Numerical Simulation of Gas Turbine Can Combustor Engine

CH UMAMAHESHWAR PRAVEEN[1*], A HEMANTH KUMAR YADAV[2]

1. Engineer, CDG BOEING Company, Chennai, India.
2. B.Tech Aeronautical Engineer 2012 passout, Bharat Institute of Engineering and Technology, Hyderabad, India.

*email: chpraveen52@gmail.com, *Ph: +91-09677143692

Abstract— In the Gas Turbine combustion system, the flow into combustor play one of the key roles in controlling pressure loss, air flow distribution around the combustor liner, and the attendant effects on performance, durability, and stability. This paper produces design methodology and describes a computational fluid dynamics (CFD) simulation of the flow in the can combustor. The main aim is to study and examine the performance influence of turbulence models, combustion models and rate mechanism on the prediction of the chemical species concentration and temperature fields which are useful to design the combustor within design limits. Numerical simulations are performed to analyze the combustion behavior in three-dimensional combustor model by solving swirl flow field inside pre-mixer, reacting flow field inside the liner, to understand the combustion phenomena and resultant emissions.

Keywords— Gas Turbine Engines, Combustion Chamber, Combustion Aerodynamics, Stoichiometric ratio, Combustion Models, Turbulence Models, NO_X formation, Non-Premixed Combustion, Combustion Chemistry.

INTRODUCTION

The development of the gas turbine engine as an aircraft power plant has been so rapid that it is difficult to appreciate that prior to the 1950s very few people had heard of this method of aircraft propulsion. The possibility of using a reaction jet had interested aircraft designers for a long time, but initially the low speeds of early aircraft and the unsuitably of a piston engine for producing the large high velocity airflow necessary for the 'jet' presented many obstacles. 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. C. by the work done during compression, giving a temperature rise requirement of 650 to 1250 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.

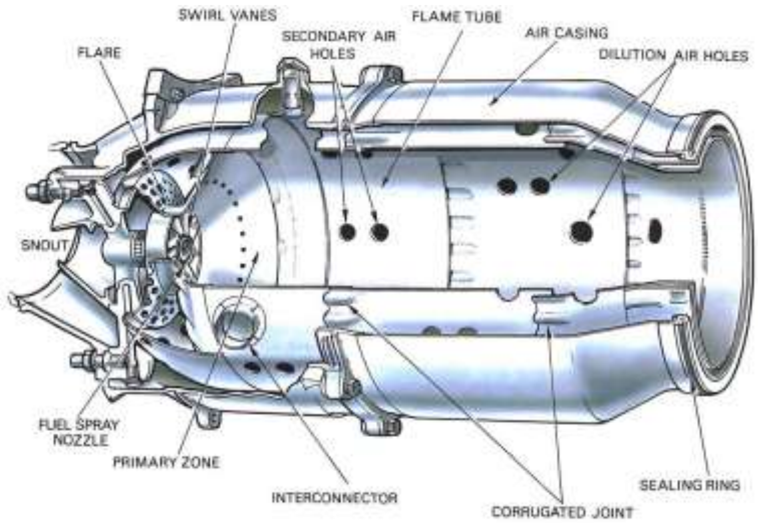

Figure 1: Can combustor

The study of air-fuel ratio, swirler angle at air inlet, axial position of holes is 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.

DESIGN OF 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. For simplicity, the static conditions are assumed equal to those at stagnation for both temperature and pressure. This is a reasonable assumption since the Mach number everywhere is below 0.25. For a Mach number of 0.25, the static pressure and temperature deviate from stagnation by only 4 percent and 1 percent, respectively.

Design point of view for reference:

Maximum Inlet temperature 480K, mass flow rate 1.3 kg/s with pressure 3.6 bar.

Thus, for each point the following conditions are known:

• The air flow rate, m_3

• The inlet temperature, T_3

• The outlet temperature, T_4

• The inlet pressure, P_3

And from the ideal gas law, we know the inlet density,

$$\rho_3 = \frac{P3}{Ra * T3}$$

Where Ra is the ideal gas constant for air.

The casing size for most combustors is dictated by the overall pressure loss ΔP_{3-4}.

