Control Of Starting Flow Mach Number To Obtain Optimum Performance In A Hypersonic Inlet

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Abstract

When pressures and temperatures become so high in supersonic flight that it is no longer efficient to slow the oncoming flow to subsonic speeds for combustion, a scramjet (supersonic combustion ramjet) is used in place of a ramjet. Currently, the transition to supersonic combustion generally occurs at a free stream Mach number around 5.0 to 6.0. This research details analysis completed towards extending scramjet operability to lower Mach numbers. The specific goal is to determine whether the scramjet starting Mach number can be lowered to Mach 3.50 or less and, if not, what the constraints are that prohibit it and what the lowest possible starting Mach number for a scramjet is with today's technology. This analysis has produced many significant insights into the current and required capabilities for overall engine design in lowering the starting Mach number; these results are presented here. The analysis has shown that a scramjet inlet with various starting Mach numbers was presented. However, a scramjet with a starting Mach number of 4.00 is possible with today's existing technology. This paper has designed the engine flow path for this case; its specifications and theoretical calculations were presented.

Keywords- *Starting Mach number, scramjet inlet supersonic combustion,*

1.Introduction

The desire for hypersonic flight within the atmosphere has motivated multiple generations of aerodynamicists, scientists and engineers. In the late 1950's and early 1960's it became clear that while rocket propulsion had the potential for access-to-space and the ability to reach many parts of the globe on ballistic trajectories, only an air breathing propulsion system could facilitate practical hypersonic flight. Antonio Ferri1 aptly described the important differences between rockets and air breathing engines as:

1) The potential specific impulse of air breathing propulsion is much larger than any chemical rocket, due to the fact it carries only fuel and not oxidizer.

2) Structural weight of an air breathing engine is larger for the same thrust than a rocket, because it must process air (oxygen and nitrogen) and have an intake, whereas the rocket has an oxidizer tank and pressurization system.

3) The thrust of an air breathing engine is a function of flight Mach number and altitude. Large thrust per unit frontal area can only be obtained in the dense atmosphere, while rockets can operate at high thrust per unit frontal area in a vacuum.

The air breathing engine cycle best suited to hypersonic flight is the supersonic combustion ramjet, or scramjet. This type of engine can be properly viewed as an extension of the very successful ramjet engine cycle, which uses shock wave compression in the inlet in lieu of the compressor in a gas-turbine engine. In a ramjet, air entering the combustor is first decelerated to subsonic speeds, where fuel is injected and burnt, and finally expanded through a second throat to a thrust nozzle. As flight speeds increase above Mach 5, reducing the air to subsonic conditions produces two problems; (1) significantly increased shock losses in the inlet, particularly at the terminal normal shock, and (2) significantly increased flow temperatures in the combustor. The second of these problems not only creates material/structural issues in the combustor, but also leads to chemical dissociation in the nozzle expansion and a consequent energy loss from the engine cycle.

Performance analysis of scramjet inlets involves the determination of the flow conditions at the inlet throat (station 2 of Fig 1). A common parameter used to quantify the efficiency of the fore body/inlet compression is the kinetic energy efficiency, η_{KE}.

