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EXPERIMENTALLY DETERMINING PERFORMANCE OF A RAMJET COMBUSTOR.

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CONTENTS

Introduction

This report on the combustion efficiency of a ramjet combustion chamber has been done, based on data collected by the Data Acquisition System (DAS). The static test on a connected pipe test facility does away with the air inlets meaning we wouldn't be dealing with the losses in total pressure associated with air being decelerated down the shockwaves and bends in the air inlet. So we restrict the analysis, done using Matlab, only to what happens inside the combustion chamber. This would also not involve the aerothermodynamics within the chamber.

Required thermodynamic parameters' values would be taken in from NASA Chemical Equilibrium Analysis to calculate the values required to determine the combustion efficiency.

Nibodh Boddupalli,
Summer of 2014.

Acknowledgements

Firstly, I would convey my deep gratitude to Dr. Subhash Chandran for providing me this opportunity of exposure to the huge advancements in technology that have been made indigenously.

My guide, Shri Abhishek Richhariya, who catered all the time that he could keep aside from his busy schedule and also taught me that every problem, however troublesome, should be approached in a try-angle.

His fellow scientists Darshan sir, Vivek sir and Behera sir for their help, and invaluable words of wisdom.

All the other employees of Liquid Ramjet Propulsion Department who made this experience of mine complete by showing around the various projects, workshop and the Ramjet Test Facility.

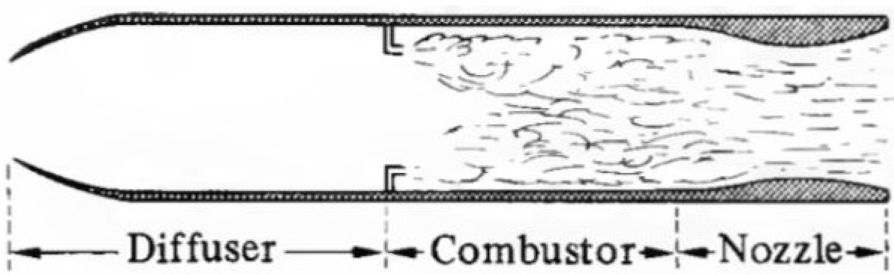
I got to see spectacular static tests of rockets small and large, read many books on subjects of Jet propulsion, Gas dynamics and Combustion at the library, and met many people with scientific temperament which was very inspiring.

Nibodh Boddupalli,
Summer of 2014.

What is a Ramjet

The ramjet engine is an air-breathing jet propulsion device. It gives higher fuel efficiency by using inlet air as a source of oxygen unlike rockets which carry an oxidizer as well. Because the ramjet depends only on its forward motion to effectively compress intake air, the "ram" air effect, the engine employs no moving parts unlike turbine engines; it is almost a simple tube.

Hence capable of simplicity, lightness, and high flight speed not possible in other air-breathing engines. Plus the high thermal efficiency make it a coveted choice for supersonic propulsion of missiles.



WORKING:

Thrust is generated as a reaction to the change in momentum brought about in the air (which is the working fluid) that enters and is slowed down before sending into the combustor, though a little fuel is added, plus due to the difference in ambient pressure and pressure at the exit nozzle.

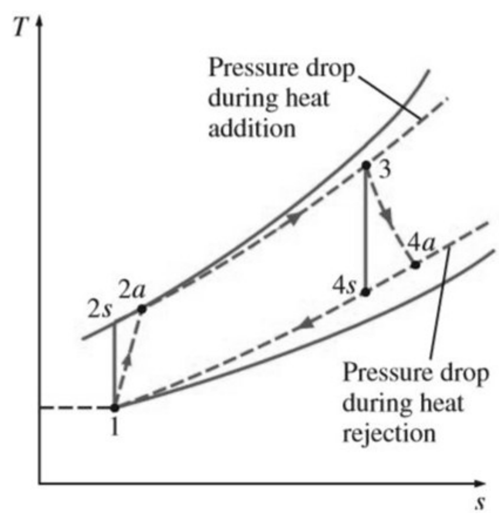
$$\boldsymbol{Thrust} = (\dot{\boldsymbol{m}}_{\boldsymbol{air}} + \dot{\boldsymbol{m}}_{\boldsymbol{fuel}})\boldsymbol{v}_{\boldsymbol{exit}} - \dot{\boldsymbol{m}}_{\boldsymbol{air}}\boldsymbol{v}_{\boldsymbol{entry}} + \boldsymbol{A}_{\boldsymbol{nozzle}}(\boldsymbol{P}_{\boldsymbol{exit}} - \boldsymbol{P}_{\boldsymbol{ambient}})$$

Where

- $\dot{m}_{air}$ and $\dot{m}_{fuel}$ - mass flow rates of air and fuel injected respectively.
- v_{exit}- velocity of exhaust.
- v_{entry} - velocity at which air enters (which is the aircraft velocity).
- A_{nozzle} - area of cross section of the nozzle.
- P_{exit} and $P_{ambient}$ - the pressure inside the nozzle and ambient pressure respectively.

This increase in velocity of the working fluid is a result of the expansive combustion of the fuel which is then accelerated through the nozzle to higher velocities while bringing down the pressure of the jet.

The working cycle would be best described by the Brayton cycle, the T-S graph of which is given to the right with deviations from ideal cycle (given by dotted lines) arising due to irreversibilities involved.

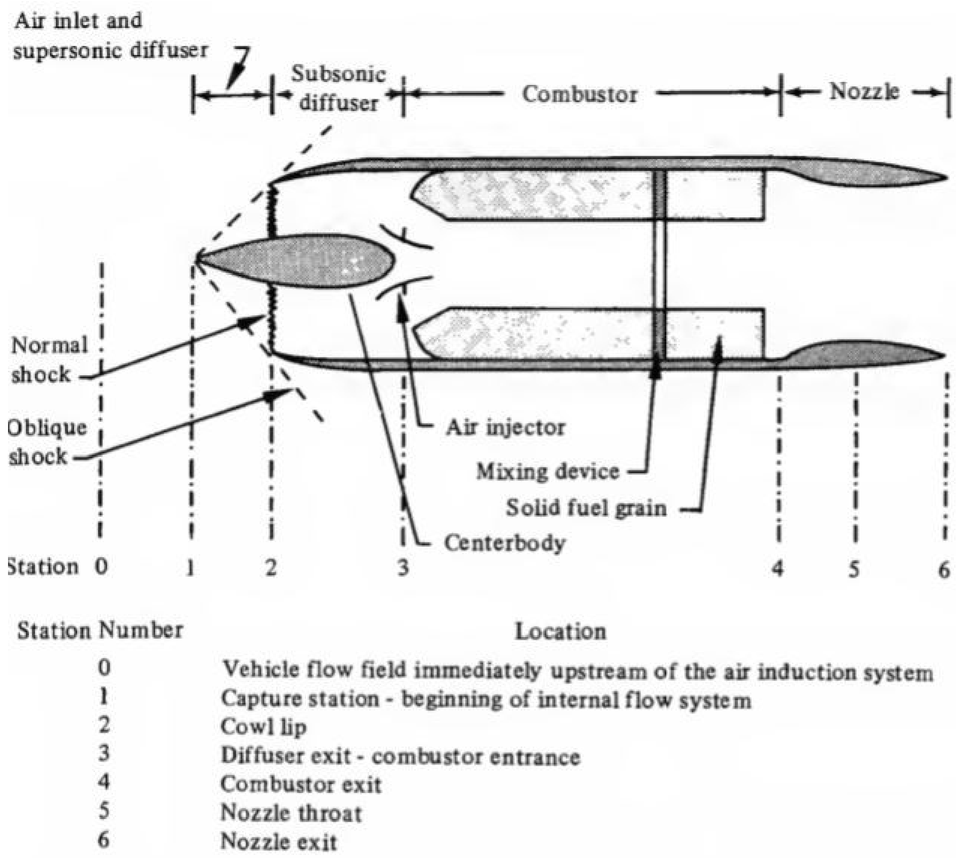


Types and Modifications

Types

- ***Liquid-fueled ramjet (LFRJ) -*** Liquid fuel is introduced in the combustor which includes a flame-holder, the combustion zone, and an exhaust nozzle. It requires a separate fuel storage system that can supply fuel to the delivery system. Despite the added components, LFRJs have the advantage of throttling fuel with the control system as required (which varies with vehicle altitude and flight speed) to control flight speed of the vehicle as desired.

- ***Solid-fueled ramjet (SFRJ) -*** The main characteristic that distinguishes the SFRJ from the LFRJ is the absence of fuel tankage, delivery, and control systems, since the fuel is entirely contained in the combustor at the beginning of the duty cycle. In addition, the combustor is usually simpler because there is no liquid phase fuel to be atomized and mixed with air. Instead, there is an air injector to increase the turbulence of the air as it enters the combustor so as to improve flame-holding. There may be a mixer to ensure that fuel-rich and air-rich gases are thoroughly mixed to improve combustion efficiency, which is always a key consideration in any combustion process involving a gas and a solid.

