Volume 5, Special Issue, June 2018 ISSN: 2348-8565 (Online)

International Journal of Modern Engineering and Research Technology

Website: http://www.ijmert.org Email: editor.ijmert@gmail.com

National Conference on
Advances in Mechanical Engineering and Nanotechnology (AMENT2018)
29-30 June, 2018
Organized by
Department of Mechanical Engineering, University College of Engineering (A), Osmania University, Hyderabad, TS, India

Analysis of Key Design Parameters of a Pure Scramjet Engine to Reduce its Starting Mach Number

V. Uma Maheshwar
Department of Mechanical Engineering
University College of Engineering (A),
Osmania University,
Hyderabad (T.S.) [INDIA]
Email: mahesh.v@uceou.edu

Y. Vijrumbana
Department of Mechanical Engineering
University College of Engineering (A),
Osmania University,
Hyderabad (T.S.) [INDIA]
Email: vijrumbhana@gmail.com

ABSTRACT

A scramjet engine starts at a hypersonic free stream Mach number 5.0. In order to propel to those speeds, staging with turbojet engine (up to M~3-4) and ramjet engine (up to M~5-6) is done in series or simply rocket engine is used prior to the scramjet engine. By reducing starting Mach number of a scramjet engine, the propulsion system can be simplified by reducing weight and complexity. The key design parameters of a pure scramjet engine (a scramjet with one combustor and a non-variable flow path) are varied and manipulated for lowering the starting Mach number. A one-dimensional flow analysis is presented in the paper. The effect of driving design parameters for a scramjet is studied and they were found to have a significant impact in lowering the starting Mach number. According to one-dimensional analysis, it has been found that a scramjet with a starting Mach number of 3.50 is possible without fuel additives or other additions to the overall system.

Keywords*:—*Pure Scramjet Engine, Key Design Parameters, Hydro Carbon Fuel.

I. INTRODUCTION

Satellites are launched into orbit by multi-staged satellite launch vehicles, generally are carrier rockets. These launch vehicles carry oxidiser along with the fuel for combustion to produce thrust. They are expensive and their efficiency is low because they can carry only 2-4% of their lift-off mass to orbit. Thus, there is a worldwide effort to reduce the launch cost.

Nearly 70% of the propellant (fuel-oxidiser combination) carried by today's launch vehicles consists of oxidiser. Therefore, the next generation launch vehicles must use a propulsion system which can utilise the atmospheric oxygen during their flight through the atmosphere. It will considerably reduce the total propellant required to place a satellite in orbit. Even it will help in increasing the payload thereby decreasing the satellite placing cost.

Ramjet, Scramjet and Dual Mode Ramjet (DMRJ) are the three concepts of air-breathing engines which are being developed by various space agencies.

Ramjet engines have no moving parts. The free stream supersonic air decelerates to subsonic speeds during compression. Air with increased temperature and pressure is then combusted with the fuel. Lastly, a nozzle accelerates the exhaust to supersonic speeds, resulting in thrust. Due to the deceleration of the free stream air, the pressure, temperature and density of the flow entering the burner are "considerably higher than in the free stream".

At flight Mach numbers of around Mach 6, the increase of pressure, temperature and density make it inefficient to continue to decelerate the flow to subsonic speeds. Thus, the flow is no longer slowed down to subsonic speeds, but it is slowed to acceptable supersonic speeds. Such ramjet is termed as 'supersonic combustion ramjet,' resulting in the acronym SCRAMJET. Figure 1 below shows a two-dimensional schematic of a scramjet engine.

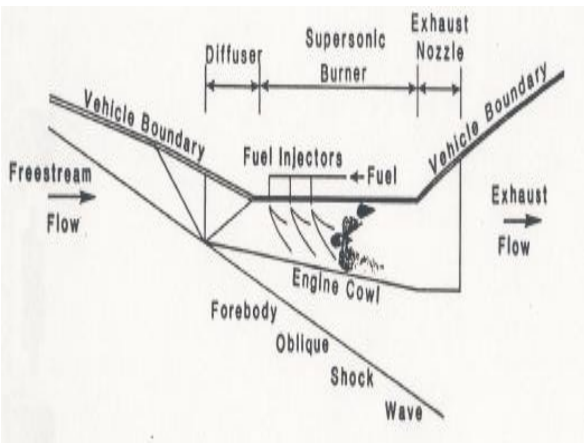

Figure 1: Two-Dimensional Schematic of a Scramjet Engine

As ramjets, a scramjet relies on high vehicle speed to forcefully compress and decelerate the incoming air before combustion. Airflow in a scramjet is supersonic throughout the entire engine. It allows the scramjet to efficiently operate at extremely high flight speeds.

II. Literature Review

Scramjet engines are a type of jet engine, and rely on the combustion of fuel with oxygen to produce thrust. It is composed of three basic components: a converging inlet, where incoming air is compressed and decelerated; a combustor, where gaseous fuel is burned with atmospheric oxygen to produce heat; and a diverging nozzle, where the heated air is accelerated to produce thrust. Figure 2 shows the sectional view of a Scramjet engine.

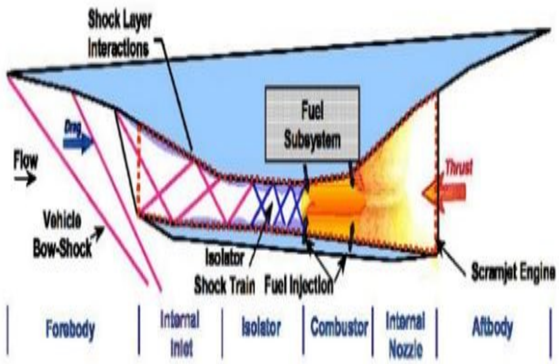

Figure 2: Sectional View of a Scramjet Engine

Similar to conventional jet engines, scramjet-powered aircraft carry the fuel on board. They obtain the oxidizer by the ingestion of atmospheric oxygen (as compared to rockets, which carry both fuel and an oxidizing agent). Such requirement limits scramjets to suborbital atmospheric flight, where the oxygen content of the air is sufficient to maintain combustion. Scramjets are designed to operate in the hypersonic flight regime, beyond the reach of turbojet engines, and, along with ramjets. They fill the gap between the high efficiency of turbojets and the high speed of rocket engines. Unlike a typical jet engine, such as a turbojet or turbofan engine, a scramjet does not use rotating or fan-like components to compress the air. The achievable speed of the aircraft moving through the atmosphere causes the air to compress within the inlet as such, no moving parts are needed in a scramjet. In comparison, typical turbojet engines require inlet fans, multiple stages of rotating compressor fans, and multiple rotating turbine stages, all of which add weight, complexity, and a greater number of failure points to the engine.