The cross-sectional area of the casing, A_{ref}, is determined from

$$A_{ref} = [\frac{Ra}{2}\left(m3 * \frac{T3^{0.5}}{P3}\right)^2 * \left(\frac{\Delta P3-4}{P3}\right)^{-1}]^{\wedge}0.5$$

Suitable values for $\Delta P_{3-4}/P3$ and $\Delta P_{3-4}/q_{ref}$ are chosen by the designer based on
The casing diameter for a can may then be derived from geometry.

$$D_{ref} = \sqrt{(4/\pi)Aref + 2t\ liner}$$

The liner diameter must be chosen carefully. It would appear desirable to select the largest diameter possible and reduce the velocity of the flow within the liner. This increases the residence time in the combustor and promotes stability. However, for any given casing area, increasing the liner diameter results in a reduction of the annulus flow area A_{an}. The smaller area results in higher annulus velocities that decrease the static pressure in the annulus. Therefore, a larger liner diameter is undesirable since a high static pressure drop across the liner admission holes is necessary to provide adequate penetration of the jets. Sawyer (1985) provided a general rule of thumb for selecting the liner diameter of conventional combustors. The author stated that the liner cross-sectional area should be kept within 60-72 percent of the casing area for conventional combustors. A value of 70 percent is chosen and hence, the liner cross-sectional area is

$$A_{liner} = 0.7*A_{ref}$$

The diameter of the liner is

$D_{liner} = \sqrt{(4/\pi) * Aliner}$

and the annulus cross-sectional flow area is

$A_{an} = [D^2ref - (D_{liner} + 2t_{\ liner})^2]* \frac{\pi}{4}$

The liner diameter of the contracted section is

$\mathrm{Dliner,2} = \sqrt{0.7 * \mathrm{Dliner}}$

and the length of the contraction is

$L_2 = \frac{\mathrm{Dliner-Dliner,2}}{2\tan\theta}$

The wall angle contract is chosen to be 30 degree

To maintain a constant annulus area, the casing diameter shall be

$\mathrm{Dref,2} = \sqrt{(\mathrm{Dliner,2} + \ \mathrm{2tliner}\ + (4/\pi) * Aan)\text{^}2}$

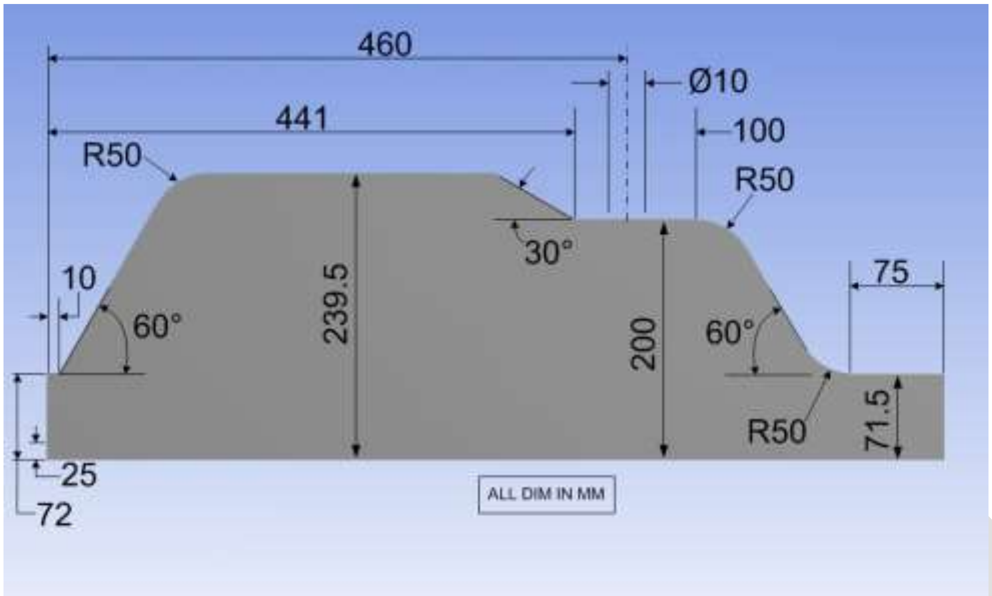

Figure 2: Combustor Dimensions

Can Combustor diameter is 479 mm in the Y direction outlet diameter of combustor is 143 mm, Area of air Inlet is 0.0142922 mm^2, Area of Fuel Inlet is 0.0159984 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 45^0

MODELLING, MESHING AND BOUNDARY CONDITIONS

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.