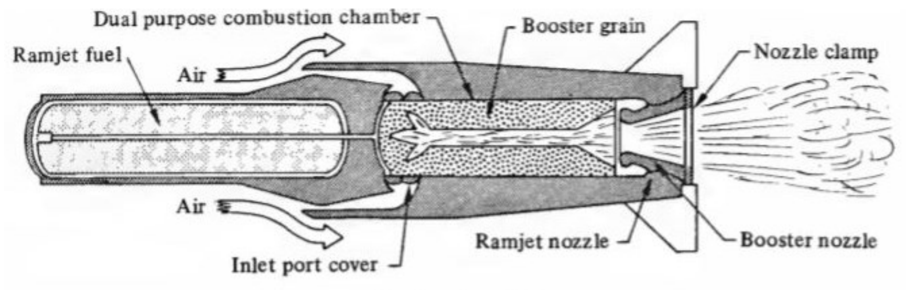


Solid-fueled Ramjet Nomenclature

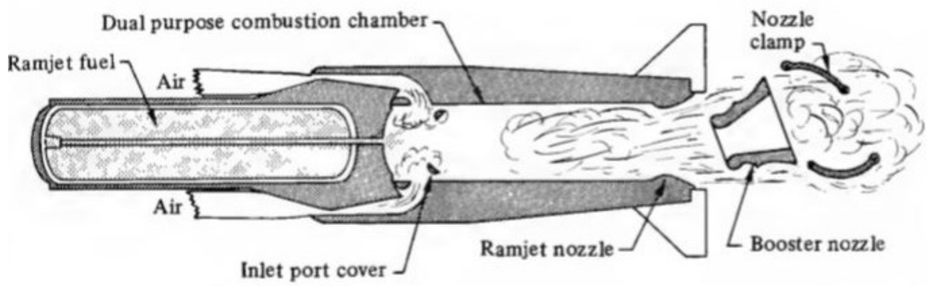
Modifications

Since a ramjet depends on the ram air effect to compress air, it cannot generate thrust from standstill. Usually a booster is used to propel the vehicle to a certain velocity (Rocket Assisted Take-Off) and is ejected or dropped off whereon the ramjet takes over to sustain flight.

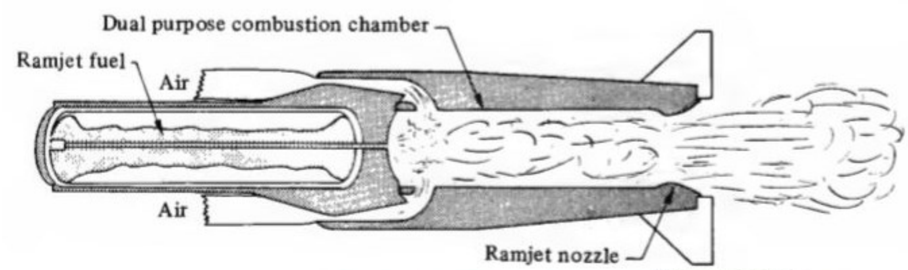
- ***Integral rocket ramjet (IRR) -*** Employs a dual-purpose combustor that first serves as a rocket combustion chamber for booster propellant cast into it which burns and accelerates the vehicle to a high speed. Then inlet air is allowed to enter the combustor where it encounters either the fuel. The fuel then burns in the combustor in the normal manner of a ramjet. They can be solid or liquid fueled.

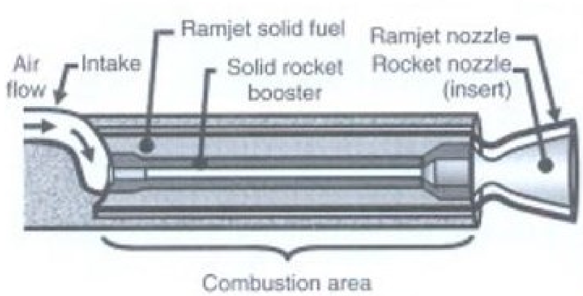


Booster Operation

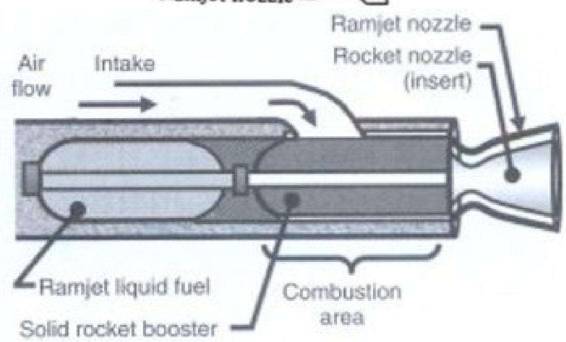


Transition



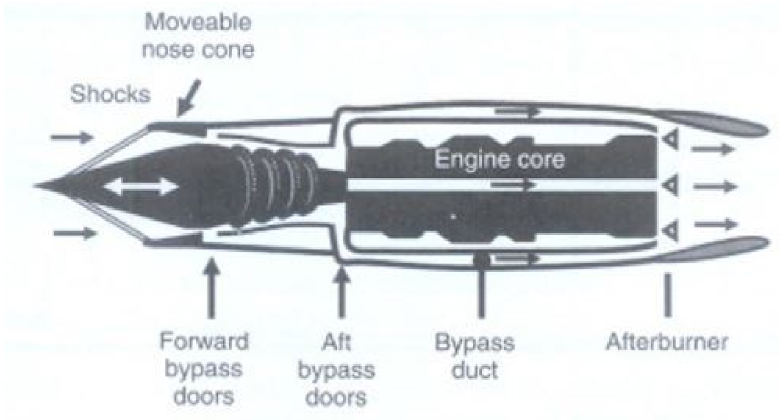


(a) Solid fuel ramjet

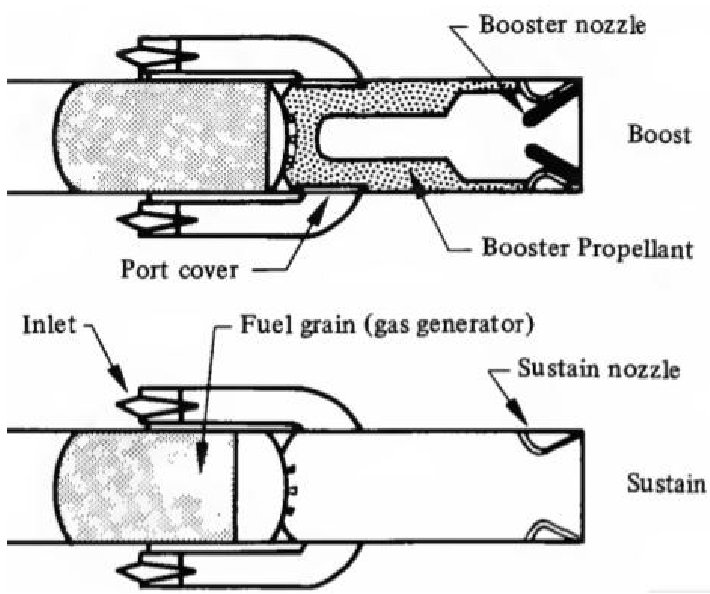


(b) Liquid fuel ramjet

- ***Turbo-ramjet –*** A turbine engine which uses a turbine to compress the air to propel from standstill like a turbojet and an air bypass mechanism to feed the ramjet for super-cruise. Used in the high altitude reconnaissance SR-71 aircraft.



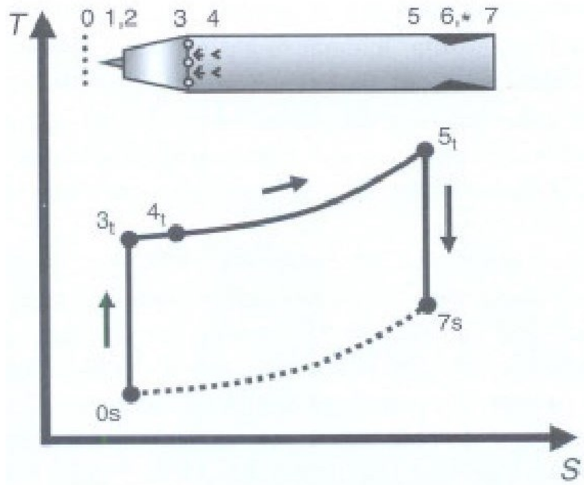
- ***Ducted rocket –*** Which is an air augmented rocket, close to the ramjet, is fueled by a propellant grain containing oxidizer which produces fuel rich gases which then burn in presence of air in the ramjet combustion chamber. Because the ducted rocket contains part of its oxidizer, it does not have a performance potential as high as a pure ramjet. This disadvantage is offset by increased operational flexibility. The ducted rocket therefore represents one of the simplest forms of ramjet-type engines in that there is a reduced dependence on flight parameters.



Combustion efficiency

In terms of enthalpy change

Combustion efficiency would be expressed as ratio of the actual enthalpy change to the maximum possible enthalpy change.



$$\eta_b = \frac{\dot{m}_5 h_{0\,5} - \dot{m}_4 h_{0\,4}}{\dot{m}_f h_{calorific\ value\ of\ fuel}}$$

Where the numerator is the rise in total enthalpy across combustor (4→5)

In terms of characteristic velocity

But in thi study, combustion quality as defined in rocketry would be used because the prime motive is the quality of thrust produced.