Due to the nature of their design, scramjet operation is limited to near-hypersonic velocities. As they lack mechanical compressors, scramjets require the high kinetic energy of a hypersonic flow to compress the incoming air to operational conditions. Thus, a scramjet powered vehicle must be accelerated to the required velocity by some other means of propulsion, such as turbojet, railgun, or rocket engines. While scramjets are conceptually simple, actual implementation is limited by extreme technical challenges. Hypersonic flight within the atmosphere generates immense drag. The temperatures found on the aircraft and within the engine can be much greater than that of the surrounding air. Maintaining combustion in the supersonic flow presents additional challenges. Also the fuel must be injected, mixed, ignited, and burned within milliseconds. While scramjet technology has been under development since the 1950s, only very recently have scramjets successfully achieved powered flight.

III. Problem Description and Scope of Current Work

A turbojet engine can provide thrust from takeoff to a speed of Mach 3 or 4. Therefore, if a scramjet is designed with a starting Mach number of about 3.50, presumably only two propulsion systems would be needed for the entire mission, whether that is up to Mach 8-10 for a hydrocarbon-powered scramjet or up to Mach 15-20 for a hydrogen-powered scramjet.

The advantage of the technology is clear—the resulting reduction in overall vehicle weight, lower mass fraction required for the propulsion system, and fewer systems that must work in succession reliably, thereby increasing overall vehicle safety.

Even use of rocket engines only up to M~ 3.5 instead up to M~ 5, will considerably reduce the overall engine weight there by complexity and cost.

3.1. Manipulation of "Pure" Scramjet Engine Key Design Parameters

"Pure" scramjet engine is a scramjet with one combustor and a non-variable flow path. There are a few key parameters of the pure scramjet engine, that are able to be varied and manipulated to perhaps lower the starting Mach number of a scramjet. For instance, as the cycle static temperature ratio (φ = T3/T0) increases, the Mach number of the flow entering the burner (M3) decreases. Thus, T3/T0 directly affects the freestream Mach number (M0) at which the flow entering the burner (M3) becomes supersonic. Due to this, it is possible that the manipulation of T3/T0 would yield a lower freestream Mach number at which supersonic combustion can occur. Additionally, the key design parameters of fuel selection and fuel-to-air ratio (f) for the scramjet may have an impact on the scramjet starting Mach number.

3.2. Scramjet Reference Station Designations

Before setting out in the analysis, it is first necessary to establish the reference station designations of the scramjet engine. Using Heiser and Pratt's designations, Figure 3 and figure 4 show the designation system used in the paper.

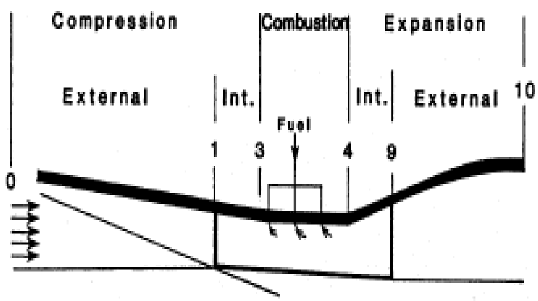

Figure 3: Scramjet Reference Station Designations

Reference Station	Engine Location
0	Undisturbed or freestream conditions
	External compression begins
1	External compression ends
	Internal compression begins
	Inlet or diffuser entry
3	Inlet or diffuser exit
	Internal compression ends
	Burner or combustor entry
4	Burner or combustor exit
	Internal expansion begins
	Nozzle entry
9	Internal expansion ends
	Nozzle exit
	External expansion begins
10	External expansion ends

Figure 4: Engine Locations for Scramjet Reference Station Designations

IV. DESIGN PARAMETERS TO REDUCE SCRAMJET STARTING MACH NUMBER

The one-dimensional stream thrust performance analysis inputs and how they are determined is given in Table 1. As seen in the table, the vast majority of the inputs are set by the freestream Mach number, are properties that remain constant for air, earth, etc., and are assumed based on reasonable values within a typical range. The values for the constant and assumed inputs can be seen in Table 2 below. All assumed values were chosen based on recommendations from Reference 1. Additionally, all constants were determined based on information in Reference 1.

Table 1: Performance Analysis Inputs and Corresponding Determination Methods

Performance Analysis Inputs	How Determined
M0, V0, T0, p0	Set by Mach Number
Cpc, R, Cpb, Cpe, hf, g	Constant
Vfx/V3, Vf/V3, Cf·(Aw/A3), ηc, ηb, ηe, T0, p10/p0, γc, γe, γb	Assumed
T3/T0, f, hPR	Variation

4.1. Stream Thrust Analysis Method for Hypersonic Air breathing Engine Performance Analysis

The stream thrust method requires more initial information and uses the entire set of control volume conservation equations. It leans heavily on momentum relationships and offers a different approach than other energy methods. The Stream Thrust Analysis method has the greatest depth. It is able to account for several phenomena that considerably influence performance, namely: "the mass, momentum, and kinetic energy fluxes contributed by the fuel, the geometry of the burner, and exhaust flows that are not matched to the ambient pressure". Due to these benefits, the Stream Thrust Analysis method will be employed here, as it accounts for the most engine parameters and influences compared to other one-dimensional flow analysis methods.

Table 2: Stream Thrust Inputs: Values for Constant and Assumed Values

Constants	
Cpc	1110.15 J/kg-K
R	295.063 J/kg-K
Cpb	1534.824 J/kg-K
Cpe	1534.824 J/kg-K
hf	0.00 kJ/kg
G	9.81m/s2
Assumed values	
Vfx/V3	0.5
Vf/V3	0.5
Cf·(Aw/A3)	0.1
ηc	0.9
ηb	0.9
ηe	0.9
T0	222 K
p10/p0	1.4
γc	1.362
γe	1.238
γb	1.238

Before discussing the equations necessary for the analysis, it is first necessary to define the control volume under consideration. Figure 5 shows the defined control volume for a scramjet. In this definition, the outside surface of the engine lines up with the dividing streamlines that constitute the internal and external flow boundaries.

The one-dimensional flow analysis method assumes that all of the flow through the engine is aligned in the axial direction, and therefore, that the through flow area is perpendicular to the axial direction. Also, it is assumed that the perfect gas constant (R) is constant across the engine, as the molecular weight of air does not vary enough across the engine to make a significant difference in the calculations.