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 discretized and solved inside each subdomain. For this thesis work mesh adaption has done with tetrahedral and prism elements at boundary layer. The mesh generated automatically with 431389 elements and 88982 nodes.

BOUNDARY CONDITIONS

Boundary condition can be varying 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 Inlet 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: CH_4 mass fraction as 1, mass flow rate for fuel is 0.00352 kg/s, Total Inlet 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. The finite volume Eddy dissipation combustion model is used with total energy heat transfer, thermal radiation set to P1, turbulence model with k-epsilon.

RESULTS

COMBUSTOR OUTLET TEMPERATURE

The contour of predicted gas temperature for the methane combustion is 1271 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 finding out by using technical formula:

$(M_a * Cp * Tair) + (M_f * H_f) = (M_a + M_f) * C_{pg} * T\ exit$

Where

M_a = Mass of air Inlet

M_f = Mass of fuel Inlet

Tair = Temperature of air inlet

H_f = Calorific value of methane fuel

C_{pg} = Const. pressure of gas

(0.2636*1005*300) + (0.00352*55000) = 0.26712*1.182* T exit

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.

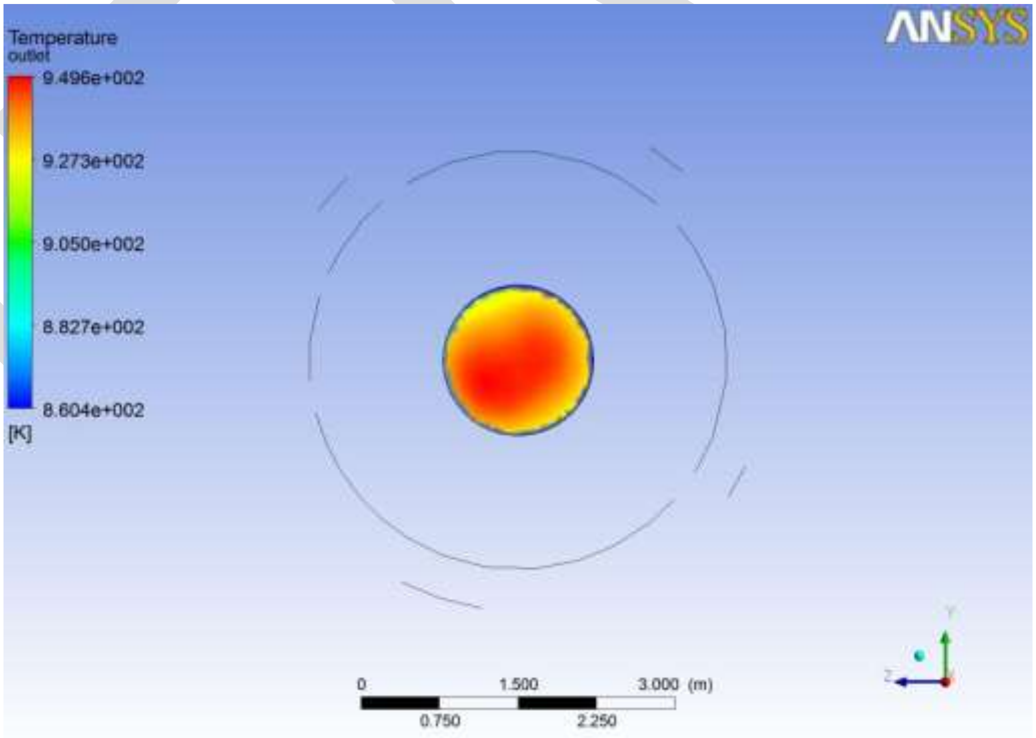