From the definitions:

- Coefficient of thrust $C_f = \frac{Thrust}{Chamber\ pressure\ \times Throat\ area}$
    - Characteristic velocity $\boldsymbol{C^*} = \frac{\boldsymbol{Chamber\ pressure\ \times Throat\ area}}{\boldsymbol{mass\ flow\ rate}}$ ...(1)

C^* turns out to be a characteristic property of the combustion reaction. It is a fundamental parameter of the energy available after combustion and can be used to compare different reactions independent of chamber pressure.

$$\xi_b = \frac{Thrust_{experimental}}{Thrust_{theoritical}} = \frac{C_{f\ exp}\, C^*{}_{exp}\, \dot{m}_{exp}}{C_{f\ th}\, C^*{}_{th}\, \dot{m}_{th}}$$

But as C_f depends on the geometry → $C_{f\ exp} = C_{f\ th}$

And $\dot{m}_{exp}$ *is same as* $\dot{m}_{th}$

$$\boldsymbol{\xi_b} = \frac{\boldsymbol{C^*{}_{exp}}}{\boldsymbol{C^*{}_{th}}} \qquad \dots (2)$$

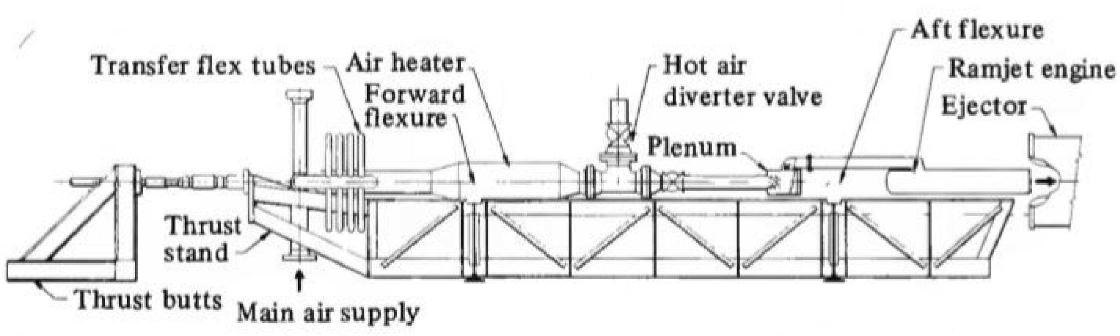
Where $C^*{}_{exp}$ is the characteristic velocity observed from the experiment and $C^*{}_{th}$ is the theoretical value of the characteristic velocity.

Various factors like the fuel injection, inlet air dump angle, combustor length, flame stabilization, etc (which haven't been studied upon in the test) also affect combustion efficiency widely because it is how the fuel is injected, atomized and carbureted before burning in particular air-fuel ratio, at the given flight conditions like speed and altitude, that determines the combustion efficiency.

Experimental setup

Of the three main methods used to test ramjet engines: freejet, semi-freejet and connected pipe, the third has been used.

Connected pipe tests are adequate for trying the combustion system, since the combustion process is not affected by flow outside the engine. In these tests, air at conditions corresponding to those leaving the intake diffuser is fed directly to the combustion system.



Direct-connect Ramjet Facility

The test facility is mainly constituted by the following:

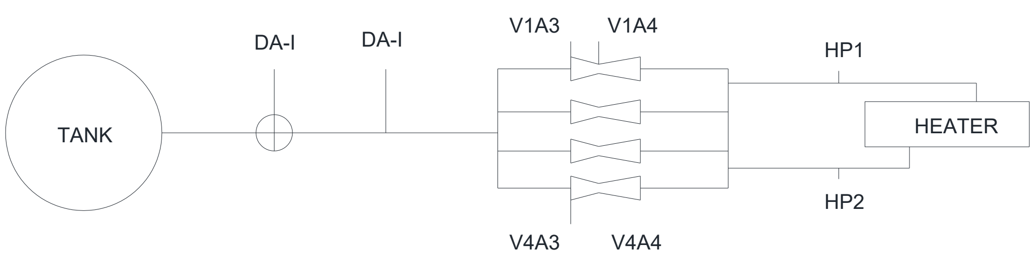
- Air-line
- Fuel line
- Test bed

Additionally a Hydrogen line for supplying hydrogen to heat the air in the heater and a Nitrogen line to pressurize (and purge) fuel and hydrogen.

The test facility being spread over a vast area with pipes and flexible pipes of various dimensions has many elbow, flange, nutter-nipple and ferrule joints, and manifolds.

It also has ball, needle, gate, electro-pneumatic and solenoid valves, and rotary actuators. Most of which function at 24V signals.

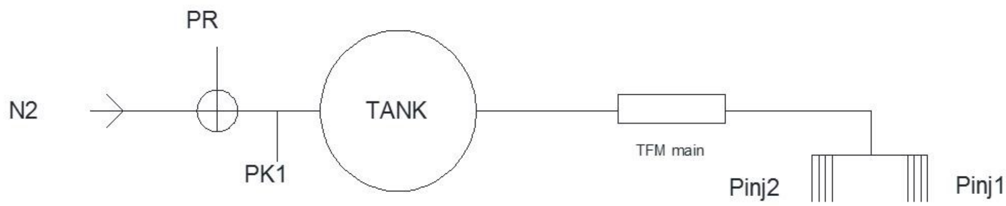
- **Air-line:**



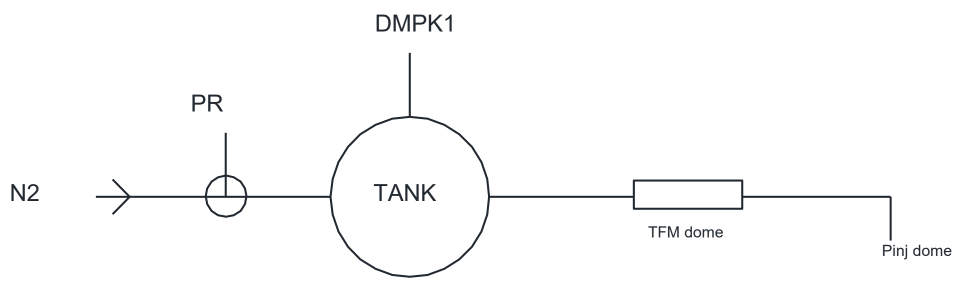
Atmospheric air is compressed by a compressor then stored in vessels at 120bar.

When the test is run, the discharge from these is via dome regulators which keep the pressure as desired. This is flow then split and measured further down the line by venturi tubes which are connected to a manifold which is connected to the heater. After this there is the still chamber whereon air is evenly distributed according to the number of intakes and fuel is injected into it and this enters the combustion chamber.

- **Fuel-line:**



Main fuel line

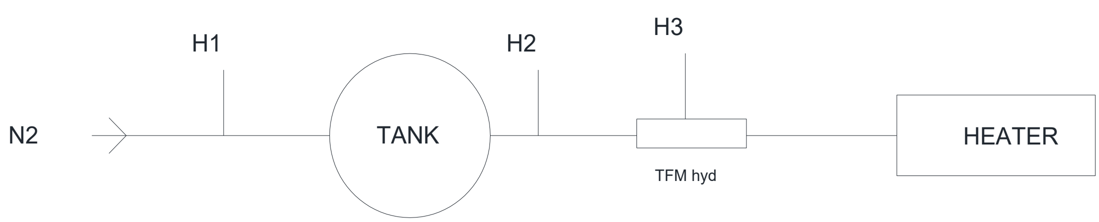


Dome fuel line

Fuel is stored for the swirl injector dome and the main injectors in two separate vessels.

This is pressurized using nitrogen (inert) and the outflow from the vessels is recorded using turbine flow meters for each line. The main fuel is injected via solenoid valves to the fuel injector block and the dome fuel to the swirl injector.

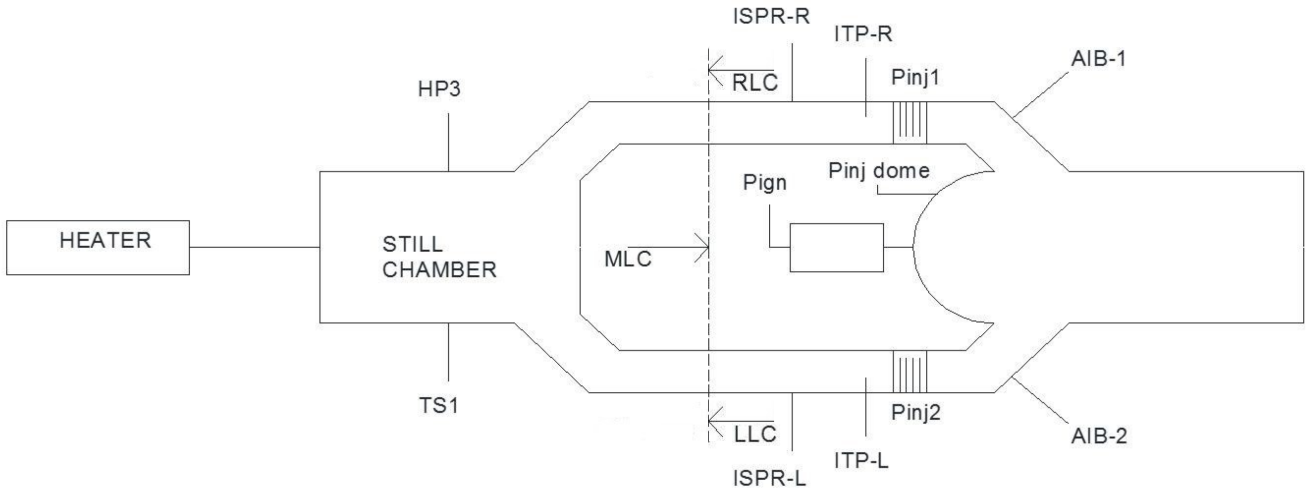
- **Hydrogen line:**



Hydrogen cylinders are kept in the hydrogen bay.

