Design and Analysis of Can Combustor

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Abstract—The design of combustion chamber is always less than exact science. This thesis documents the development of a design algorithm for a modern combustor. The algorithm includes a set of preliminary design procedures. This design model captures the Chemical reaction, Temperature distribution, Pressure variance, Velocity stream lines etc. The preliminary design procedures were verified using the advanced numerical techniques of computational fluid dynamics (CFD). These techniques are used to solve the swirling flow field inside the pre-mixer, the reacting flow field inside the liner, to understand the combustion phenomena and resulted emissions.

Keywords—Combustor, Eddy Dissipation Combustion Model, Air-Fuel ratio, Methane – Air Mixture, Gas Turbines, Combustion Aerodynamics, Combustion Emissions.

I. INTRODUCTION

Jet propulsion is a practical application of Sir Isaac Newton’s third law of motion which states that, ‘for every force acting on a body there is an opposite and equal reaction’. For aircraft propulsion, the ‘body’ is atmospheric air that is caused to accelerate as it passes through the engine. The force required to give this acceleration has an equal effect in the opposite direction acting on the apparatus producing the acceleration. A jet engine produces thrust in a similar way to the engine/propeller combination. Both propel the aircraft by thrusting a large weight of air backwards, one in the form of a large air slip stream at comparatively low speed and the other in the form of a jet of gas at very high speed. This same principle of reaction occurs in all forms of movement and has been usefully applied in many ways. Jet reaction is definitely an internal phenomenon and does not, as is frequently assumed, result from the pressure of the jet on the atmosphere. In fact, the jet propulsion engine, whether rocket or turbo-jet, is a piece of apparatus designed to accelerate a stream of air or gas and to expel it at high velocity. The engine is proportional to the mass or weight of air expelled by the engine and to the velocity change imparted to it. In other words, the same thrust can be provided either by giving a large mass of air a little extra velocity or a small mass of air a large exit velocity. In practice the former is preferred, since by lowering the jet velocity relative to the atmosphere a higher propulsive efficiency is obtained.

The types of jet engine, whether ram jet, pulsejet, rocket, gas turbine, turbo/ram jet or turbo-rocket, differ only in the way in which the ‘thrust provider’, or engine, supplies and converts the energy into power for flight. When forward motion is imparted to it from an external source, air is forced into the air intake where it loses velocity or kinetic energy and increases its pressure energy as it passes through the diverging duct. The total energy is then increased by the combustion of fuel, and the expanding gases accelerate to atmosphere through the outlet duct.

The mechanical arrangement of the gas turbine engine is simple, for it consists of only two main rotating parts, a compressor and a turbine and one or a number of combustion chambers this simplicity, however, does not apply to all aspects of the engine. They result from the high operating temperatures of the combustion chamber and turbine, the effects of varying flows across the compressor and turbine blades, and the design of the exhaust system through which the gases are ejected to form the propulsive jet. The combustion chamber has the difficult task of burning large quantities of fuel, supplied through the fuel spray nozzles, with extensive volumes of air, supplied by the compressor and releasing the heat in such a manner that the air is expanded and accelerated to give a smooth stream of uniformly heated gas at all conditions required by the turbine. This task must be accomplished with the minimum loss in pressure and with the maximum heat release for the limited space available. The amount of fuel added to the air will depend upon the temperature rise required. However, the maximum temperature is limited to within the range of 850 to 1700 deg. C. by the materials from which the turbine blades and nozzles are made. The air has already been heated to between 200 and 550 deg by the work done during compression, giving a temperature rise requirement of 650 to 1150 deg. C from the combustion process. Since the gas temperature required at the turbine varies with engine thrust, and in the case of the turbo-propeller engine upon the power required, the combustion chamber must also be capable of maintaining stable and efficient combustion over a wide range of engine operating conditions. Efficient combustion has become increasingly important because of the rapid rise in commercial aircraft traffic and the consequent increase in atmospheric pollution, which is seen by the general public as exhaust smoke.
The study of air-fuel ratio, swirler angle at air inlet axial position of holes are changed to investigate the effects of parameters on combustion chamber performance and emissions. The outcome of results will help in finding the correct geometry for a particular operation.