Figure 3: Temperature profile on outlet

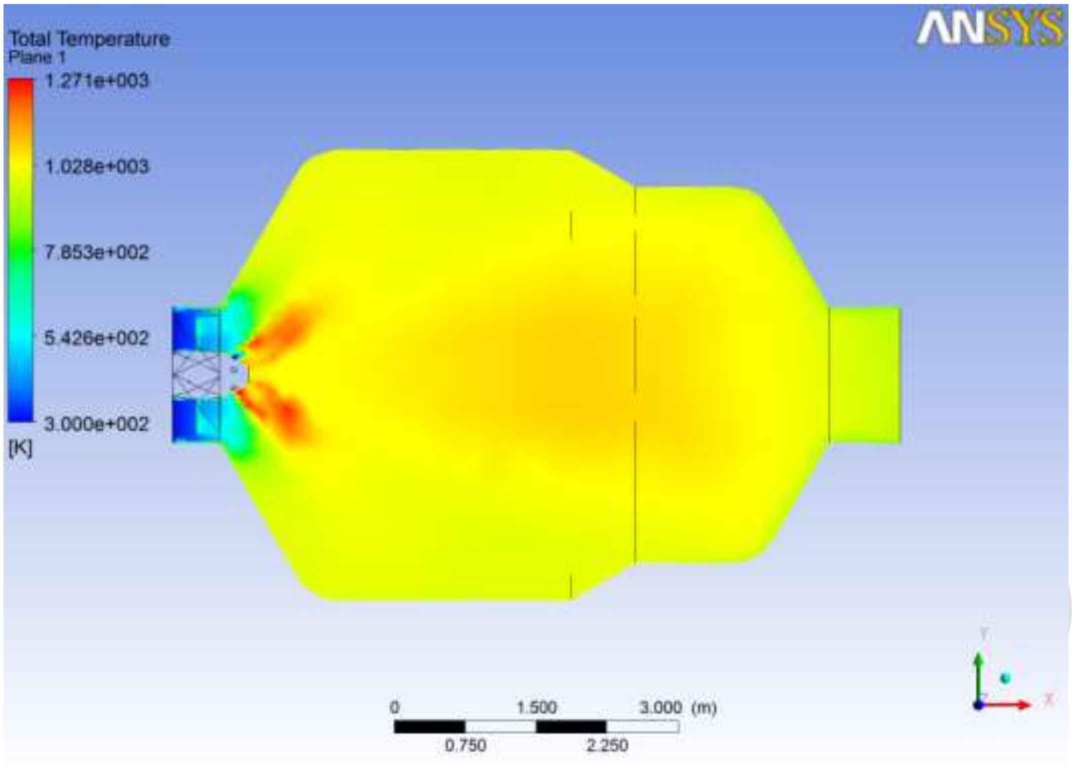

Figure 4: Temperature plane on longitudinal plane

PATTERN FACTOR

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

Pattern factor = $\frac{\text{Max}\,\textit{record Exhaust Temperature} - \textit{Mean Exhaust Temperature}}{\textit{Mean Exhaust Temperature} - \textit{Inlet air Temperature}}$

$$= \frac{949.60 - 935.663}{935.663 - 300}$$

= 0.02

Holdeman et al. (1987) and Holdeman (1993) provided empirical correlations for the determination of the jet centreline 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). Pattern factor should be less than 0.3 as per design standards. The result of pattern factor lies within the design limit.

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. The Pressure loss should not be greater than 8% including factor concerns which results in lack of Performance for jet Engine.

Pressure loss percentage = $\frac{\textit{Total air pressure Inlet} - \textit{Total air pressure Outlet}}{\textit{Total air Pressure Inlet}} * 100$

Pressure loss Percentage = $\frac{102105.314-101690.475}{102105.314} * 100$

= 0.4% Pressure loss

As we used the angle of swirler as 45^{o} 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 & 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.