In the Test run, this is pressurized using nitrogen and the discharge pressure is regulated and flow rate recorded using a turbine flow meter. This is fed to the heater where it is burnt to heat the air.

- **Test bed:**



The thrust stand is firmly fixed to the concrete bed. On it lay the air heater, the still chamber the fuel injector block, the swirl injector dome attached to the combustion chamber arranged such that reaction to the thrust produced by the motor (combustion chamber) is given by the three piezoelectric load cells which give the readings of the thrust.

Measurements in test facility

At each point where the readings are to be taken, the corresponding instruments are placed.

- Surface pressure transducers - static pressure.
- Total pressure transducer - total pressure.
- K-type and R-type thermocouples – temperature.
- Load cells – normal reaction to thrust.
- Turbine flow meters (TFM) – fuel and hydrogen flow rates.
- Venturi meters – air flow rate.

Pressure transducer – May be piezoelectric or diaphragm based like the ones used. Static pressure is measure at the surface while total pressure is measured by a probe into the flow.

$$P = output \times g \times 10e4 + 101325$$

Output is given in *Kg-force*$/cm^2$ gauge pressure which is converted to *Pascals* absolute pressure.

Thermocouple – Work based on Seebeck effect producing a voltage difference across temperature variation.

$$T = output + 273.15$$

Output is given in °C which is converted to *Kelvin.*

Load cells – Piezoelectric load cells are placed to give a reading of the normal reaction exerted on the combustor.

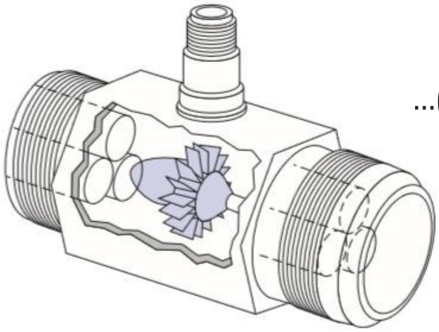
$$F = g \times output \quad \text{...(3)}$$

Output is given in *Kilogram-force.*

Turbine Flow Meter – A free-running vaned impeller may be mounted in a cylindrical section of tube (right) to make a turbine flow meter. The rate of rotation of the impeller is made closely proportional to volume flow rate over a wide range. Rotational speed of the turbine element is sensed using a magnetic pickup external to the meter. This sensing method therefore requires no penetrations or seals in the duct.

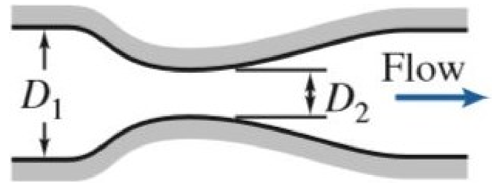
Mass flow through a Turbine Flow Meter (right) is given by the expression:

$$\dot{m}_f = \frac{\boldsymbol{TFM\ reading \times fuel\ density}}{\boldsymbol{TFM\ constant \times 1000}} \quad \text{...(4)}$$



Where TFM reading is the output given in Hz and fuel density is in SI units.

Venturi Meter – Mass flow q_m through a venturi tube is calculated using Pressure transducers' readings at the throat and upstream of it.



In a horizontally placed venturi meter (right) the flow rate is found using Bernoulli's equation:

$$\frac{p_1}{\rho} + \frac{V_1^2}{2} + \cancel{gz_1} = \frac{p_2}{\rho} + \frac{V_2^2}{2} + \cancel{gz_2}$$

Assuming a steady, incompressible flow along a streamline with uniform velocity and no friction or streamline curvature at sections 1 or 2, so pressure is uniform across them.

Then, from the Bernoulli equation,

$$p_1 - p_2 = \frac{\rho}{2}(V_2^2 - V_1^2) = \frac{\rho V_2^2}{2}\left[1 - \left(\frac{V_1}{V_2}\right)^2\right]$$

and from continuity

$$(-\rho V_1 A_1) + (\rho V_2 A_2) = 0$$

or

$$V_1 A_1 = V_2 A_2 \qquad \text{so} \qquad \left(\frac{V_1}{V_2}\right)^2 = \left(\frac{A_2}{A_1}\right)^2$$

Substituting gives

$$p_1 - p_2 = \frac{\rho V_2^2}{2}\left[1 - \left(\frac{A_2}{A_1}\right)^2\right]$$

Solving for the theoretical velocity, V_2,

$$V_2 = \sqrt{\frac{2(p_1 - p_2)}{\rho[1 - (A_2/A_1)^2]}}$$

The theoretical mass flow rate is then given by

$$\begin{aligned} \dot{m}_{\text{theoretical}} &= \rho V_2 A_2 \\ &= \rho \sqrt{\frac{2(p_1 - p_2)}{\rho[1 - (A_2/A_1)^2]}} A_2 \end{aligned}$$

or

$$\dot{m}_{\text{theoretical}} = \frac{A_2}{\sqrt{1 - (A_2/A_1)^2}} \sqrt{2\rho(p_1 - p_2)}$$

But this ideal formula would have added coefficients according to ISO 5167-4 2003(E) to accommodate for compressibility and actual discharge and hence would take the following form:

$$q_m = \frac{C}{\sqrt{1-\beta^4}} \varepsilon \frac{\pi}{4} d^2 \sqrt{2 \Delta p \rho_1} \qquad \text{...(5)}$$

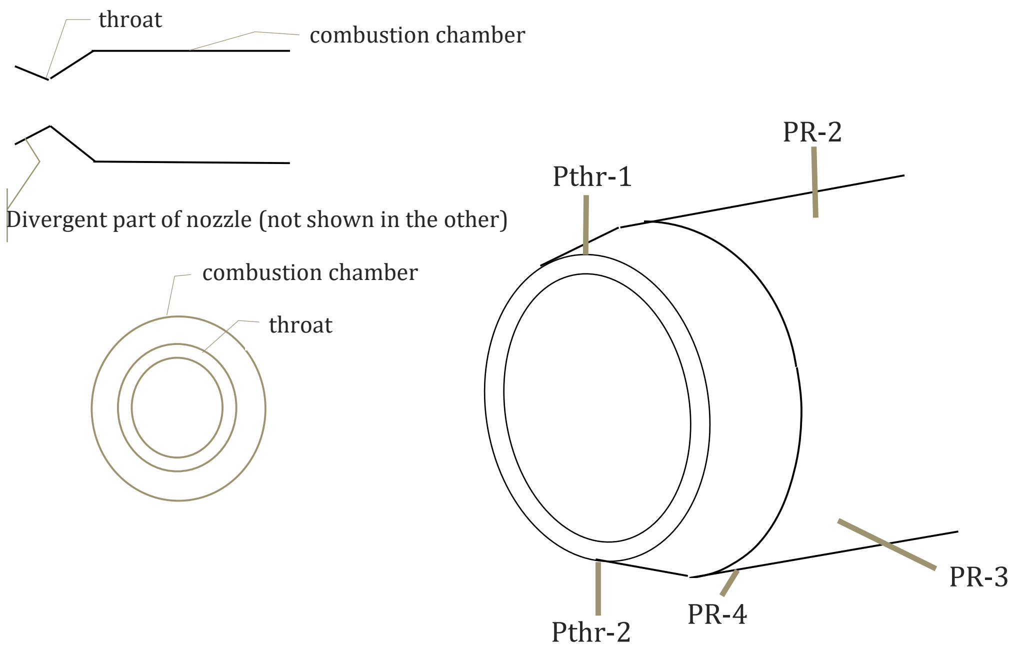
Where, the variables are assigned according to convenience.

- C is the coefficient of discharge of the venture meter C_{d}_air

- $\beta^2 = \frac{A_4}{A_3}$
- $$\varepsilon = \sqrt{\left(\frac{\kappa\tau^{2/\kappa}}{\kappa - 1}\right)\left(\frac{1-\beta^4}{1-\beta^4\tau^{2/\kappa}}\right)\left(\frac{1-\tau^{(\kappa-1)/\kappa}}{1-\tau}\right)}$$
- $\frac{\pi d^2}{4} = throat\ area\ A_4$
- $\Delta p = upstream\ pressure\ VA_3 - throat\ presssure\ VA_4$
- $\rho_1 = \frac{upstream\ pressure\ VA_3}{287 \times 300}$
- $\tau\ =\ \frac{pressure\ at\ throat\ VA_4}{pressure\ upstream\ VA_3}$
- $\kappa = \gamma = \frac{C_p}{C_v}$. ...(6)