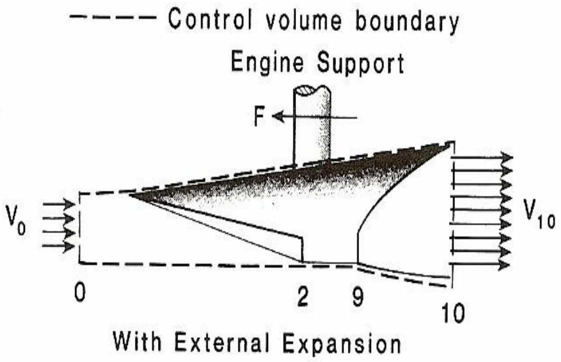

Figure 5:Scramjet Control Volume Definition

4.2. Theory and equations

Heiser and Pratt noted that the perfect gas law was used "repeatedly to eliminate density from the equations" and that p10/p0 is treated as an independent parameter in the equations for the analysis.

It is best to break the engine down into separate functional parts, as "significantly different physical phenomena are at work in each"; therefore, the present analysis operates with the following component breakdown:

Compression Component (Reference Stations 0 to 3): Includes compression surfaces (internal and external), isolator, intake, etc. up to the combustor entrance.

1. To check the condition for Mach number at combustion entrance ($M_3 \geq 1$)

$$M_3 = \sqrt{\frac{2}{\gamma_c - 1}\left\{\frac{T_0}{T_3}\left(1 + \frac{\gamma_c - 1}{2} {M_0}^2\right) - 1\right\}} \qquad (1)$$

2. Stream thrust function at freestream conditions.

$$Sa_0 = V_0\left(1 + \frac{RT_0}{{V_0}^2}\right) \qquad (2)$$

3. Combustor entrance temperature

$$T_3 = \varphi T_0 \tag{3}$$

4. Combustor entrance velocity

$$V_3 = \sqrt{{V_0}^2 - 2C_{pc}T_0(\varphi - 1)} \tag{4}$$

5. Stream thrust function at combustor entrance

$$Sa_3 = V_3\left(1 + \frac{RT_3}{{V_3}^2}\right) \tag{5}$$

6. Ratio of combustor entrance pressure to freestream pressure

$$\frac{p_3}{p_0} = \left\{\frac{\varphi}{\varphi(1 - \eta_c) + \eta_c}\right\}^{\frac{C_{pc}}{R}} \tag{6}$$

7. Ratio of combustor entrance area to freestream entrance area

$$\frac{A_3}{A_0} = \varphi \cdot \frac{p_0}{p_3} \cdot \frac{V_0}{V_3} \tag{7}$$

Combustion Component (Reference Stations 3 to 4): Consists of the combustor and all parts that make combustion happen including fuel injectors, etc. that lie within the combustor.

There are two methods for calculating the combustion properties, depending on the type of combustor designed: constant-pressure or constant-area combustion. The constant-pressure combustor is able to achieve results closest to ideal, since it is designed to conserve pressure, therefore generating less total pressure loss which in turn gives the engine a higher overall efficiency. Therefore, it is the combustor that will be used in the current work and the combustor type to which the proceeding equations apply.

However, there are some considerations to be made before these equations are listed. An absolutely constant- pressure burner is not feasible in terms of current manufacturing capabilities, so the application of an isolator which prevents inlet unstart should be used. Additionally, the burner walls will need to be more or less straight with a small variation in area in the axial direction. A constant ratio of the area at the combustor exit to entrance will be instilled; that is, variable geometry will not be used. The combination of a small variation in axial area combined with the use of isolator helps to achieve nearly equal pressures from the burner entry to the burner exit. The following equations apply to the constant-pressure combustor case. For the constant-pressure burner design p4/p0=p3/ p0 is assumed.

1. Combustor exit velocit

$$V_4 = V_3\left\{\frac{1 + f \cdot \frac{V_{fx}}{V_3}}{1 + f} - \frac{C_f \cdot \frac{A_w}{A_3}}{2(1 + f)}\right\} \tag{8}$$

2. Combustor exit temperature

$$T_4 = \frac{T_3}{1+f}\left\{1 + \frac{1}{C_{pb}T_3}\left[\eta_b f h_{PR} + f h_f + f C_{pb} T^0 + \left(1 + f \cdot \frac{V_f^2}{V_3^2}\right)\frac{V_3^2}{2}\right]\right\} - \frac{V_4^2}{2C_{pb}} \tag{9}$$

3. Ratio of area at combustor exit to combustor entrance

$$\frac{A_4}{A_3} = (1 + f) \cdot \frac{T_4}{T_3} \cdot \frac{V_3}{V_4} \tag{10}$$

4. Stream thrust function at combustor exit conditions

$$Sa_4 = V_4\left(1 + \frac{RT_4}{{V_4}^2}\right) \tag{11}$$

Expansion Component (Reference Stations 4 to 10): Includes all expansion surfaces after the combustor exit up to the engine exit.

1. Temperature at engine exit

$$T_{10} = T_4\left\{1 - \eta_e\left[1 - \left(\frac{p_{10}}{p_0}\cdot\frac{p_0}{p_4}\right)^{R/C_{pe}}\right]\right\} \quad (12)$$

2. Velocity at engine exit

$$V_{10} = \sqrt{V_4{}^2 + 2C_{pe}(T_4 - T_{10})} \quad (13)$$

3. Stream thrust function at engine exit conditions

$$Sa_{10} = V_{10}\left(1 + \frac{RT_{10}}{V_{10}^2}\right) \quad (14)$$

4. Ratio of area at engine exit to area at freestream entrance

$$\frac{A_{10}}{A_0} = (1+f)\cdot\frac{p_0}{p_{10}}\cdot\frac{T_{10}}{T_0}\cdot\frac{V_0}{V_{10}} \quad (15)$$

Overall Engine Performance Measures (Across Stations 0 to 10): With the analysis equations defined for the three engine components, it is now possible to establish the equations necessary to evaluate the overall engine performance. These equations are shown below, and also taken from Reference 1.