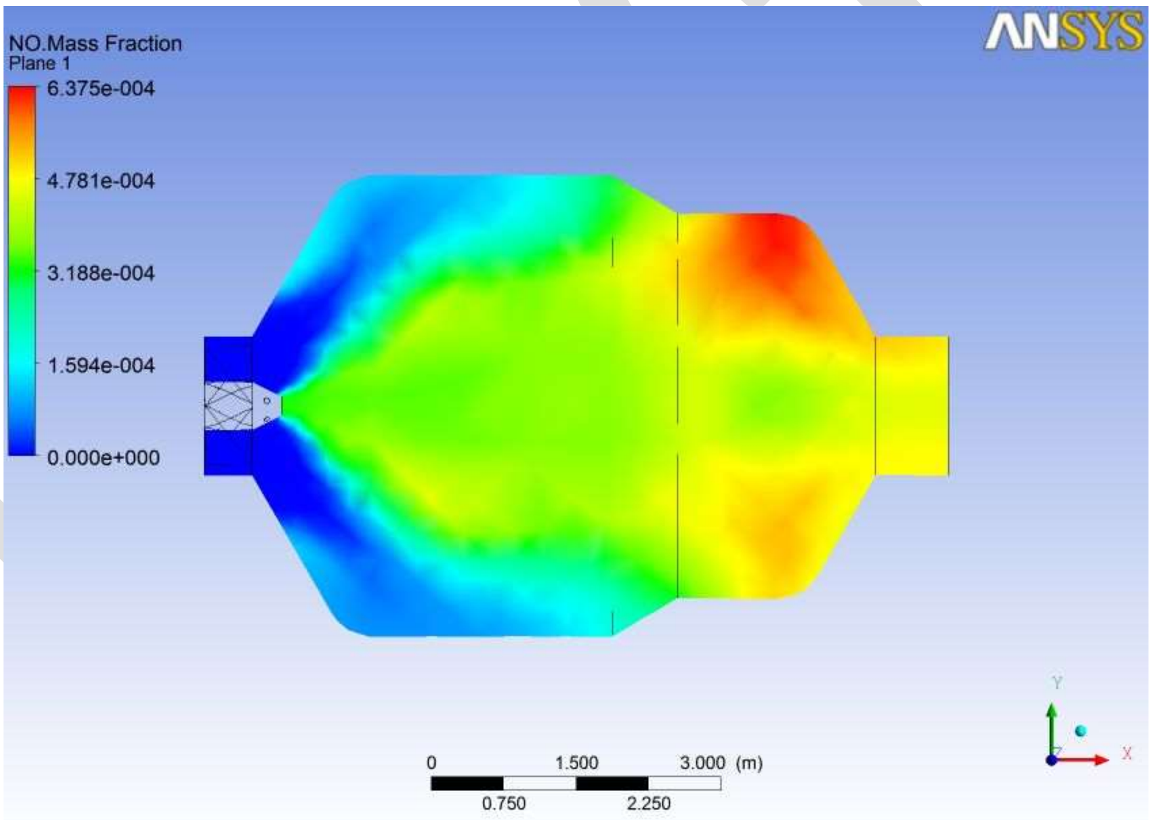

Figure 5: NO Mass fraction.

