simplifies to

$$C_xH_y + a\,(O_2 + 3.76\,N_2) \longrightarrow b\,CO_2 + dH_2O + fO_2 + 3.76\,aN_2 \tag{17}$$

The stoichiometric quantity of oxidizer is just that amount needed to completely burn a quantity of fuel. The combination is referred to as fuel lean, or simply lean, if more oxidizer is added than is necessary to achieve stoichiometry. By using the above equations stoichiometric fuel-air ratio can be determined. In the present case for octane, the stoichiometric fuel-air ratio was approximately 1:15.

To maintain the required fuel to air ratio for stoichiometry, the fuel pressure and fuel flow to the fuel injector must be high enough. But fuel flow should not exceed the stoichiometric range or else the flow will be saturated with fuel, or it will become fuel rich, to the point that the flow will not ignite.

In an ideal condition, the fuel injector atomizes the fuel flow to improve the fuel-air mixture inside the engine. In most cases, a flame holder is employed to ensure flame stability. To help the flame to propagate, the flame holder further slows the flow and establishes a recirculation region. A flat plate can serve as a flame holder. The flame holder shelters the flame and facilitates better fuel mixing.

To avoid excessive high temperature, avoid soot formation and to gain larger combustion time, a lean fuel-air mixture was used. For the reasons, the minimum possible equivalence ratio 0.51 [25], was used. Actual fuel-air ratio was determined by multiplying the stoichiometric fuel-air ratio with equivalence ratio, ϕ. The equivalence ratio, ϕ, is commonly used to indicate quantitatively whether a fuel-oxidizer mixture is reach, lean, or stoichiometric. The equivalence ratio is defined as

$$\Phi = \frac{(\frac{A}{F})_{stoic}}{(\frac{A}{F})} = \frac{(\frac{F}{A})}{(\frac{F}{A})_{stoic}} \tag{18}$$

From the above equation, it could be deduced that for stoichiometric mixture, ϕ equals unity. While for the fuel-rich mixture, ϕ>1 and fuel-lean mixture, ϕ<1.

Given a mass flow rate for the air travelling through the section and actual fuel to air ratio, mass flow rate for the fuel is

mass flow rate of fuel = mass flow rate of air x fuel-air ratio (19)

From there a new mass flow rate for the overall flow can be determined,

$$\dot{m}_{flow} = \dot{m}_{air} + \dot{m}_{fuel} \tag{20}$$

Dividing the mass flow rate of the fuel by the mass flow rate of the newly mixed flow, a new fuel to flow ratio is determined

$$f_{flow} = \frac{\dot{m}_{fuel}}{\dot{m}_{flow}} \tag{21}$$

Using that new ratio along with the known heat energy of the fuel,

$$Q_{flow} = Q_{fuel} * f_{flow} \tag{22}$$

the heat energy of the flow can now be determined.

For constant pressure adiabatic flame temperature, the absolute enthalpy of reactants was made equal to absolute enthalpy of products. Absolute enthalpy is the sum of an enthalpy that takes into account the energy associated with chemical bonds, the enthalpy of formation, h_f, and an enthalpy associated with temperature, the sensible enthalpy, Δh_s. Thus, we can write molar absolute enthalpy for species 'i' as

$$\overline{h_i}(T) = \overline{h^0{}_{f,i}}\left(T_{ref}\right) + \Delta\overline{h_{s,i}}\left(T_{ref}\right) \tag{23}$$

Where,

$\overline{h_i}(T) = Absolute\ enthalpy\ at\ temperature\ T$

$\overline{h^0_{f,i}}\left(T_{ref}\right) = Enthalpy\ of\ formation$

$\Delta\overline{h_{s,i}}\left(T_{ref}\right) = Sensible\ enthalpy\ change$

Mathematically, $$\overline{h_{s,i}}\left(T_{ref}\right) = \overline{h_i}\,(T) - \overline{h^0{}_{f,i}}\left(T_{ref}\right) \tag{24}$$

By using sensible enthalpy table and heat of formation table, the combustion chamber temperature was found by iterative process. The temperature at which, the enthalpy of reactants and products were found to same, was selected as the optimum temperature of combustion chamber to get better combustion without wastage of heat.

To have a complete combustion of fuel in the combustion chamber, a combustion residence time of few milliseconds is required. For octane, the residence time may be taken as 2 ms (milliseconds) [4, 26, 27]. The length of combustion chamber was determined by multiplying the residence time with flow velocity in the chamber.

As the flow in a combustion chamber takes place in constant area and heat addition takes place, the flow is Rayleigh flow in the chamber.