II. DESIGN OF CAN COMBUSTOR

Combustor sizing refers to the definition of the reference area (or diameter) to provide sufficient stability without incurring excessive pressure losses. All operating points (i.e., idle, full power) are considered and the smallest size that provides stability over the entire range of operation is chosen. Inlet temperature as 480K, mass flow rate 1.3 kg/s with pressure 3.6 bar.

Area reference \( A_{\text{ref}} = \frac{\text{Re}}{2} \left( \frac{m_3}{\text{Ps}_{\text{gas}}} \right)^{\frac{2}{3}} \left( \frac{\text{Ps}_{\text{gas}} - 1}{\text{Ps}_{\text{gas}}} \right)^{-\frac{1}{3}} \times 0.5 \)

Diameter reference \( D_{\text{ref}} = \sqrt{\frac{4}{\pi}} A_{\text{ref}} + 2t \text{liner} \)

Area of Liner = 0.7 * \( A_{\text{ref}} \)

Diameter of Liner = \( \sqrt{(4/\pi)^{1/4}} A_{\text{liner}} \)

Diameter of Liner \( L_{\text{dome}} = \frac{D_{\text{liner}} - D_{\text{swirl}}}{2} \times \tan \theta \)

Can Combustor diameter is 479 mm in the Y direction outlet diameter of combustor is 143 mm. Area of air Inlet is 0.0142922 \( \text{mm}^2 \), Area of Fuel Inlet is 0.0159984 \( \text{mm}^2 \) (for 5 holes), where diameter of each fuel Inlet hole will be 6.25 mm. Swirler length and diameter has to match to diameter of air inlet, where swirler angle blade geometry considered as 45°.

III. MODELING, MESHING AND BOUNDARY CONDITIONS

A. Eddy Dissipation Combustion Model

In Eddy Dissipation Combustion chemical reaction takes place very fast in molecular level relative to transport processes to the flow. When reactants mix at lower molecule level, instantaneously form products. The model assumes reaction rate is directly proportional to time required to mix reactant. In case of turbulent flow reaction rate is directly proportional to Kinetic Energy and Kinetic dissipation. In this model mixing rate is dominated by eddy properties.

B. Meshing

The partial differential equations which give solutions for fluid flow and heat transfer is not simple. To achieve solution domains are splitted into subdomains. The governing equations are then discretised and solved inside each subdomains. For this thesis work mesh adaption has done with tetra hedral and prism elements at boundary layer. The mesh generated automatically with 431389 elements and 88982 nodes.

C. Boundary Conditions

Boundary condition can be vary for air inlet and fuel inlet, since the relation depends upon the Air-fuel mixture ratio. The boundary conditions for air inlet are: mass flow rate is 0.2636 kg/s, total temperature of 300 K, component as oxygen with flow normal to the direction, low turbulence intensity and eddy viscous ratio. The boundary conditions for fuel inlet are: \( \text{CH}_4 \) mass fraction as 1, mass flow rate for fuel is 0.00352 kg/s, Total Temperature 300K with flow normal to the direction, medium intensity and Eddy viscosity ratio. The boundary condition of the outlet of combustion is defined by zero pressure value. Heat transfer will set to adiabatic with no slip wall condition.

IV. RESULTS AND DISCUSSION

A. Combustor Outlet Temperature

The contour of predicted gas temperature for the methane combustion is 1361 K which shows effective combustion between air and methane fuel mixture. The theoretical temperature flame produced from natural gas or methane fuel with atmospheric conditions (1bar, 200C) is 1950K with fast combustion rate.
The maximum outlet temperature can be found by using the technical formula:

\[(Ma \times Cp \times Tair) + (Mf \times Hf) = (Ma + Mf) \times Cpg \times Texit\]

Where
- \(Ma\) = Mass of air Inlet
- \(Mf\) = Mass of fuel Inlet
- \(Tair\) = Temperature of air inlet
- \(Hf\) = Calorific value of methane fuel
- \(Cpg\) = Const. pressure of gas

\[(0.2636 \times 1008 \times 300) + (0.00352 \times 55000) = 0.26712 \times 1.182 \times Texit\]

Total Temperature at Exit = 864.9 K (Manually)

Total Temperature at Exit = 949.6 K (Software)

Since software calculation is approximate to manual calculation, the both values reached nearby condition.