1. Specific thrust

$$F/\dot{m}_0 = (1+f)Sa_{10} - Sa_0 - \frac{R_0T_0}{V_0}\left(\frac{A_{10}}{A_0} - 1\right) \quad (16)$$

2. Specific fuel consumption

$$S = \frac{f}{F/\dot{m}_0} \quad (17)$$

3. Specific impulse

$$I_{sp} = \left(\frac{h_{PR}}{gV_0}\right)\eta_0 \quad (18)$$

4. Overall efficiency

$$\eta_o = \frac{V_0}{h_{PR}S} \quad (19)$$

5. Thermal efficiency

$$\eta_{th} = \frac{\left[(1+f)\frac{V_{10}^2}{2}\right] - \frac{V_0^2}{2}}{f h_{PR}} \quad (20)$$

6. Propulsive efficiency

$$\eta_p = \frac{\eta_0}{\eta_{th}} \quad (21)$$

7. Mach number

$$M = \frac{V}{\sqrt{\gamma RT}} \quad (22)$$

V. Analysis of Key Design Parameters

For one-dimensional flow analysis, there are only three inputs that are able to be varied to alter the performance results, simply there are three key design parameters. These are the cycle static temperature ratio (T3/T0), the fuel selected, and the fuel-to-air ratio (f). Therefore, these three variables are discussed and analyzed to check the possibility of starting Mach number of 3.5.

5.1. Influence of Cycle Static Temperature Ratio on the Scramjet Starting Mach Number:

The cycle static temperature ratio (T3/T0) has a powerful impact on the starting Mach number of a scramjet. As Equation 1 shows, T3/T0, the ratio of specific heats during compression (γ_c), and the free stream Mach number (M0) are the only variables which determine the burner entry Mach number (M3). The specific heat ratio can be considered constant in the compression component. With the requirement that M3>1 to ensure supersonic combustion, lowering M0 can apparently be achieved by lowering T3/T0. Therefore, T3/T0 has perhaps the greatest impact on whether it is possible to lower the scramjet starting Mach number. Thus, it is necessary to determine the required value of T3/T0 to achieve a starting scramjet Mach number of 3.5. Using Equation 1 with a range of free stream Mach numbers, and γ_c=1.362, the necessary T3/T0 for each free stream starting Mach number can be determined where M3≥1. The result can be seen in figure 6 from reference 6.

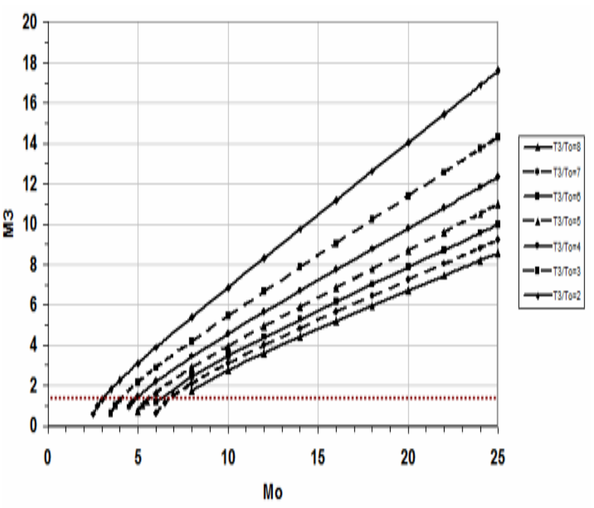

Figure 6: M_3 as a Function of M_0 and Cycle Static Temperature Ratio

As seen in figure 6, the value of T3/T0 at which supersonic flow (around Mach 3.5) is achieved at the entrance to the combustor is between 2 and 3. Specifically, solving Equation 1 with M0=3.50, γ_c=1.362, and M3 = 1, gives a T3/T0 of 2.72.

The constant dynamic pressure trajectory gives the corresponding values of velocity, temperature, altitude and pressure for the various free stream Mach numbers. For scramjet engine using hydrocarbon fuel, i.e. for free stream Mach range of 3.5-10, the flow properties for each Mach number on a constant pressure (q0=47,880 N/m2) trajectory are given in the table 3 below, taken from reference 1.

Table 3: Flow Properties for Each Mach Number on a Constant Dynamic Pressure (q_0=47,880 N/m_2) Trajectory

q_0	47880.26	N/m_2		
γ_c	1.362	-		
M	V_0 (m/s)	H (m)	T_0 (K)	P_0 (Pa)
3.5	1032.74	19936.97	216.65	5582
4	1184.56	21646.90	218.22	4275
4.5	1337.23	23167.85	219.73	3377
5	1490.40	24539.45	221.09	2736
5.5	1644.02	25786.08	222.33	2260
6	1798.05	26929.08	223.47	1900
6.5	1952.46	27989.78	224.52	1619
7	2107.21	28974.29	225.49	1397
7.5	2262.29	29894.78	226.41	1217
8	2417.68	30760.42	227.26	1069
8.5	2573.34	31494.85	228.07	982
9	2731.26	32146.45	229.17	864
9.5	2895.80	32968.78	231.21	735
10	3061.05	33790.13	233.16	684

The information obtained from the trajectory (V0, T0, p0) as well as M0=3.50, T3/T0=2.72, a standard fuel- to-air ratio (f) of 0.04, a standard fuel heat of reaction (hPR) of 87806.5 kJ/kg (which falls between hydrogen and methane's heats of reaction), and the constants and assumed values shown in Table 2 can then be entered into the series of equations shown in Equations 2 through 22 for a constant-pressure combustor to determine a reasonable estimate of the performance of a scramjet with a starting Mach number of 3.50. The results of doing such a thing can be seen in table 4 in the next page.

The performance values for specific impulse and specific thrust are reasonable, and the combustor entrance temperature (T3) is

Table 4: Preliminary Performance Results for Scramjet with Starting Mach Number of 3.50 to 10 with f=0.04 and h_{PR}=87806.5 kJ/kg