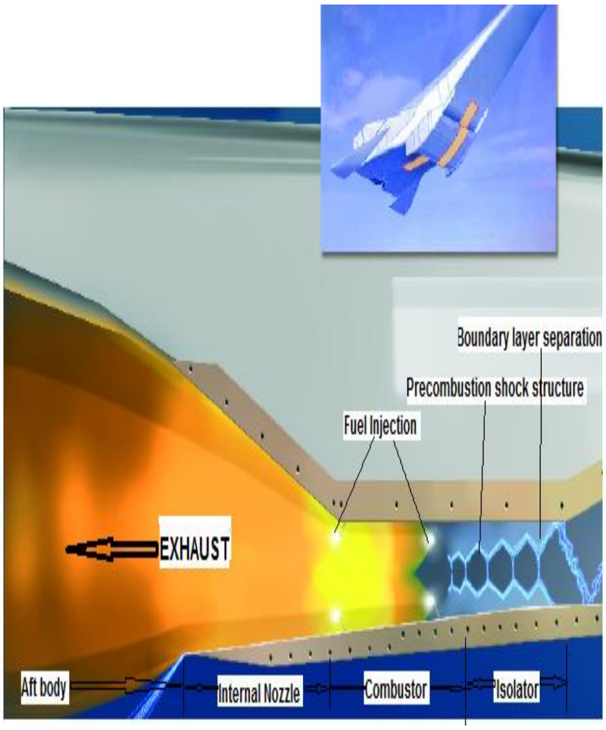

Figure 1.Scramjet Propulsion System

The five major engine components are : internal inlet, isolator, combustor, internal nozzle and the fuel supply subsystem. The internal inlet compression provides the final compression of the propulsion cycle. The fore body along with the internal inlet is designed to provide the required mass capture and aerodynamic contraction ratio at maximum inlet efficiency. The air in the captured stream tube undergoes a reduction in Mach number with an attendant increase in pressure and temperature as it passes through the system of shock waves in the fore body and internal inlet. It typically contains non-uniformities, due to oblique reflecting shock waves, which can influence the combustion process. Scramjet air induction phenomena includes vehicle bow shock and isentropic turning Mach waves, shock-boundary layer interaction, non-uniform flow conditions, and three-dimensional effects.

2. Fundamental considerations of hypersonic air breathing vehicles

1.Uninstalled thrust F : The total thrust exerted by the engine, assuming ideal flow. It equal and opposite to the difference in momentum fluxes between entering and leaving flows.

2.Specific thrust is the ratio of uninstalled thrust to the entry air mass flow rate $= F/m_o$

3. Specific fuel consumption is the ratio of fuel mass flow rate to the uninstalled thrust SFC $= m_f/F$

4. Specific impulse is of the ratio of the uninstalled thrust to the fuel weight flow rate $= I_{sp} = F/g_o m_f = 1/SFC$

5. Fuel/Air ratio is the ratio of the fuel mass flow rate to the air mass flow rate $f = m_f/m_o$

6. Stoichiometric fuel/air ratio is the value of fuel/air ratio corresponding to complete mutual combustion of all oxygen present in the air with all the reactants available in the fuel.

3. Scramjet Inlet Types

Hypersonic inlets used in scramjets fall into three-different categories, based on the type of compression that is utilized. These three types are (i) external compression, (ii) mixed compression and (iii) internal compression. In external compression all the compression is performed by flow turning in one direction by shock waves that are external to the engine. These inlet configurations have large cowl drag, as the flow entering the combustor is at a large angle relative to the free stream flow. In mixed compression compression is performed by shocks both external and internal to the engine, and the angle of the external cowl relative to the free stream can be made very small to minimize external drag. These inlets are typically longer than external compression configurations. In internal compression the compression is performed by shock waves that are internal to the engine. This type of inlet can be shorter than a mixed compression inlet, but it does not allow easy integration with a vehicle. It maintains full capture at Mach numbers lower than the design point, but its most significant limitation is that extensive variable geometry is always required for it to start.

4. Scope of the current work

However, if the necessary scramjet starting Mach number is reduced, a reduction in the number of required additional propulsion systems is possible, as the gap is bridged between the maximum possible velocity of the low speed engine(s) and the scramjet start velocity. This would have direct advantages from the resulting reduction in overall vehicle weight, the lower mass fraction required for the propulsion system (thereby resulting in more available payload weight), and fewer systems that must work in succession reliably, thereby increasing overall vehicle safety. The focus of this project is to address this issue of reducing the starting Mach number.