Measurement in combustion chamber

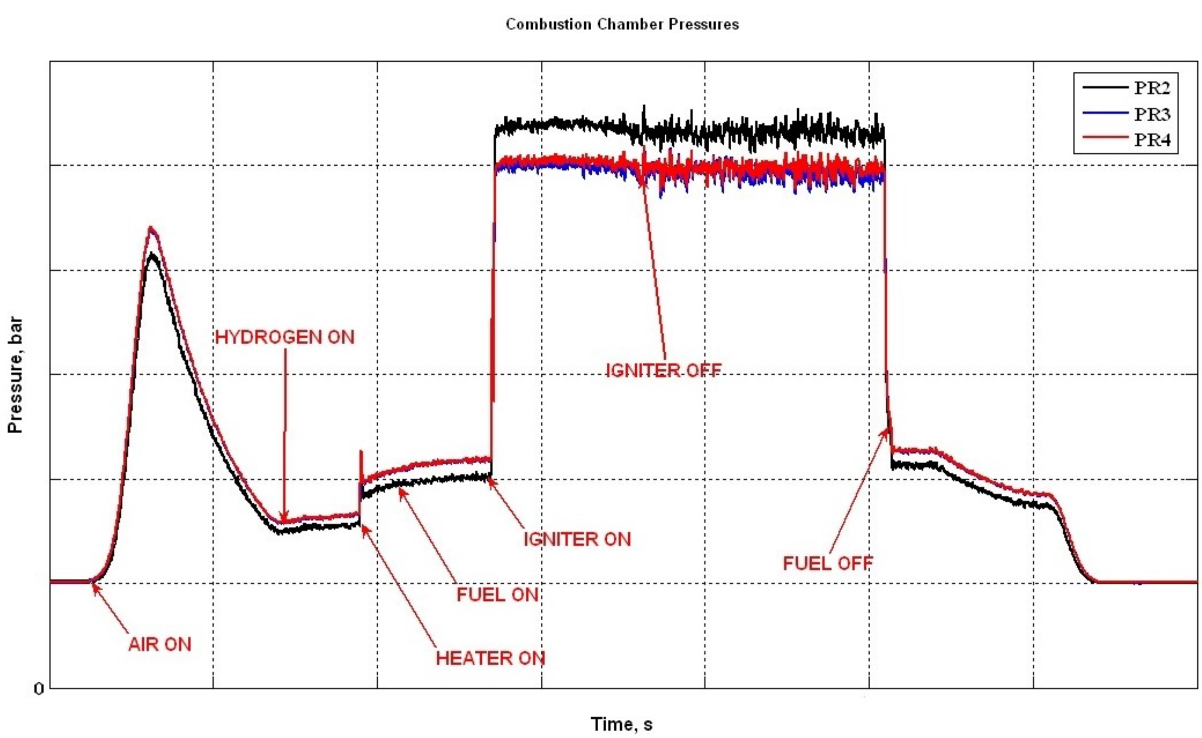
5 Pressure transducers, 3 on the combustor and 2 on the nozzle throat, pick up pressure readings.

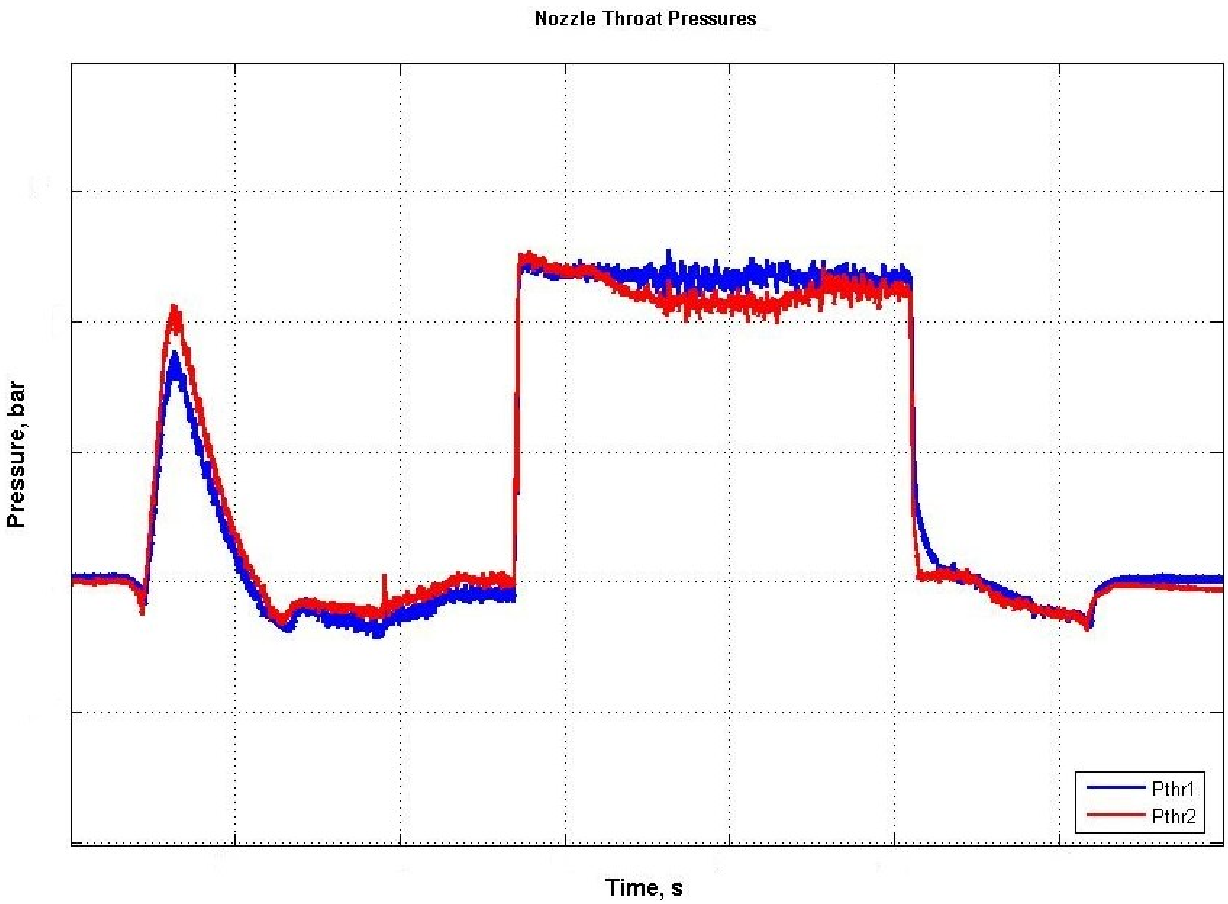


Note that the diverging part of the nozzle after the throat section has been shown only in one of the above

PR-2, PR-3 and PR-4 give the achieved chamber pressures on combustion.

Pthr-1 and Pthr-2 give the pressures at the nozzle throat.




Nozzle Throat Pressures
Pressure, bar
Time, s
Pthr1
Pthr2

Method of estimation of combustion efficiency

As seen earlier, the combustion efficiency (2) would be calculated by the ratio of experimental C^* and theoretical C^*.

$$\xi_b = \frac{C^*{}_{exp}}{C^*{}_{th}}$$

Theoretical C^* would be calculated as $C^*{}_{th} = \frac{\sqrt{gas\ constant\ R \times Combustion\ temperature\ T_0}}{\Gamma}$ ...(7)

Where, using (6), $\Gamma = \sqrt{\gamma \times \frac{2}{\gamma+1}^{\gamma+1/\gamma-1}}$ ...(8)

And

Experimental C^* (1) would be $C^*{}_{exp} = \frac{estimated\ total\ pressure \times C_D\ of\ nozzle \times Throat\ area}{total\ mass\ flow\ rate}$

Where C_D is the estimated coefficient of discharge of nozzle. ...(9)

First, the following assumptions are made that:

- The flow is one dimensional.
- The air is an ideal gas.
- The heat losses through the combustor may be neglected.
- The irreversibilities don't arise due to frictional losses.
- The nozzles are isentropic.
- The pressure and temperatures represent appropriate averaged values across the flow area.
- The flow at the igniter nozzle and the convergent part of the combustor nozzle is choked (calculating C^* by definition).

The assumptions would give results practically close to actual while keeping the procedure simple.

Theoretical C^*

Mass flow

Air: Would be calculated from flow through venturi (5) tubes and summed to get total flow. And would be taken as 0 if downstream pressure VA_4 is nil or greater or equal to upstream pressure VA_3.

$$\dot{m}_a = \dot{m}_1 + \dot{m}_2 + \dot{m}_3 + \dot{m}_4 \qquad ...(10)$$

Fuel: Main line and Dome line flows would be calculated from TFM (4) readings and summed to get total fuel flow if respective injection pressure values *Pinj* and *Pinj_dome* are higher than a predefined value of pressure required for injection. Else they are taken as 0.

$$\dot{m}_f = \dot{m}_{main} + \dot{m}_{dome} \qquad ...(11)$$

Air-fuel ratio is then calculated keeping it within predefined limits. If the air and fuel flow rates are very less, it is taken as 0. But if both are 0 meaning that the flows haven't begun, the upper limit is assigned.

Igniter mass flow should also be taken in mass flow rate *M_tot_ign* if the pressure *Pign* inside the ignition chamber is higher than a predefined value of pressure signifying ignition. Knowing the C^* of the igniter fuel, the area to its nozzle's throat and *Pign*, the mass flow rate of igniter gases is calculated using the definition of characteristic velocity (1). Else would not be considered.

Thermodynamic parameters ...(12)

Now when ignition isn't done or the air-fuel ratio is zero, meaning that the combustion hasn't begun yet, Ideal gas values are taken.