### B. Pattern factor

Modeling of the jets issuing from liner holes is essential to their design. Modeling of the jet trajectory is essential to ensure that good mixing and a suitable exit pattern factor are provided. Temperature variance should be linearly increase in radial direction. The pattern factor is defined as (Lefebvre, 1999).

\[
\text{pattern factor} = \frac{\max \text{ record exhaust temp} - \text{mean exhaust temp}}{\text{mean exhaust temp} - \text{inlet air temperature}}
\]

\[
= \frac{949.60 - 935.663}{935.663 - 300}
\]

\[
\text{Pattern factor} = 0.02
\]

Holdeman et al. (1987) and Holdeman (1993) provided empirical correlations for the determination of the jet centerline trajectory, jet temperature profile, and the jet width for confined ducts with single-sided and opposed rows of jets. These models correspond well with experimental and numerical data. Much of the work on single and multiple jets in a confined cross flow has been summarized by Lefebvre (1999).

### C. Pressure Loss factor

The difference in total pressure between the inlet and outlet of the combustor, called the overall total pressure loss together with the reference velocity head determine the size of the combustor. The quantity is of great importance to combustor design and are generally quoted pressure loss factor.

\[
\text{Pressure Loss} = \frac{\text{Total air pressure Inlet} - \text{Total air Pressure Outlet}}{\text{Total air Pressure Inlet}} \times 100
\]

\[
= \frac{102105.314 - 101690.475}{102105.314} \times 100
\]

\[
= 0.4\% \text{ Pressure Loss.}
\]

As we used the angle of swirler as 45° the zone covered by the primary air inlet of combustion chamber is more also a recirculation zone is ahead of swirler which helps in efficient combustion. The sudden rise in temperature observed near the tip of the injector indicates the generation of shocks which help in superior air-fuel mixing. Superior air-fuel mixing resulting in better quality of combustion and thus better performance. As predicted, the results obtained from this study show an enhanced air-fuel mixing and a proper combustion which can be attributed to the geometry of the ramp injector considered in this study.

Fig 4 shows the turbulent intensity is high in the immediate vicinity of the ramp injector indicating a superior air-fuel mixing.
A very high turbulent intensity indicates a superior air-fuel mixing. The high value of mass fraction of NO formed indicates an efficient combustion process. The peak gas temperature is located in the primary zone where combustion of mixture air and methane takes place. The fuel from 5 injectors is first mixed in the swirling air before burning in the primary reaction zone. The gas temperature decreases after the primary zone. In case there will be dilution holes are provided at dilution zone, to reduce the temperature this can be done when the temperature inside the combustor is high. The air required from dilution zone is get from compressor.

\[ \text{Total mass of the reactants equals the total mass of the products leading to the insight that the relations among quantities of reactants and products typically form a ratio of positive integers. This means that if the amounts of the separate reactants are known, then the amount of the product can be calculated.} \]

\[ \text{CH}_4 + 2 \text{O}_2 \rightarrow \text{CO}_2 + 2 \text{H}_2\text{O}. \]

V. CONCLUSION

Analysis on Can Combustors shows following results
1. For Methane as fuel and with initial atmospheric conditions, the flame temperature produced by flame with fast combustion was reached to 1361K which shows effective combustion.
2. Temperature profile at outlet of combustor shows radially increased, which states there will be less thermal stresses on the turbine blades which are located next to the combustor.

3. Outlet Temperature of the can combustor results are approximate equivalent to manual calculations, since the software calculation done by Finite Volume Method which is approx. to manual calculation method.

4. Pressure Loss factor determines the efficiency in combustion. Pressure loss factor should be minimum as possible, there by more pressurized gas will exists it can be determined from inlet and outlet pressures. In the model there is low pressure loss which shows effective pressure in the gas at the exit of the combustor.

REFERENCES