5. Theoretical calculations

Due to the large design impact of T_3/T_0, this subsection will present the theory and governing equations that the preliminary investigation of this parameter requires. The largest factor in changing the free stream Mach number at which supersonic combustion begins is the cycle static temperature ratio, T_3/T_0. As T_3/T_0 increases for a given free stream Mach number (M_0), the Mach number of the flow entering the combustor decreases . Thus, T_3/T_0 directly affects the M_0 at which the flow entering the burner (M_3) becomes supersonic. So, with a range of free stream Mach numbers, the necessary T_3/T_0 can be determined based on M0 and the ratio of specific heats at compression (γ_c) where M_3=1 by the following equation

$M_3 = \sqrt{2/\gamma c-1\{To/T3(1+\gamma c-1/2*M_o^2) - 1\}}$

The initial conditions were obtained as follows:
From the definition of Mach number,

$M_o = Vo/a_o$

Speed of sound, $a_o = \sqrt{\gamma R T_o}$

Normally the value of a_o = 330 m/s (approx)

$330 = \sqrt{1.4*287*To}$

Since the value of γ =1.4[Ratio of specific heats]

R = universal gas constant

$330 = \sqrt{401.8*To}$

$16.46 = \sqrt{T_o}$

$T_o = 271.06$ K

$P_o = \rho R T_o$

$= 1.23*287*271$

$P_o = 0.95$ bar

T_o =271.06 K, P_o = 0.95 bar, M_o = 4

From the reference paper the formula obtained below is of the form

$$M_3 = \sqrt{2/\gamma_c-1\{T_o/T_3(1+\gamma_c-1/2*M_o^2) - 1\}}$$

Assume the value of M_3 = 1 and γ_c =1.362

$1= \sqrt{5.525\{ To/T_3(3.896)-1\}}$

$4.64\sqrt{To/T_3} -2.35 = 1$

$\sqrt{To/T_3} = 3.35/4.64$

$To/T_3 = 0.521$

$T_3 = 519.89$ K

Compression component:

1. Stream thrust function (Sa_o)

   $Sa_o= V_o (1+RT_o/V_o^2)$

   $= 1320(1+287*271/1320^2)$

   $= 1378.92$
2. Combustor entrance temperature

   $T_3 = \varphi T_o$

   $\varphi = T_3/ T_o$

   $= 1.92$
3. Combustor entrance velocity

   $V_3 = \sqrt{V_o^2 -2CpcTo(\varphi-1)}$

   C_{pc}= 1090 J/kg K

   V_3 = 1094.93 m/s
4. Stream thrust function at combustor entrance

   $Sa_3= V_3(1+RT_3/V_3^2)$

   $= 1094.9(1+287*1002.2/(1094.9)^2)$

   $= 1231.17$
5. Ratio of combustor entrance pressure to free stream pressure

   $p_3/p_o = \{ \varphi/ \varphi(1-\eta c)+ \eta c \}^{Cpc/R}$

   Take the value of ηc as 0.9 (reference paper)

   $p_3/p_o = 8.52$
6. Ratio of combustor entrance area to free stream entrance area

   $A_3/A_o = \varphi* po/p3 *Vo/V3$

   $= 1.92*0.117*1.2$

   $= 0.271$

From the definition of Mach number,

$M_o = V_o/a_o$

$V_o = M_o* a_o$

=1980 m/s

Input conditions were taken as same as in the previous trial

To =271.06 K, Po =0.95 bar, Mo = 6

From the reference paper the formula obtained below is of the form

$$M_3 = \sqrt{2/\gamma c-1\{To/T_3(1+\gamma c-1/2*Mo^2) - 1\}}$$

Assume the value of M_3 = 1 and γ_c =1.362

$1= \sqrt{5.525\{ To/T_3(7.516)-1\}}$

$6.44\sqrt{To/T_3} -2.35 = 1$

$$\sqrt{To/T_3} = 3.35/6.44$$
$$To/T_3 = 0.2704$$
$$T_3 = 1002.2\ K$$

Compression component:

1. Stream thrust function (Sa_o)

$$Sa_o = V_o\,(1+RT_o/V_o^2)$$
$$= 1980(1+287*271/1980^2)$$
$$= 2019.28$$

2. Combustor entrance temperature

$$T_3 = \varphi\, T_o$$
$$\varphi = T_3/\,T_o$$
$$= 3.69$$

3. Combustor entrance velocity

$$V_3 = \sqrt{V_o^2 - 2CpcTo(\varphi-1)}$$
$$C_{pc} = 1090\ J/kg\ K$$
$$V_3 = 1526.8\ m/s$$

4. Stream thrust function at combustor entrance

$$Sa_3 = V3(1+RT_3/V_3^2)$$
$$= 1526.8(1+287*1002.2/(1526.8)^2)$$
$$= 1715.18$$

5. Ratio of combustor entrance pressure to free stream pressure

$$p_3/p_o = \{\ \varphi/\ \ \varphi(1-\eta c)+\ \eta c\ \}^{Cpc/R}$$

Take the value of ηc as 0.9 (reference paper)

$$p_3/p_o = 57.29$$

6. Ratio of combustor entrance area to free stream entrance area

$$A_3/A_o = \varphi * po/p3 *Vo/V_3$$
$$A_3/A_o = 3.69*0.0174*1.296$$
$$= 0.0832$$

From the calculation we came to know about the effect of Mach number by showing these two results.

S.NO	M_O	T_3 (K)	T_3/T_O	V_3 (m/s)	P_3/P_O
1	7	1314.3	4.85	1749.98	115.35
2	6	1002.2	3.69	1526.8	57.29
3	4	519.89	1.92	1094.93	8.92

Table 1.Flow parameters at station 3

From the above table we can able to understand the effect of varying Mach number and its flow parameters.

6. Inlet design

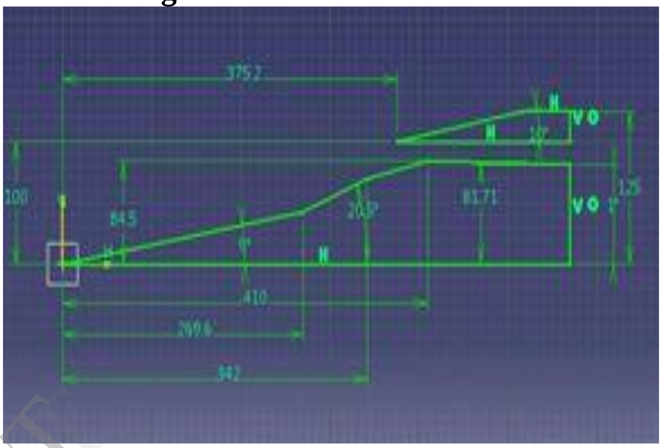

Figure 2.Inlet geometry

For the above geometry, calculations were done and it was compared to computational results. The above geometry was drawn according to the requirement of the performance of the inlet.