M	V_0 (m/s)	T_0 (K)	$F/\dot{m}_0$ (N-s/kg)	S (kg/N-s)	I_{sp} (s)	η0	ηth	ηp	M3	M4	M10	T_3 (K)	T_4 (K)	T_{10} (K)
3.5	1032.74	216.65	900.26	4.44E-05	2294.25	0.265	0.319	0.830	1.01	0.47	2.34	589.29	2562.99	1594.77
4	1184.56	218.22	832.28	4.81E-05	2120.99	0.281	0.316	0.888	1.55	0.73	2.44	593.56	2577.98	1604.10
4.5	1337.23	219.73	771.44	5.19E-05	1965.95	0.294	0.313	0.938	1.99	0.93	2.55	597.66	2594.42	1614.33
5	1490.40	221.09	717.05	5.58E-05	1827.35	0.304	0.310	0.983	2.38	1.12	2.67	601.36	2612.07	1625.31
5.5	1644.02	222.33	668.32	5.99E-05	1703.19	0.313	0.306	1.023	2.76	1.29	2.79	604.74	2631.01	1637.10
6	1798.05	223.47	624.57	6.40E-05	1591.69	0.320	0.301	1.061	3.12	1.46	2.92	607.84	2651.31	1649.73
6.5	1952.46	224.52	585.17	6.84E-05	1491.27	0.325	0.297	1.096	3.47	1.62	3.05	610.69	2672.99	1663.22
7	2107.21	225.49	549.58	7.48E-05	1400.56	0.330	0.292	1.130	3.81	1.78	3.19	613.33	2696.09	1677.59
7.5	2262.29	226.41	517.32	7.73E-05	1318.34	0.333	0.286	1.164	4.15	1.93	3.33	615.84	2720.69	1692.90
8	2417.68	227.26	487.98	8.20E-05	1243.57	0.336	0.281	1.197	4.48	2.08	3.47	618.15	2746.76	1709.12
8.5	2573.34	228.07	461.21	8.67E-05	1175.34	0.338	0.274	1.232	4.80	2.22	3.61	620.35	2774.36	1726.29
9	2731.26	229.17	436.39	9.17E-05	1112.11	0.339	0.268	1.268	5.13	2.36	3.75	623.34	2804.68	1745.16
9.5	2895.80	231.21	412.73	9.69E-05	1051.20	0.340	0.260	1.308	5.45	2.51	3.90	628.89	2840.25	1767.29
10	3061.05	233.16	390.96	0.000102	996.330	0.341	0.252	1.351	5.77	2.65	4.05	634.19	2877.52	1790.48

below the maximum allowed value range of 1440-1670K to prevent oxygen dissociation. Additionally, the combustor entrance pressure (p3) is within the allowable range of 0.5 atm to 10 atm (50.66 to 1013.25 kPa) up to Mach 6. The lower limit of the range is to support combustion and the upper limit is constrained by the structural and weight limits of the vehicle.

Although these results show reasonable performance values, there is one area that is of concern as it means that supersonic combustion did not occur throughout the combustor: M4. As Table 4 shows, the generic scramjet (designed only with T3/T0, a generic hPRand f, with a constant q trajectory) analysed here does not maintain supersonic combustion throughout the combustor with M3 and M4>1 until a free stream Mach number of 5.00. This is made even clearer by figure 7 which plots the combustor inlet and exit Mach numbers versus the free stream Mach numbers. The red dashed line shows the lowest limit for supersonic combustion to occur.

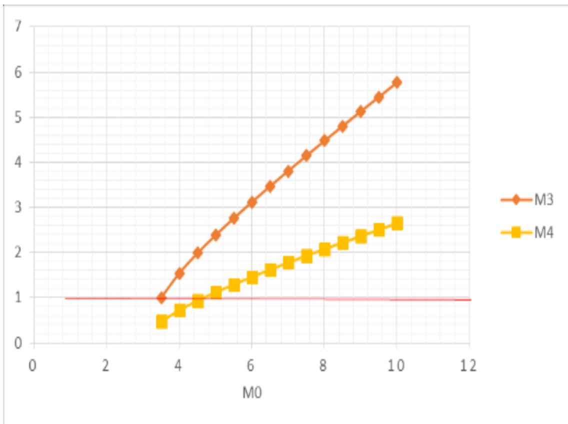

Figure 7: Preliminary Combustor Mach Numbers for Scramjet with Starting M_0=3.50

Therefore, it is necessary to determine if a different T3/T0 would produce supersonic combustion throughout the entire burner. M3 and M4 have been determined for a range of T3/T0 values so as to ascertain whether supersonic combustion can be maintained throughout the burner at a lowered starting Mach number. All other values remained the same as the previous calculation. Table 5 below shows these results.

Table 5: T_3/T_0 Required for Each Lowered Starting Mach Number

M_0	T_3/T_0	M_3	M_4
3.5	1.25	2.95	1.02
3.75	1.50	2.74	1.03
4	1.75	2.60	1.04
4.25	2.00	2.51	1.04
4.5	2.50	2.19	1.00

According to the analysis, the maximum T_3/T_0 for a generic scramjet with a starting Mach number of 3.50 to maintain supersonic combustion throughout the combustor is 1.25. Using a T_3/T_0 of 1.25 with the other variables remaining the same, the performance results can be seen in Table 6 in next page.

Although a T_3/T_0 value exists for lowering the starting scramjet Mach number to 3.50, the overall performance values shown in Table 6 are quite low. Though scramjet overall efficiencies are commonly around 50%, and are often as low as 25%, the overall efficiency here is only 9%. This is a very low efficiency and one that severely impedes any possible benefits for starting the scramjet at M0=3.50. Also, the values of specific impulse and specific thrust are significantly reduced as T_3/T_0 is very small. Therefore, it is clear that further investigation of the remaining design parameters is needed to aid in lowering the scramjet starting Mach number, so that the T_3/T_0 value may be selected as high as possible.

5.2. Influence of Fuel Selection on Lowering the Scramjet Starting Mach Number:

In seeking to lower the scramjet starting Mach number, it is therefore imperative that the fuel is carefully selected to ensure not only the goal is met for the current work, but also the fuel selection will not inhibit the resulting engine's application potential. A list of fuels, resulting from research into commonly applied fuels for scramjets as well as fuels used in other applications to be analysed are is used and analysed in the paper. The purpose of fuel in a propulsion system is, in the most basic sense, to convert the chemical energy stored in the fuel to thermal energy which can produce thrust. It is therefore beneficial to distinguish between the various fuels on a chemical level.

Table 6: Performance Results for T_3/T_0 = 1.25 at M_0=3.50

M_0	3.5	η_p	2.27
V_0	1032.74(m/s)	M_3	2.95
T_0	216.65(K)	M_4	1.00
F/m_0	316.81 (N-s/kg)	M_{10}	1.30
S	1.26×10^{-4}(kg/N-s)	T_3	270.81 (K)
I_{sp}	807.36(s)	T_4	2280.47(K)
η_0	0.093	T_{10}	2124.83 (K)
η^{th}	0.041		

The fuels which will be analysed here can be seen in descending order of h_{PR} in table 7 taken from reference 6.