γ=1.4 C_p=1005 $kJ/kg.K$ R=287 $J/kg.K$ T_0=TS1(still chamber temperature)

When ignition just starts and air and fuel flow, combustion takes place and the resultant mixture's parameters are determined by interpolating NASA CEA values on air-fuel ratios *afr* with the air-fuel ratio *afrat.*

γ =interpolation (afr, gama, afrat) C_p=interpolation (afr, CPNASA, afrat)
R=8314/interpolation (afr, MOLwt, afrat) T_0=interpolation (afr, TR, afrat)

When the igniter pressure *Pign* is above a predefined value, meaning igniter is firing, igniter mass is taken. And the thermodynamic parameters are calculated by weighted average of the air-fuel mixture and igniter.

M_tot_ign = m_tot + m_ign (total mass)

C_pt_ign = (m_tot*C_p + m_ign*C_p_ign) / (m_tot + m_ign)

Rt_ign = (m_tot*R + m_ign*R_ign) / (m_tot + m_ign)

C_vt_ign = C_pt_ign - Rt_ign

γt_ign = C_pt_ign / C_vt_ign

Tot_ign = (m_tot*C_p*T_0 + m_ign*C_p_ign*T_0_ign) / (m_tot + m_ign)

γ = γt_ign C_p = C_pt_ign R = Rt_ign T_0= Tot_ign

In all cases where combustion doesn't happen, ideal gas values are taken

γ=1.4 C_p=1005 $kJ/kg.K$ R=287 $J/kg.K$ T_0=TS1(still chamber temperature)

And then, using the thermodynamic parameters (12), the Γ (8) is calculated using which the theoretical C^* (7) is calculated as:

$$\boldsymbol{C^*}_{\boldsymbol{th}} = \frac{\sqrt{\boldsymbol{R \times T_0}}}{\boldsymbol{\Gamma}} \quad ...(13)$$

Experimental C^*

Total pressure inside combustor

Combustor mach number can be found out using the coefficient of discharge of the nozzle (9),cross sectional areas in the combustion chamber A_c and the nozzle throat A^* which are related to the mach number in the combustor by the following relation which would be solved iteratively to obtain its roots:

$$\frac{A_C}{A^* C_D} = \frac{1}{M}\left(\frac{2}{\gamma + 1} + \frac{\gamma - 1}{\gamma + 1} M^2\right)^{\gamma+1/_{2(\gamma-1)}} \quad \dots (14)$$

With the solution M_{ccx} (14), the total pressure in the combustor can be found using the achieved static pressure *PR3* as:

$$P_0 cn\_cx\_exp = PR3 \times \left(1 + \frac{(\gamma - 1)}{2} \times M_{ccx}{}^2\right)^{\gamma/_{\gamma-1}} \quad \dots (15)$$

Then the experimental C* (1) can be calculated, using total pressure (15) and mass flows of air (10) and fuel (11), as:

$$C^*{}_{exp} = \frac{P_0 cn\_cx\_exp \times C_D \times A^*}{\dot{m}_a + \dot{m}_f} \quad \dots (16)$$

Now the combustion efficiency can be calculated using experimental C^* (16) and theoretical C^* (13) as:

$$\xi_b = \frac{C^*{}_{exp}}{C^*{}_{th}}$$

Calculating the Thrust and Specific Impulse

Thrust

Inlet air

Rate of change of inlet air (10) momentum, of density 1.544Kg/m^3 as per INSAT data for ambient air at 10m above sea level, is calculated as:

$$Inlet = \dot{m}_a^{\,2} / (1.544 \times Air\ inlet\ area) \qquad ...(17)$$

Exhaust

Force generated by exhaust is balanced by the normal reaction produced by the test stand which is read by the load cells (3).

$$Exhaust = 9.81*(MLC + restraint - (RLC + LLC) \qquad ...(18)$$

Where *restraint = 0.02* MLC*

Thrust generated would be the net of inlet (18) and exhaust (19) forces:

$$\boldsymbol{Thrust = Exhaust - Inlet} \qquad ...(19)$$

Specific impulse

Specific impulse is defined as thrust generated per unit weight of fuel consumed.

$$\boldsymbol{I_s = Thrust / Weight\ of\ fuel\ consumed} \qquad ...(20)$$

This (20) is calculated from the thrust (19) and fuel consumed (11) as:

$$\boldsymbol{I_s = Thrust / (g \times \dot{m}_f)}$$

Factors affecting combustion efficiency

Sauter Mean Diameter (SMD)

It represents the diameter of a sphere whose ratio of volume to surface area is the same as that of the entire droplet sample. Analyses showed that for combustion applications only SMD properly indicates the fineness of a spray, thus SMD is to be used to describe atomization quality.

It is usually represented by $\boldsymbol{D_{32}} = \dfrac{\sum N_i D_i^3}{\sum N_i D_i^2}$

Influences of injection pressure and ambient gas density on the SMD and air entrainment were investigated parametrically. An empirical equation for the SMD was proposed based on a dimensionless analysis of the experimental results. It was indicated that the SMD decreases with an increase in injection pressure and a decrease in ambient gas density.

For Measurement purpose, usually non-imaging optical methods are followed, like light scattering or diffraction methods, to determine the droplet sizes and distribution.

Fuel Properties

Physical properties.

Additives may like boron or aluminum for extra energy but these solid particles take some initial energy to ignite and hence might be ejected before burning completely.

Density is very important because it defines the energy content in a fixed volume of fuel.

Vapor pressure A high vapor pressure is desirable, because it ensures rapid evaporation of fuel in the primary combustion zone.

Volatility of a fuel may be assessed from a knowledge of its distillation range, vapor pressure, and flash point. Increased volatility provides easier light-up, improved stability, and higher combustion efficiency. These advantages are most apparent when combustion performance is limited by poor fuel atomization.

Viscosity depends mainly on the chemical composition of the hydrocarbons contained in the fuel. Apart from affecting the power required to pump the fuel through the fuel system, viscosity has marked effect on the formation of a well-atomized spray and hence on the rates of fuel evaporation and combustion. The higher the viscosity of a fuel, the poorer the quality of atomization.

Surface tension has a significant effect on fuel atomization, for both pressure and air-blast atomizers.

It has been observed that the liquid or gaseous phase of the fuel has little effect in reducing the combustion instabilities.

Combustion properties

Calorific value of a fuel is a measure of the heat liberated when it is burned to completion under standard conditions.

Enthalpy of a fuel is a measure of its capacity to absorb heat; the enthalpy also indicates the amount of heat required to accomplish a given change in the temperature or state of a fuel.

Spontaneous Ignition Temperature is the tendency of the fuel toward spontaneous ignition which is an important factor affecting fire and explosion hazards.

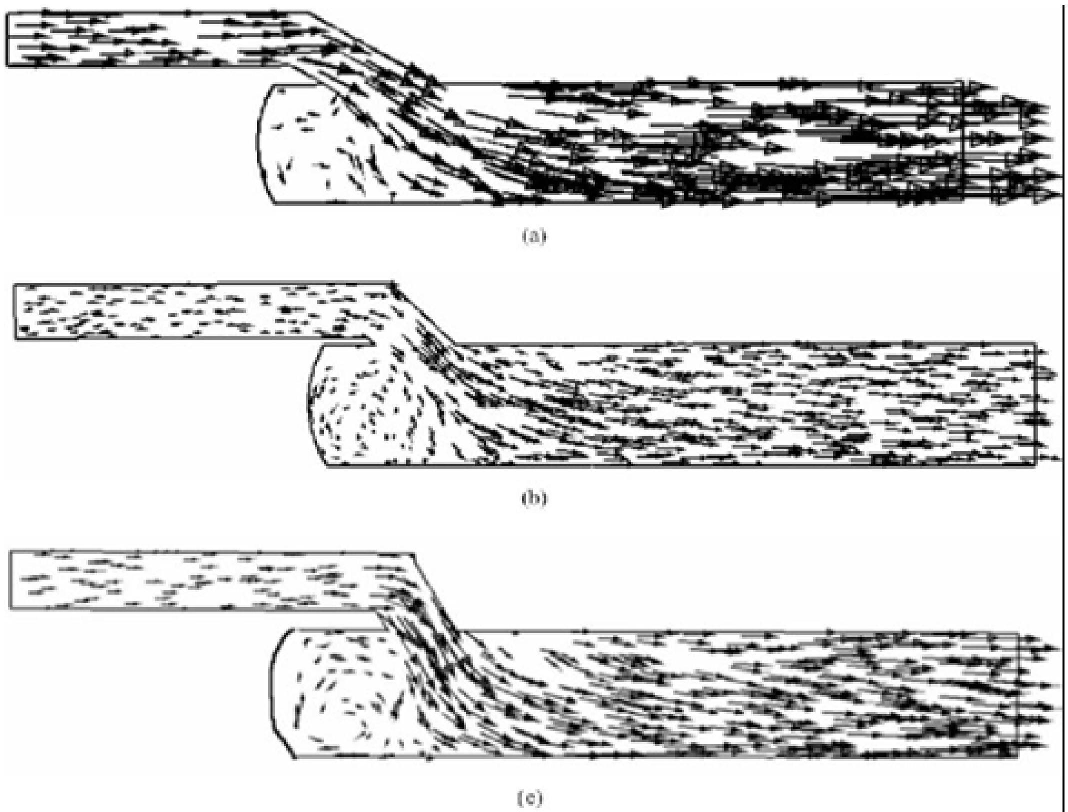
Flammability limits - Combustible gases and vapors are capable of burning in air only within closely defined limits of concentration. Within these limits, a flame, once initiated, can spread any distance away from the source of ignition. It is customary to define rich and lean flammability limits, which respectively represent the maximum and minimum fuel concentrations that produce combustible mixtures.