7.Results and Discussion

MACH NUMBER CONTOURS

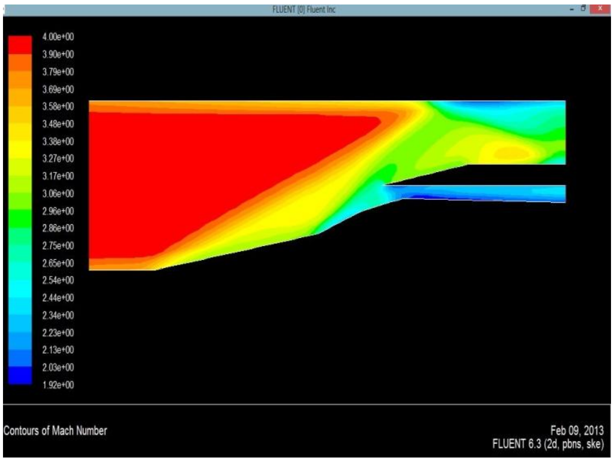

Figure 3. For M_o=4

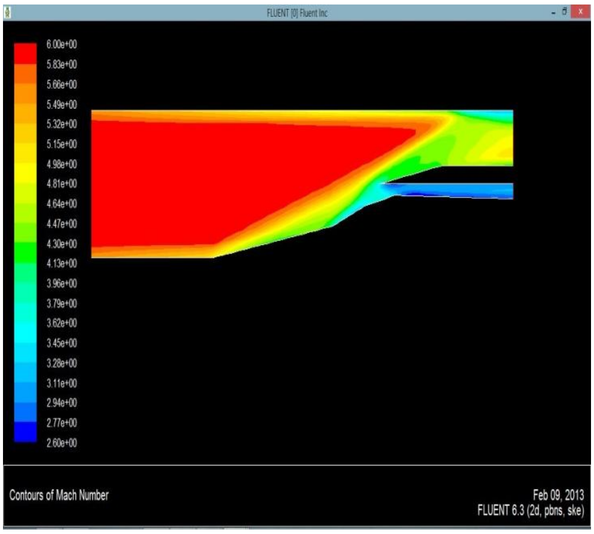

Figure 4. For M_o=6

STATIC TEMPERATURE CONTOURS

For the same conditions the plots for temperature was shown below. By seeing these contours we can able to understand the similarity between the theoretical results and computational results.

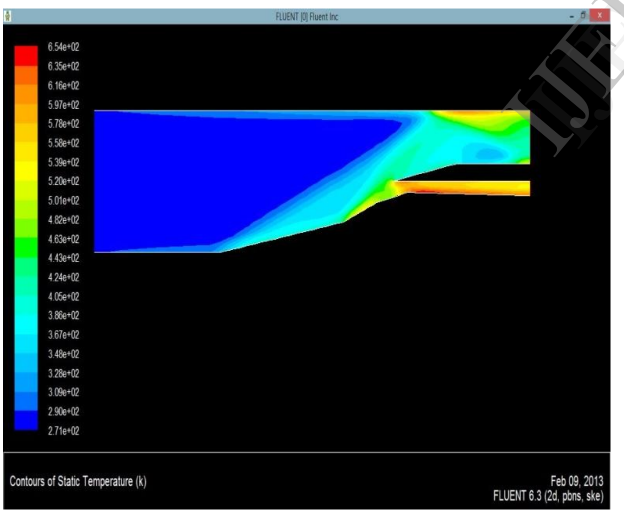

Figure 5. For M_o=4

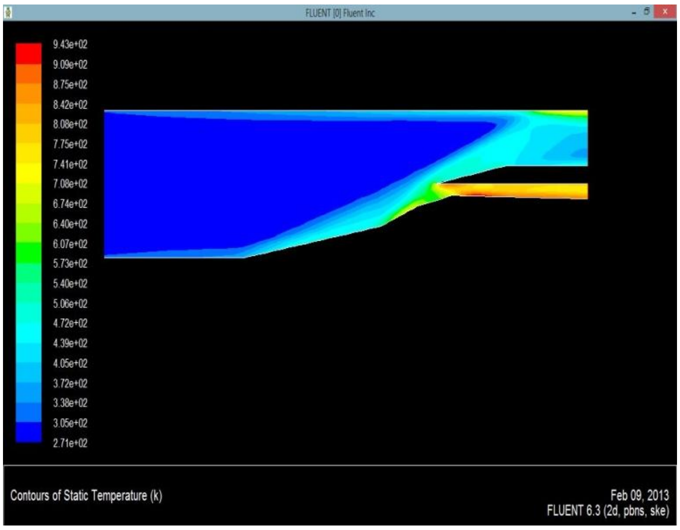

Figure 6. For M_o=6

COMPARISON

The following table shows the comparison of theoretical and computational results.

PARAMETERS AT STATION 3	THEORETICAL	COMPUTATIONAL
MACH NUMBER	4.6	4.8
STATIC TEMPERATURE (K)	1002.2	1033.8

Table 2. For M_O=6

PARAMETERS AT STATION 3	THEORETICAL	COMPUTATIONAL
MACH NUMBER	3.2	3.1
STATIC TEMPERATURE (K)	519.89	543.2

Table 3. For M_O=4

From the table values the similarity between the results can be acceptable.

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