Table 7: Heats of Reaction of Fuels for Analysis

Fuel Type	hPR (kJ/kg of Fuel)
Hydrogen	119954
Methane	50010
Ethane	47484
Hexane	45100
Octane	44786
JP-7	43903
JP-10	42100

With this characteristic established, it is now possible to consider the possible fuels. The two types of fuels most generally applied to scramjet designs are hydrogen and hydrocarbon fuels. Hydrogen fuels are often favoured for flight above Mach 8-10 whereas hydrocarbon fuels are favoured for Mach number ranges below 8. There are many advantages and disadvantages to both types of fuels which should be considered. Tables 8, 9 presents the advantages and disadvantages of hydrogen and hydrocarbon fuels respectively (from reference 1, 6, 9 and 10).

Table 8: Hydrogen Fuel Advantages and Disadvantages

	Advantages	Disadvantages
Hydrogen Fuel	Rapid burning	Very low density
	High mass specific energy content	Boil-off problems
	Shortest ignition delay	Requires largest vehicle size

Another property that should be considered is the ignition temperature of each fuel to be analysed. The ignition temperature (IT) of a fuel is the temperature at which the fuel will self-ignite in air without a flame source or spark. The IT of the various fuels is very important, as a scramjet with a starting Mach number of 3.50 will have relatively low air temperatures, therefore requiring the fuel to be able to ignite at those temperatures. The IT of the fuels which will be analysed can be seen in Table 10 (from reference 1, 6, 9 and 10).

Table 9: Hydro-Carbon Fuel Advantages and Disadvantages

	Advantages	Disadvantages
Hydro- Carbon Fuels	Storable	Slow burning
	Handling is familiar	Long ignition delay time
	11 times the storage density of hydrogen	Require quick vaporization before mixing
	3.5 times more energy content per volume than hydrogen	Exposing to high temperatures can result in coking
	Some offer smaller vehicles, logistic simplicity	
	Safer to handle than hydrogen	
	Realistic ground testing in existing facilities is possible	

Table 10: Ignition Temperatures of Fuels for Analysis at 1 ATM

Fuel Type	IT (K)
Hydrogen	845.15
Methane	810.15
Ethane	745.15
JP-10	518.15
JP-7	514.15
Hexane	498.15
Octane	479.15

The heat of reaction and the stoichiometric fuel-to-air ratio of each of the fuels to be analysed will be applied in order to determine whether fuel selection impacts the starting Mach number by an amount significant enough to enable scramjet to start at a free stream Mach number of 3.50. The constants and assumed values shown in Table 2 will be used here. The stoichio metric fuel-to-air ratio (f_{st}) is calculated for each of the fuels by equation 23 below.

$$f_{st} = \frac{36x + 3y}{103(4x + y)} \quad (23)$$

Where the fuel is represented in the form of C_xH_y

Table 11 shows the results from calculating f_{st} by Equation 23 for each of the fuels studied in this analysis with the h_{PR} and f_{st} established, the one-dimensional flow analysis equations shown in Equations 2 through 22 have been run repeatedly for a range of free stream Mach numbers to determine the maximum T3/T0 possible with supersonic flow maintained throughout the burner, that is, where M3 and M4 are both greater than 1. The advantage of using a range of freestream Mach numbers is that if the fuel is not able to maintain supersonic combustion at a T3/T0 for Mach 3.50 with practical performance, it will be possible to determine the minimum freestream Mach number at which the fuel can do so. The ignition temperatures listed in Table 10 must be taken into account to determine the T3/T0 required for each fuel to ignite in the airflow at each freestream Mach number. Therefore, the results for the maximum possible T3/T0 for each fuel were then compared to the necessary T3/T0 to ignite each fuel for each freestream Mach number. The results of such comparison is shown in figures 8-14 (from reference 6).

Table 11: Stoichiometric Fuel-to-Air Ratios for Fuels for Analysis

Fuel	Chemical Formula	f^{st}
Hydrogen	H2	0.0291
Methane	CH4	0.0583
Ethane	C2H6	0.0624
Hexane	C6H14	0.0659
Octane	C8H18	0.0664
JP-7	C12H25	0.0674
JP-10	C10H16	0.0707