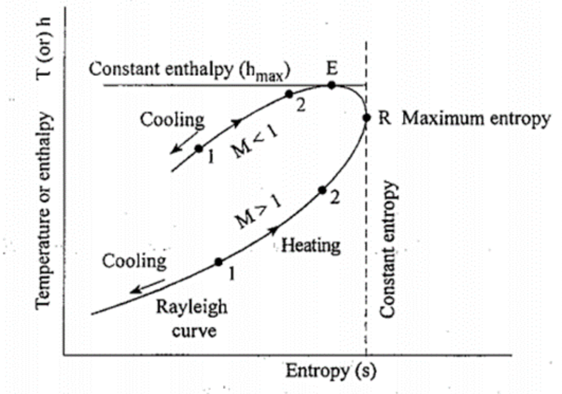


Figure 7. Rayleigh line or curve [28]

The thermodynamic parameters at exit of combustion chamber were found by Rayleigh flow equations. The combustion chamber temperature was maintained below the thermal choking temperature. The isentropic expansion factor γ, was taken as 1.3 for the air-fuel mixture in the combustion chamber and the nozzle. The Rayleigh flow equations used were as follows :

$$\frac{T_0}{T_0^*} = \frac{2M^2(1+\Upsilon)}{(1+\Upsilon M^2)^2} \times \left(1+\frac{(\Upsilon-1)}{2}M^2\right) \tag{25}$$

$$\frac{p_0}{p_0^*} = \frac{1+\Upsilon}{1+\Upsilon M^2} \times \left[\frac{2\left(1+\frac{(\Upsilon-1)}{2}M^2\right)}{\gamma+1}\right]^{\frac{\gamma}{\gamma-1}} \tag{26}$$

$$\frac{p}{p^*} = \frac{1+\Upsilon}{1+\Upsilon M^2} \tag{27}$$

$$\frac{T}{T^*} = M^2 \times \left[\frac{1+\Upsilon}{1+\Upsilon M^2}\right]^2 \tag{28}$$

Where, T_0 = Stagnation temperature
T_0^* = Critical stagnation temperature or choking temperature
P_0 = Stagnation pressure
P_0^* = Critical stagnation pressure
P = Static pressure
T = Static temperature

2.4 Design of Nozzle

After the combustion chamber, nozzle was used to increase the velocity from subsonic speed at exit of combustor to supersonic speed at exit of nozzle. The nozzle expels the exhaust at an accelerated rate to produce thrust. To increase the subsonic speed at combustor exit to supersonic speed at exit of engine, convergent-divergent nozzle was used. The flow in the nozzle was assumed to be ideal. The losses were neglected and efficiency was considered as 1. To calculate the parameters at throat and exit of nozzle, isentropic flow conditions were used.

For the present study, nozzle was designed as optimal expansion condition at sea level. To calculate the properties, isentropic flow tables or equations could be used. In the present study, the isentropic flow equations were used and calculations were performed with the help of GNU Octave 6.4.0.

As the flow is isentropic throughout the nozzle, stagnation pressure and stagnation temperature will remain constant. Static temperature and static pressure at throat and exit of nozzle were obtained by using equations (1) and (2). Exit Mach number was obtained from pressure relation of equation (2). Area ratio was determined with the help of equation (15). Mass flow rate of air and inlet area was calculated from desired thrust. Thrust and mass - flow rate relation is as follows:

$$\text{Thrust} = \text{mass flow rate of air} * (V_{exit} - V_{inlet}) \tag{29}$$

The above procedure described could be used to design a simple conical nozzle. But simple conical nozzle efficiency could be affected due to flow-separation in the diverging portion of the nozzle. So, to improve the performance bell nozzles could be used. Bell-shaped nozzles are currently the most commonly used type of nozzle. Here, the calculated dimensions could be same, but there would be change in nozzle profiling.

2.4.1 Design of Bell Nozzle

Nozzle profiling in this case is obtained by tailoring the pressure gradients to minimize flow separation which improves the performance of the nozzle. Bell-shaped nozzles consist of two sections. The nozzle outlines taper off further downstream after diverging at a somewhat sharp angle near the throat. The divergence angle is approximately parallel to the jet axis at the portion close to the nozzle exit. This configuration allows a maximization of the performance while minimizing the weight, with the length of the nozzle being as much as 25% lower than the conical nozzle equivalent. This kind of nozzle turns the flow to create an axial flow in order to reduce divergence. Bell nozzles typically exhibit one design point where they are at their most efficient at a given altitude of operation [22]. The bell nozzle basic geometry is provided in Figure 8, with α being the initial nozzle angle immediately downstream of the throat and θ_{ex} the final nozzle exit angle.