Evaporation rates are dependent on fuel volatility, and on atomization quality that determines the surface area of the atomized fuel. Maximum rates of evaporation are achieved with fuels of low viscosity and high volatility.

Air injection geometry

Depending on how and where on the combustor the air is dumped, the mixing of the fuel will vary. Also the intake type and bend should be made taking into account the external aerodynamics and pressure loss in the bend. There is no perfect geometry that could through all the hindrances.

Adjoining is the velocity vector diagram for different dump angles in a combustor. Do notice the recirulation zones.



Combustor length

A sufficient combustor length is essential for flame stabilization and combustion to near completion while also avoiding combustion instabilities.

Then the length of the combustor in terms of residence time (for combustion) and velocity of flow

$$L = V_{ref} t_{res} = \frac{\rho_{t3} A_{ref} t_{res}}{\dot{m}}$$

Methods of improving combustion efficiency

Atomization.

Injectors can be designed in such ways as to make the fuel atomized finer and distributed as required.

Modeling.

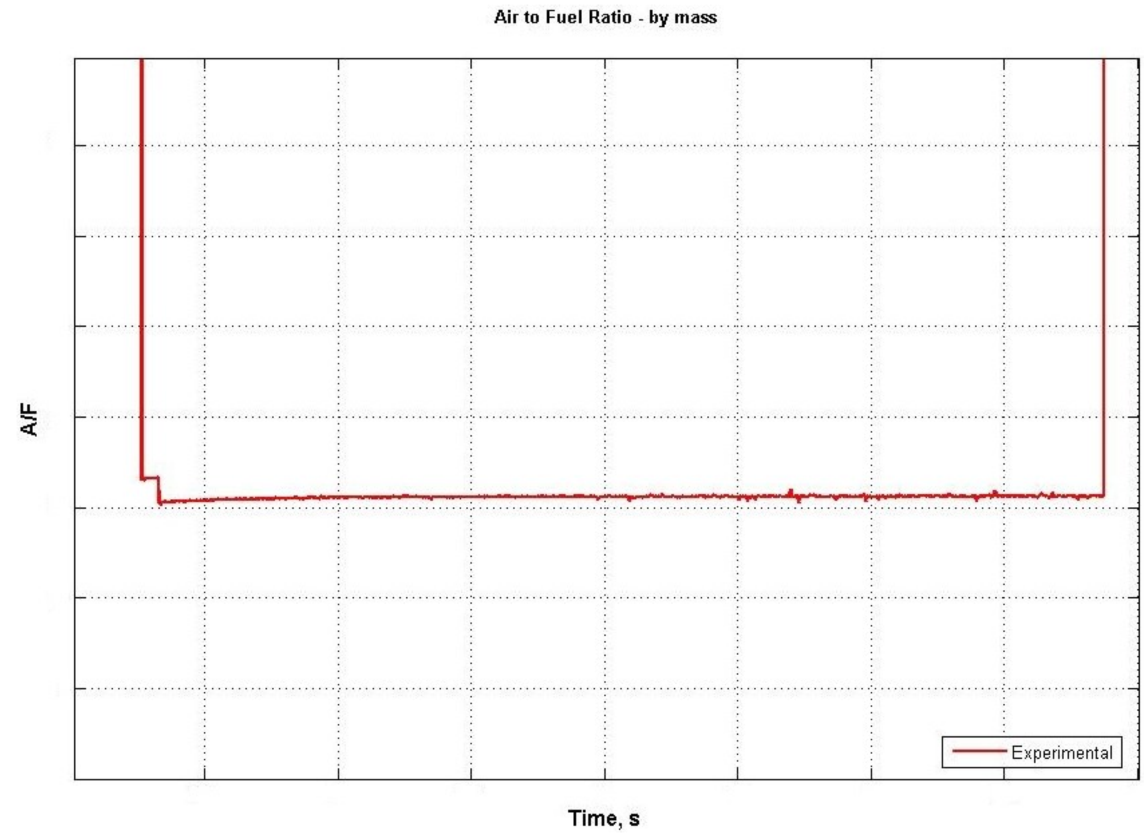
Modeling the complex phenomenon of the combustor not only helps in understanding the roles of various parameters, like air intake, etc, in varying the desired parameter but also helps in decreasing the cost of experimentation by reducing the number of tests required.

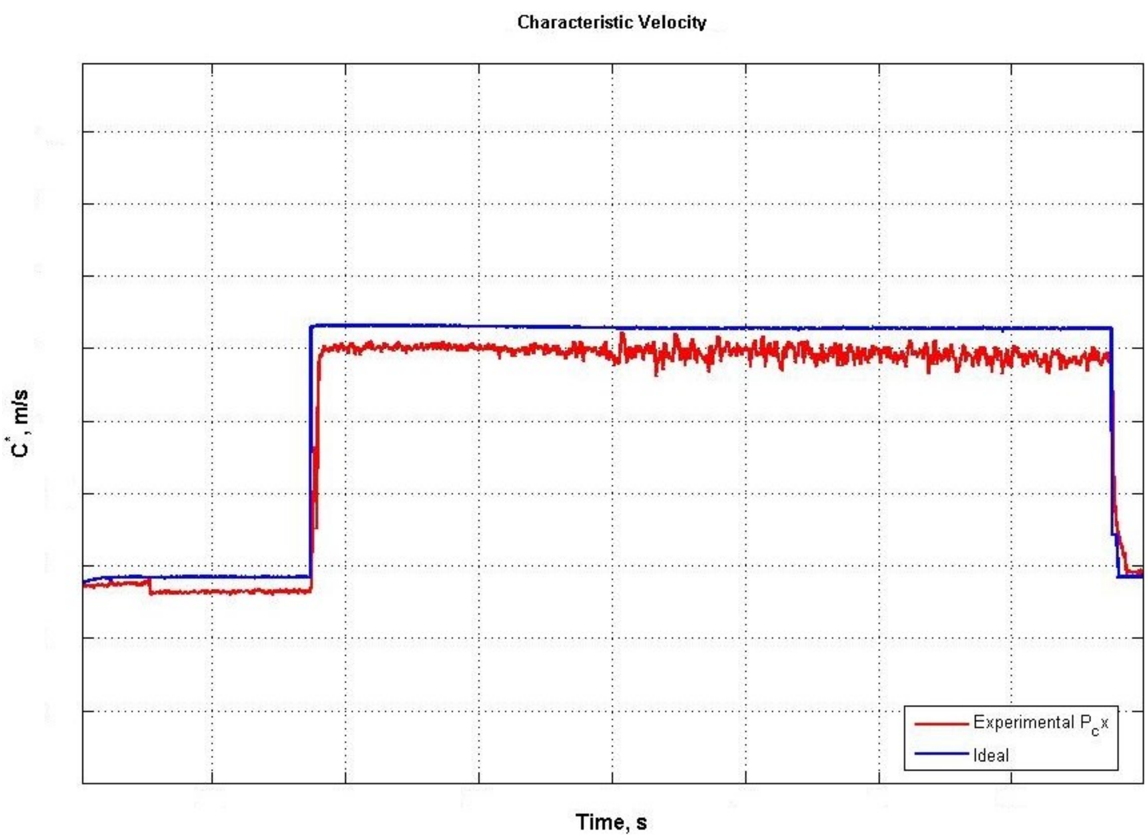
Subduing instabilities.

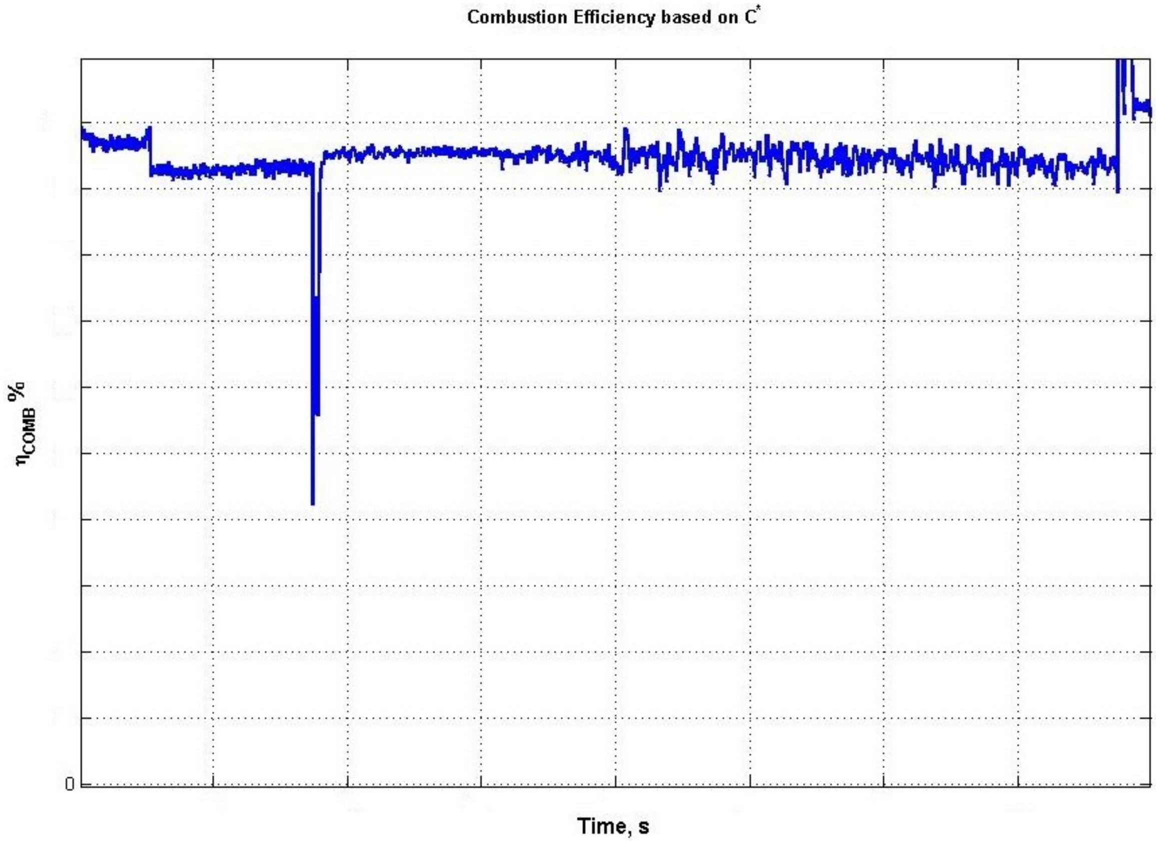
Are classified into low-frequency and high-frequency instabilities. Study has been on for years but they are not yet fully understood. Like air injection at the top of flame holders makes it possible to raise the equivalence ratio at which high-frequency instabilities appear. Or how the flow rate of fuel injectors can be varied to avoid the instabilities. Even longitudinal baffles in the combustor help. Hence tests are indispensable to know of the combustion instabilities.