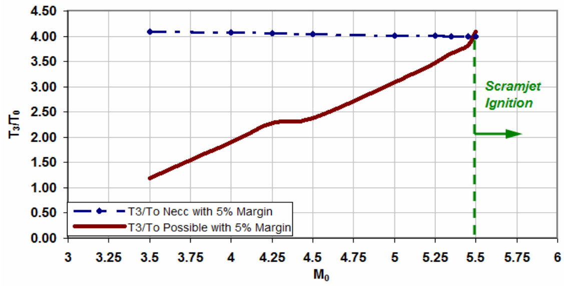

Figure 8: Lowest Possible Starting Mach Number for Hydrogen Ignition

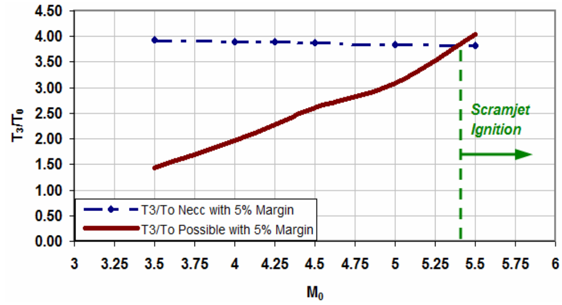

Figure 9: Lowest Possible Starting Mach Number for Methane Ignition

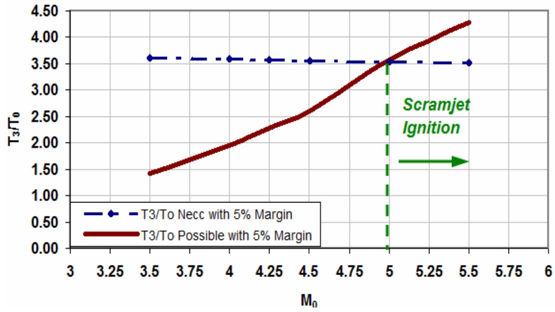

Figure 10: Lowest Possible Starting Mach Number for Ethane Ignition

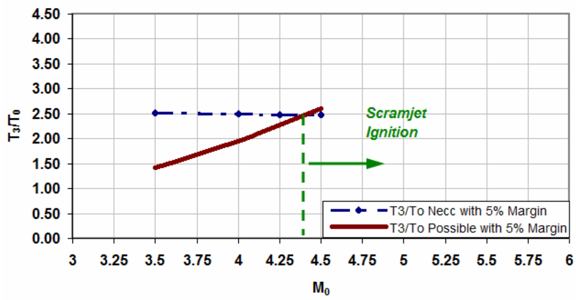

Figure 11: Lowest Possible Starting Mach Number for JP-10 Ignition

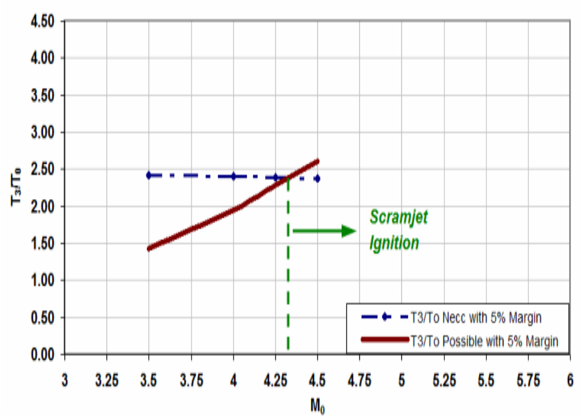

Figure 12: Lowest Possible Starting Mach Number for Hexane Ignition

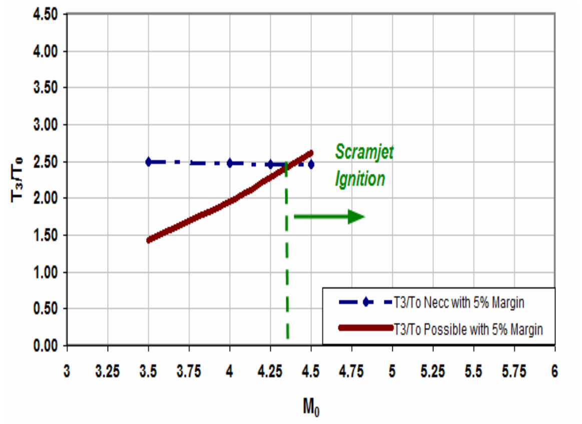

Figure 13: Lowest Possible Starting Mach Number for JP-7 Ignition

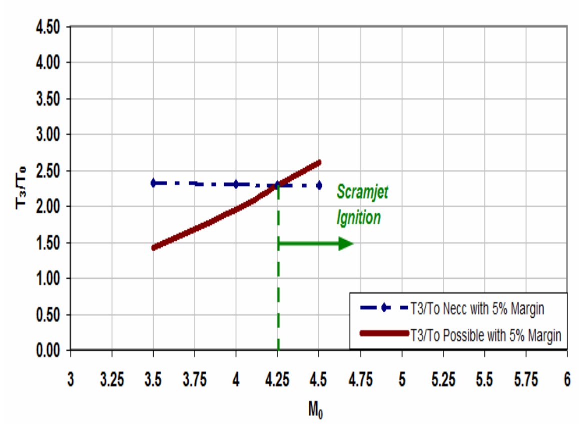

Figure 14: Lowest Possible Starting Mach Number for Octane Ignition

A summary table of the results from the previous seven figures is provided in Table 12. Methane is mentioned often as a fuel for a hypersonic cruiser with Mach 6+ cruising speed and hydrogen, as previously discussed, is applicable for Mach 8 - 10+ applications due of its high energy content. However, neither hydrogen nor methane is a feasible fuel for the current project, as their lowest starting Mach numbers are 5.50 and 5.35, respectively, therefore providing no ability to lower the starting Mach number. Ethane is not applicable as a feasible fuel either, since its lowest starting Mach number at f_{st} is 5.0. JP-10 is used in missiles and some Pulsed Detonation Engine designs. Though it reduces the starting Mach number at $f_{st}$to 4.35, it is not the best choice available for this application as there are other fuels which are able to reduce it even further.

JP-7 is used for military applications today. It reduces the starting Mach number at f_{st} to 4.3, and is a candidate fuel due to its wide availability and engine cooling capabilities. Hexane is similar in starting Mach number to JP-7, reducing it to 4.3 as well. However, hexane is not a necessary choice for further pursuit, as its results are similar to JP-7 and is not used as widely. Octane is a widely available fuel and succeeds in reducing the starting Mach number the furthest to 4.25, even its IT is also considerably low compared to JP-7, and it is comparably a cheaper fuel. Therefore it is also a candidate fuel for the project under study.

Table 12: Summary of Lowest Starting Mach Numbers and Corresponding T3/T0 for Analysed Fuels

Fuel	IT (K)	M0	T3/T0
Hydrogen	845.15	5.50	4.00
Methane	810.15	5.35	3.75
Ethane	745.15	5.00	3.50
JP-10	518.15	4.35	2.50
JP-7	514.15	4.30	2.50
Hexane	498.15	4.30	2.40
Octane	479.15	4.25	2.25

The fuels which performed the best at stoichiometric fuel-to-air ratios and were the most applicable overall are JP-7 and octane. However, neither fuel reduces the starting Mach number to 3.50. There is still another key design parameter to explore—varying the fuel-to-air ratio. Before trying to further reduce the starting Mach number further though, a decision should first be made as to which fuel is better suited for the current work. Considering the properties like lower M0, IT, and cost, Octane is selected for the current work.

The resulting performance of the scramjet engine with Octane as a selected fuel with lowest possible M0=4.25 at f_{st} = 0.0664 is given in table 13.

As can be seen in the table 13 as a result of selecting Octane as the fuel, the performance values have increased considerably over the values obtained with generic fuel values—the overall efficiency is now 27% versus the 9% obtained. The overall goal was accomplished as well, as the T3/T0 value at which supersonic combustion can be maintained was increased to 2.25 from 1.25 with the selection of Octane. However, the goal of the project has not yet been accomplished, as the desired starting Mach number of 3.50 has not been achieved; the lowest possible starting Mach number stands at 4.25 after fuel selection. Fortunately, there is another key design parameter that may be able to aide in achieving the goal —the fuel-to-air ratio.

Table 13: Performance Results with Octane Fuel at Stoichiometric *f* and T_3/T_0 of 2.25

Input			
M_0	4.25	T3/T0	2.25
V_0	1260.895(m/s)	F	0.0664
T_0	218.975(K)	hPR	44786(kJ/kg)
Engine Performance Measures			
f/m_0	641.43(N-s/kg)		
S	1.035 × 10-4(kg/N-s)		
I_{sp}	984.83(s)		
η_o	0.27		
η_{th}	0.26		
η_p	1.0543		
M_3	2.22	T3	492.70(K)
M_4	1.03	T4	2144.09(K)
M_{10}	2.33	T10	1461.35(K)

5.3. Influence of Fuel-To-Air Ratio on Lowering the Scramjet Starting Mach Number:

The stoichiometricf is the fuel-to-air ratio which "usually results in the greatest liberation of sensible energy from the breaking of molecular bonds". In the variation of f, when f is smaller than f_{st}, the available oxygen is not fully used and when f is larger than f_{st}, fuel is wasted as not all of it can be burned. Therefore, f_{st} is the ideal upper limit for the fuel-to-air ratio. In order to know the limits for the variation off, the equivalence ratio is used. Defined by Equation 24 below, the equivalence ratio is the ratio of the fuel-to-air ratio used to the stoichiometric fuel-to-air ratio (from reference 1).

$$\phi = \frac{f}{f_{st}} \tag{24}$$

Heiser and Pratt state that a general guideline for the equivalence ratio is from 0.2 to 2 for "combustion to occur within a useful timescale". Therefore, the fuel-to-air ratio has been varied across such range for Octane. The analysis and its results is shown in table 14.