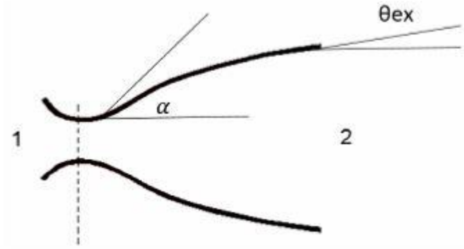


Figure 8. Bell nozzle divergence [22]

The bell-shaped nozzle design could be performed based on the theory of characteristics. The method of characteristics approach is applicable only for diverging portion, as before the throat, flow is subsonic and flow holds the consistency in the converging section [18]. The nonlinear differential equation of the velocity potential can be used to quantitatively explain the physical conditions of a two-dimensional, steady, isentropic, irrotational flow. Finding solutions to the aforementioned velocity potential using the method of characteristics is a mathematical approach that takes into account the boundary conditions and converts the governing partial differential equations (PDEs) into ordinary differential equations (ODEs). In a supersonic flow, characteristics are "lines" that are oriented in particular directions and along which disturbances (pressure waves) travel.The Method of Characteristics (MOC) is a numerical procedure appropriate for solving, among other things, two-dimensional compressible flow problems. This method allows for the calculation of flow characteristics such as direction and velocity at various locations within a flow field.

3. Results and discussion

The preliminary design of Ramjet engine was performed by using analytical equations with help of GNU octave 6.4.0. The vital parametrs were obtained for the same.

The mass flow rate of air at Mach 2 for a thrust of 10 kN and optimal expansion condition at sea-level was found to be 13.2 kg/s while the fuel flow rate was 0.45 kg/s.

The inlet area was found to be 158.2 square cm and the area at terminal of normal shock or start of diffuser was nearly 100.5 square cm. The area of combustion chamber was 290.4 square cm and the length of combustion chamber was found to be 181.9 mm. The combustion chamber maximum stagnation temperature was found to be 2350 K and the equivalence ratio of fuel in combustion chamber was 0.51, which were found to be in corroborate with available literature [29].

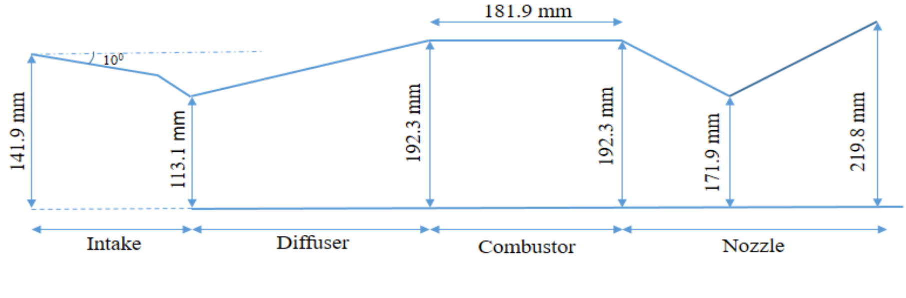


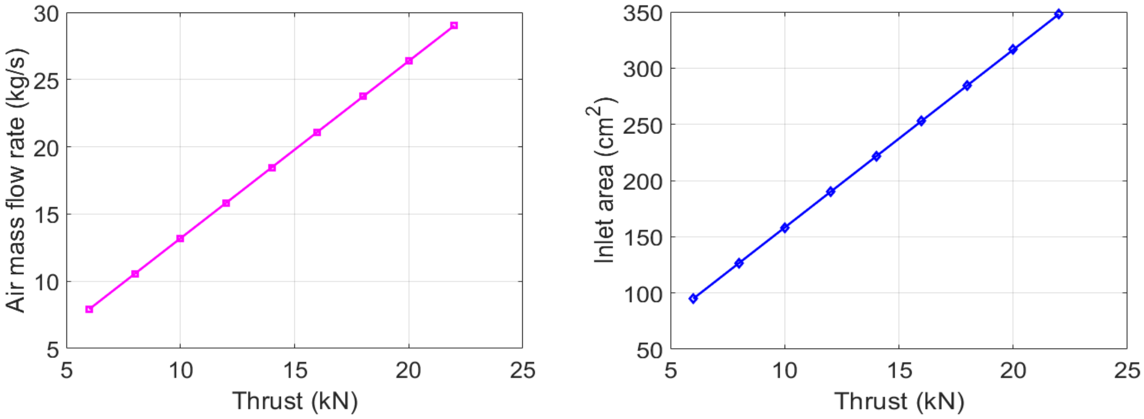
Figure 9. Diameter at different sections and length of combustion chamber of Ramjet for Mach 2 and Desired thrust of 10 kN

The area of nozzle throat and the exit area was found to be 232 square cm and 379.4 square cm respectively.