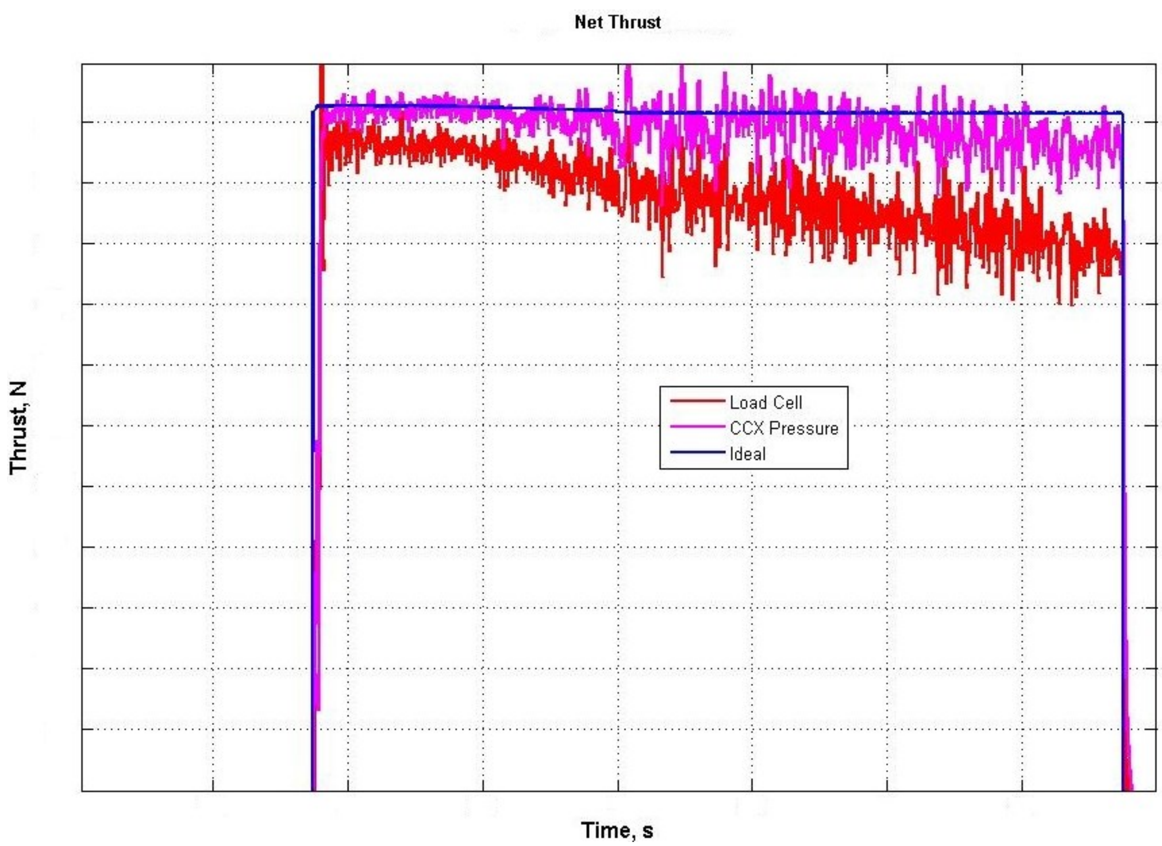
In spite of all the progress made so far in modeling, combustion tests on full scale are necessary to measure the pressure and combustion efficiencies, limits of combustibility and thrust generated at simulated flight conditions like mach number, angle of attack, altitude, etc.

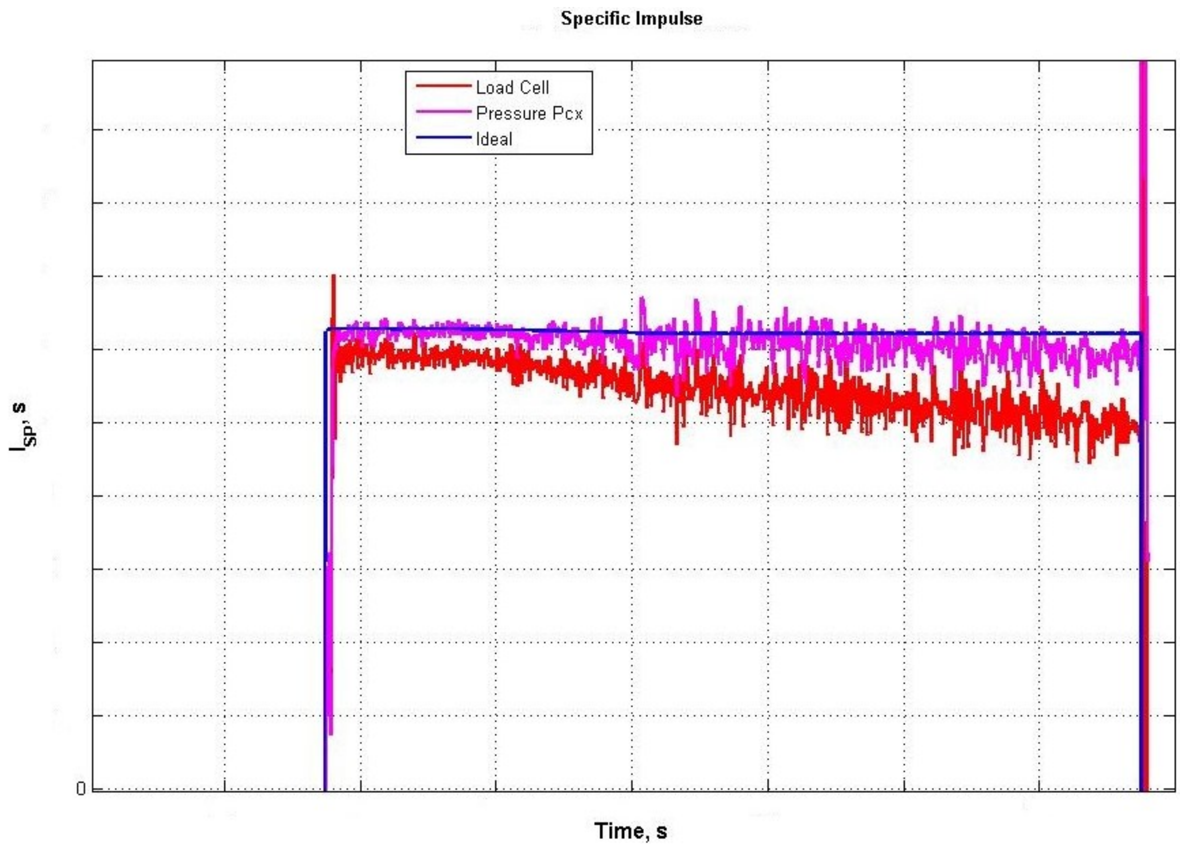
Analysis of test data and results






Combustion Efficiency based on C*
η_{COMB} %
0
Time, s


Net Thrust
Thrust, N
Load Cell
CCX Pressure
Ideal
Time, s



SUMMARY

Despite the slight compromise on the accuracy, assumptions made the analysis much simpler and brought forth the following observations.

- Good combustion efficiency of the order of 98% achieved was at an air fuel ratio of 15.6.
- Ignitability of ramjet engine even at low total temperature of was seen.
- Flame holding could be achieved at much higher air fuel ratios also.
- High C* close to the theoretical value was achieved.

FUTURE WORK

- These performance stats have to be recorded by doing another test at the same air-fuel ratio.
- Performance of combustor with the same fuel at various other air fuel ratios, namely, 21, 50 and 74, needs to be established by carrying out 3 numbers static tests at each air fuel ratio.

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Images

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ISO 5167-4 Measurement of fluid flow by means of pressure differential devices inserted in circular cross-section conduits running full.

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