Table 14: Fuel-to-Air Ratios Used for Analysis with Octane Fuel and Corresponding Equivalence Ratios

Equivalence Ratio	F
0.78	0.118
1.63	0.108
1.48	0.098
1.34	0.089
1.19	0.079
1.04	0.069
0.89	0.059
0.74	0.049
0.59	0.039
0.45	0.03
0.30	0.02

In order to determine whether the variation of the fuel-to-air ratio helps reduce the scramjet starting Mach number, the Octane fuel property of h_{PR} was applied to Equations 2 through 22 with a range of freestream Mach numbers from 3.50 to 4.25 and the range of fuel-to-air ratios listed above. Since the limiting factor is the ignition temperature of the fuel, and the T_3/T_0 has been low due to the lower Mach number goal, the maximum possible T_3/T_0 for supersonic combustion to be maintained at each f was determined. Then, from these results, the minimum possible value of T_3/T_0 which could ignite the Octane fuel was found, as it is obviously necessary for supersonic combustion to begin. The minimum possible T_3/T_0 is found to be 2.3 for lowest allowable fuel-to-air ratio for Octane at lowest possible free stream Mach number M0=3.5. The resulting performance of the scramjet engine with the parameters defined is shown in the table 15.

Considering the final key design parameters for Octane fuel, the starting Mach number has been successfully lowered to Mach 3.5. The visible drawback of lowering the fuel-to -air ratio can be seen in the large reduction of specific thrust by over 400 N-s/kg. The 0.5% reduction in overall efficiency is due the small rise in the value of T3/T0 i.e., from 2.25 to 2.3 and lowering the fuel-to-air ratio from 0.0664 to 0.02.

Table 15: Final Key Design Parameters and Preliminary Performance Measures

Input			
M_0	3.5	T_3/T_0	2.3
V_0	1032.74(m/s)	F	0.02
T_0	216.65(K)	hPR	44786 (kJ/kg)
Engine Performance Measures			
f/m_0	229.587(N-s/kg)		
S	8.71 × 10-5(kg/N-s)		
Isp	1170.18(s)		
η_O	0.265		
η_{th}	0.2092		
η_p	1.265		
M_3	1.484		
M_4	1.023		
M_{10}	2.369		
T_3	498.295(K)		
T_4	1022.116(K)		
T_{10}	689.116(K)		

VI. Conclusions and Recommendations

The purpose of the paper is to find the lowest possible free stream Mach number at which a scramjet engine can start so as to reduce the weight of the craft by eliminating one of the propulsive systems with selected key design parameters. To achieve a supersonic combustion throughout the combustor, with a generic fuel, at free stream Mach number of 3.5, the cycle static temperature ratio was very low as 1.25. The overall efficiency was 9% which is very less compared to the general efficiency range of 25-50%. Even specific thrust (300 N-s/kg) is also very less. Selecting Octane fuel considerably raised overall efficiency from 9% to 27% and specific thrust from 300 to 641 N-s/kg at its stoichiometric fuel-to-air ratio of 0.0664 with considerable rise in cycle static temperature ratio (2.25) with free stream Mach number (4.25). The starting free stream Mach number for scramjet engine using Octane fuel is successfully lowered from 4.25 to 3.5 by changing the fuel-to-air ratio from 0.0664 to 0.02. But it caused a drastic reduction in the specific thrust from 641 to 229 N-s/kg. The reduction of overall efficiency by 0.5% is seen due to small rise in the value of T3/T0 i.e., from 2.25 to 2.3 and lowering the fuel-to -air ratio from 0.0664 to 0.02.

The goal of the work has been met, as it has been determined that:

- The driving design parameters for a scramjet have a significant impact in lowering the starting Mach number.
- According to one-dimensional analysis, a scramjet with a starting Mach number of 3.50 is possible without fuel additives or other additions to the overall system.
- It is worth stating here that there are other possible ways to rise the specific thrust and other performance values at Mach 3.5 through additions to the system. A couple of these are listed here and briefly discussed.
- *Use of an ignition system* is not preferred since it usually complicates the internal flow and increases drag. Also, it may not necessarily be helpful because it increases T3, which, in turn, increases the starting Mach number.
- *Use of additives* is recommended. There are several additives like TEB (tri-ethylborane), silane etc. in addition to many other chemicals, which will help in varying the fuel properties. Possible addition of such chemicals to Octane might improve the performance of scramjet engine at Mach 3.5.

Nomenclature

A = Area (m2)

Cp = Specific Heat (J/kg-K)

Burner Effective Drag Coefficient

f = Fuel-To-Air Ratio

fst = Stoichiometric Fuel-to-Air Ratio

= Specific Thrust (N-s/kg)

g = Acceleration Due To Gravity (m/s2)

hf = Absolute Sensible Enthalpy of Fuel (kJ/kg)

hPR = Heat of Reaction (kJ/kg) T = Temperature (K)

T0 = Reference Temperature (K)

V = Velocity (m/s) Vf = Fuel Injection Total Velocity (m/s) Vfx = Fuel Injection Axial Velocity (m/s)

Isp = Specific Impulse (s) M = Mach Number

p = Pressure (Pa)

R = Perfect Gas Constant (J/kg-K)

S	*= Specific Fuel Consumption*	*(kg/N-s)*
Sa	*= Stream Thrust Function*	*(m/s)*
η	*= Efficiency*	
γ	*= Ratio of Specific Heats*	
φ	*= Cycle Static Temperature Ratio*	
ϕ	*= Equivalence Ratio*	

Subscripts

b = at engine burner

c = at engine compressor

e = at engine expander or nozzle

0 = free stream condition

3 = at station 3 (burner entrance)

4 = at station 4 (burner exit)

10 = at station 10 (engine exit)

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