Considering circular sections, diameter at different sections was calulated from the specified areas for different sections.

Fig. 9 shows the diameter at different sections of the ramjet and length of combustion chamber of the ramjet for desired conditions of Mach 2 and thrust of 10 kN, in optimal expansion condition and for the Octane fuel.

As the thrust requirement increases keeping design Mach number constant as 2, the mass flow rate of air and subsequently mass flow rate of fuel was found to be increased as shown in Fig. 10 (a). Also, increase in thrust requirement was found to be raising the requirement of increase in the cross-section area of inlet and subsequent increase of areas at all sections as shown in Fig. 10 (b).

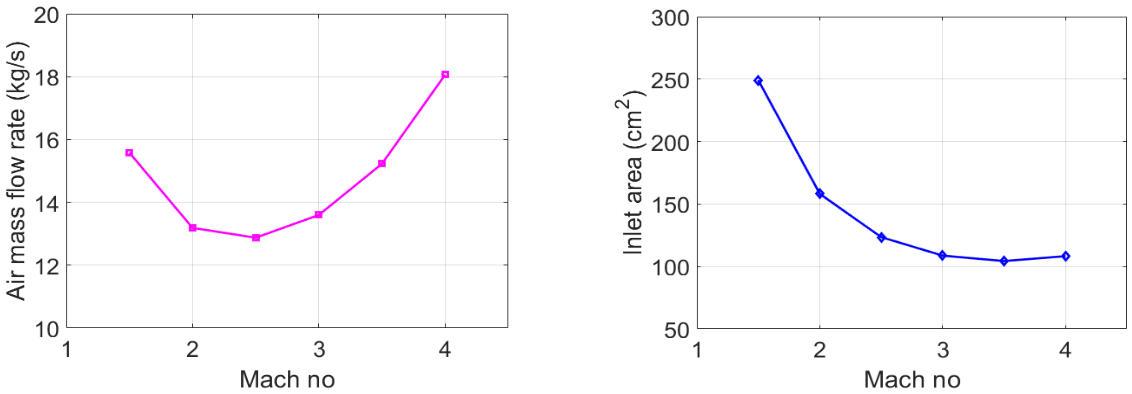


(a) (b)

Figure 10. Effect of variation of thrust on (a) Air mass flow rate (b) Inlet area

The higher thrust requirement at same mach number may arise in case of requirement of carrying higher payload or to propel a vehicle of bigger size, at same speed.

Effect of variation of Mach number on air mass flow rate and inlet area, keeping desired thrust constant as 10 KN is shown in Fig. 11. The air mass flow rate was found to be first reducing with increase of Mach number up to Mach 2.5, but with further increase in Mach number, air mass flow rate was found to be increasing (Fig. 11a). With increasing Mach number keeping desired thrust constant as 10 KN, the inlet area was found to be reducing exponentially, but at higher Mach number, there was found to be little variation in area requirements with further increase in Mach number (Fig. 11b).



(a) (b)

Figure 11. Effect of variation of Mach number on (a) Air mass flow rate (b) Inlet area

The design of the Ramjet engine in present study has been performed by preliminary approach using analytical conditions. The analytical approach has been followed considering ideal conditions, for reducing the complexity in calculations. To increase the accuracy of design, actual conditions and efficiency consideration of different components could be considered at the cost of complexity in calculations. Further, shock-boundary layer interactions effect could be considered for more accurate design.

The design of the ramjet engine may get changed by designing in under-expansion condition in case of nozzle. Most of the designs of nozzles are performed in the under-expansion condition but in the present case for the simplicity of calculations optimal expansion condition was considered.

Moreover, the design may get changed by changing the reference of atmospheric conditions or consideration of higher altitude for the design purpose. In the present study, sea-level atmospheric conditions were taken as the reference for the design.

4. Conclusion

The present study was performed for the design of a Ramjet engine using analytical approach considering ideal conditions. The current design approach may be helpful in giving an initial estimate of a ramjet engine. The current design data could be used to prepare a 3D model for numerical simulations and improvements can be made thereafter.

Thus, the analytical approach was found to be very helpful for the design of a Ramjet engine. Analytical approach can serve as a basis and provide a ground for detailed design of the Ramjet engine.

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Conflict of Interest

The authors declare no competing interests.

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