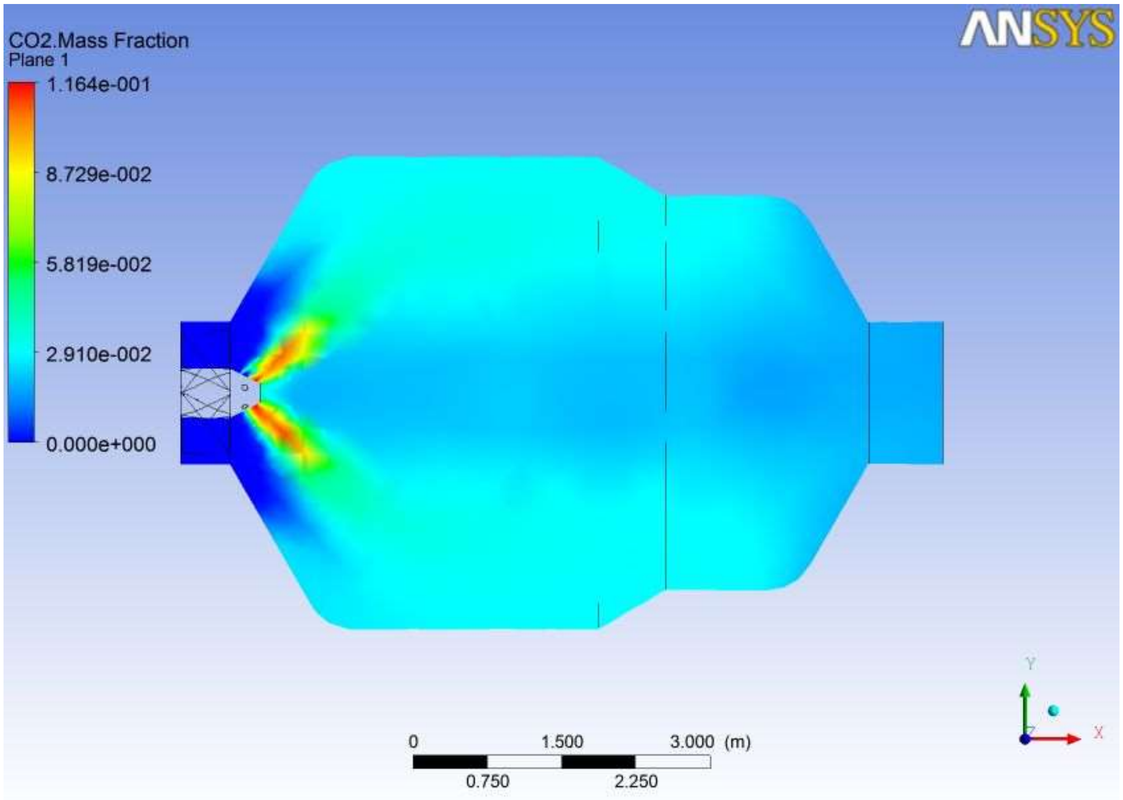

Figure 6:CO_2 Mass Fraction

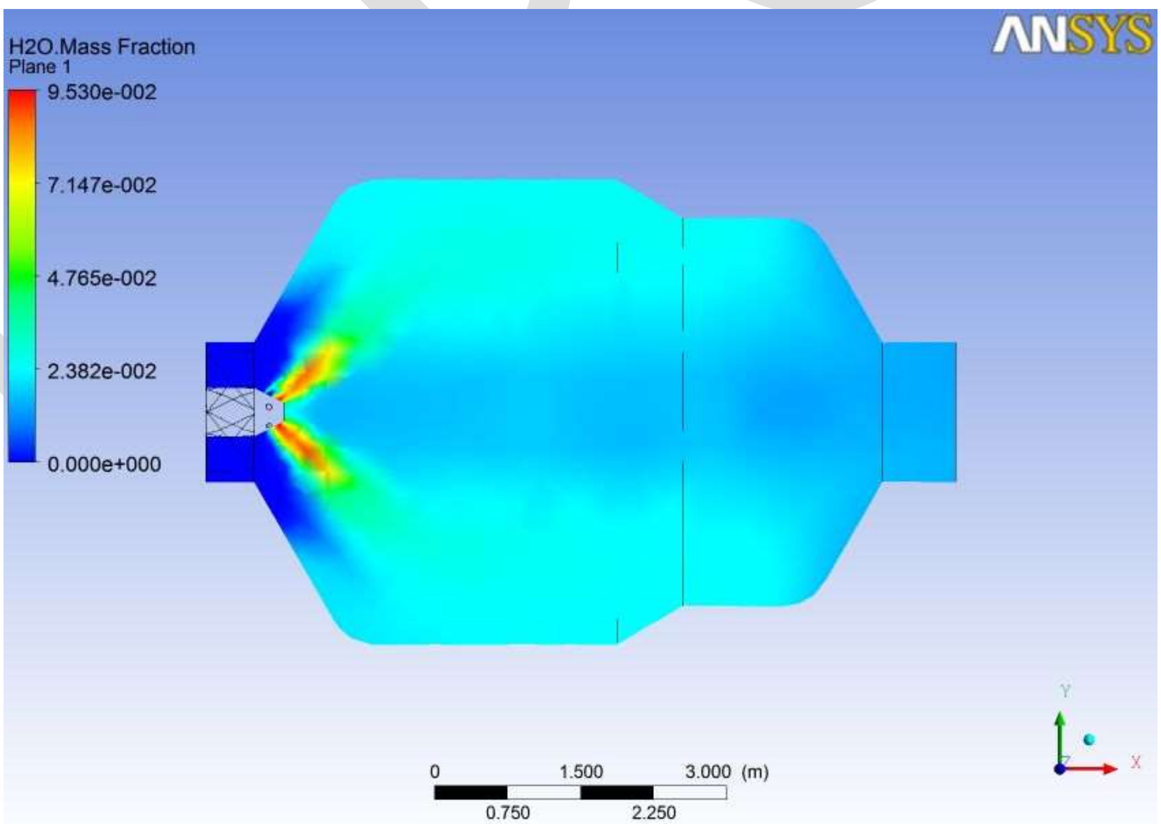

Figure 7:H_2O Mass Fraction

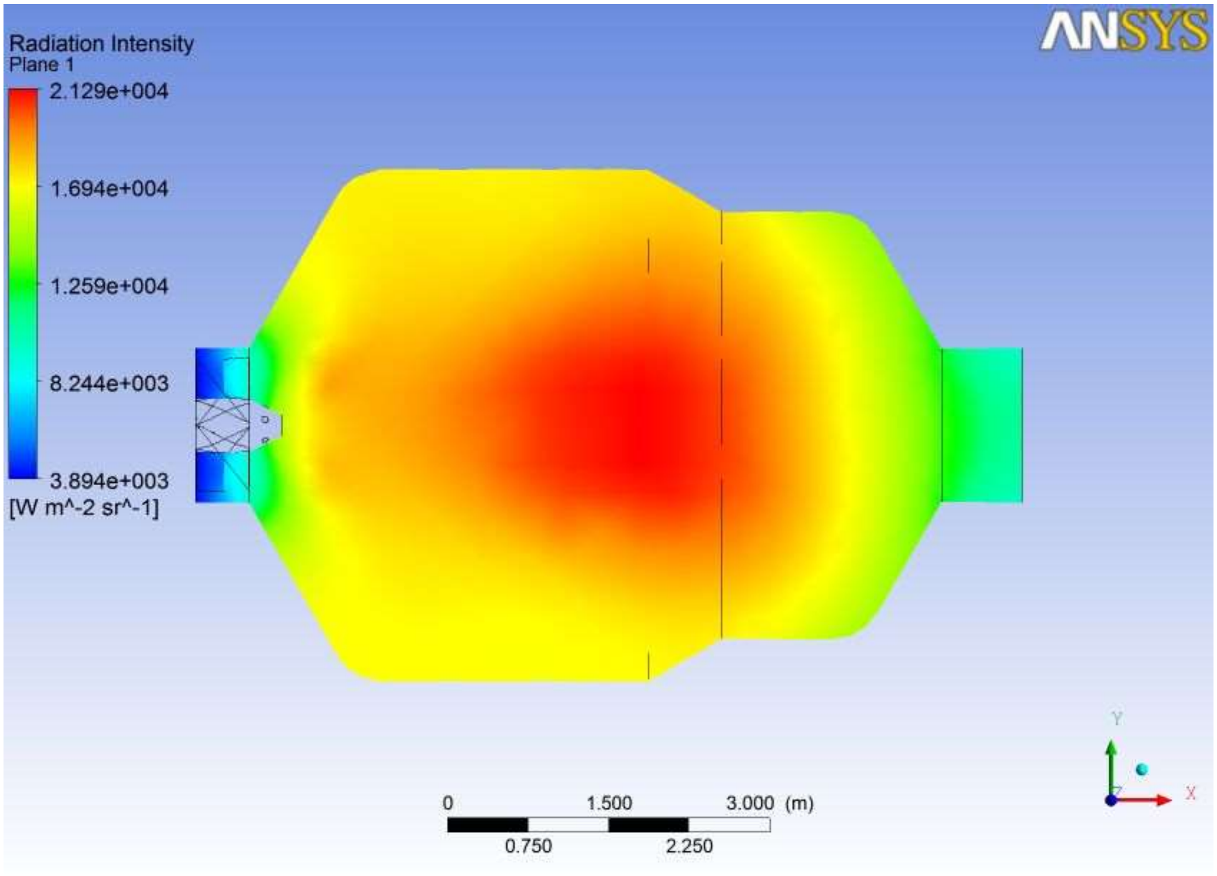

Figure 8: Radiation Intensity

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.

$$CH_4 + 2\ O_2 \rightarrow CO_2 + 2\ H_2O$$

CONCLUSION

Analysis on Can Combustors shows following results which are in design limits thereby Air-fuel ratio 75 was acceptable.

For Methane as fuel and with initial atmospheric conditions, the flame temperature produced by flame with fast combustion was reached to 1271K which shows effective combustion.

1. 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.

2. Outlet Temperature of the can combustor result is approximate equivalent to manual calculations, since the software calculation done by Finite Volume Method which is apprx. to manual calculation method.

3. 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.

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