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Characterization of a Single Injector GH_2-LO_2 for a 20 bar Cryogenic Chamber Test Facility

Sreejesh S R Chandran
Research Scholar, Department of Mechanical Engineering,
Karpagam Academy of Higher Education, Karpagam University
Coimbatore, Tamilnadu, India.
Orcid Id: 0000-0003-0470-6046

N M Sivaram
Associate Professor, Department of Mechanical Engineering,
Karpagam Academy of Higher education, Karpagam University
Coimbatore, Tamilnadu, India.
Orcid Id: 0000-0003-0661-6677

T G Ansalam Raj
Principal, Valiya Koonambaikulathamma
College of Engineering and Technology, Kollam, Kerala, India.
Orcid Id: 0000-0002-5377-4734

K E Reby Roy
Professor, Department of Mechanical Engineering,
Thangal Kunju Musaliar College of Engineering, Kollam, Kerala, India.
Orcid Id: 0000-0003-2370-8522

Abstract

Now a days high performance liquid propellant rocket engines for transfer into orbit and space exploration have well-established cryogenic propellant combinations like liquid oxygen/gaseous hydrogen (LOX/GH2), due to their high specific impulse. In order to tackle complete combustion in rocket engines, the chamber temperature and pressure in LREs must be higher, which may exceed by far the safe operation temperature of all conventional liner materials currently in use. Wide range of heat fluxes are incident on the combustion chamber and the nozzle walls. To deal with such heat fluxes, different cooling techniques were employed, out of which a combination of film cooling and regenerative cooling are found to be effective one. The current study involves the numerical analysis of a rocket engine thrust chamber with a single coaxial injector which uses gaseous hydrogen as the fuel and liquid oxygen as the oxidizer with the help of CFD software package FLUENT. For the

GH2/LOX turbulent flame, the obtained results are compared with results from literature to verify the model and the numerical scheme. However a separate analysis is carried out to determine the optimum regenerative coolant mass flow rate and coolant initial temperature and it was found that the temperature of the combustion chamber outer wall reduces to an average value of 560 K. Therefore, in the present analysis, the outer wall temperature is taken as 560 K to bring in the effects of regenerative cooling of combustion in a typical film cooled cryogenic rocket engine thrust chamber considering the combustion of the fuel, heat transfer through the chamber walls and the fluid flow simultaneously.

Keywords: Computational fluid dynamics , Cryogenic rocket engine Combustion chamber, regenerative cooling, Film cooling.

INTRODUCTION

Hydrogen is considered as the most economic fuel in cryogenic rocket engine system because of its several advantages, when compared with hydrocarbons. It is because of its safety in transport, storage as well as high energy content.Two major research groups in Europe namely ONERA in France and DLR Lampoldshausen in Germany had experimentally investigated the CH4/O2 flames and H_2/O_2 flames:. Both ONERA's Mascotte test facility and the DLR's M3 combustion chamber test facility which initially were developed for experimental investigations of liquid oxygen/gaseous hydrogen combustion (LOx/H2) have been modified to allow for the study of LOx/CH4. In their studies, pressure ranges from 0.1 to 5.5MPa, the injection temperature for LOx is 85K, and for liquid CH4 it is 125K, hence both temperatures are cryogenic. A later study concerns fuel rich conditions. Candel et al. [5] studied the flame structure of both LOx/CH4 and LOx/H2 in the transcritical range for a pressure range between 0.1 to 7 MPa and a subcritical injection temperature of liquid oxygen. Cryogenic combustion of oxygen and methane injected at pressures between 4.5 and 6MPa were investigated experimentally studied by G. Singla, P. Scouflaire, C. Rolon, Candel through ONERA test faciity. The coaxial injector delivers oxygen at a temperature of 85K and methane at a temperature of 120K or 287K. Flame Stabilization of high pressure flames formed by cryogenic liquid oxygen/gaseous hydrogen or methane has been investigated through planar laser induced fluorescence (PLIF). In the LOx/GH2 experiments, injection conditions are transcritical and the chamber pressure (6.3MPa) is above the critical value infact the temperature (80K) is below the critical value. For LOx/GCH4 experiments, the chamber pressure (2MPa) and LOx injection temperature (80K) were very below critical values. In the DLR experiments [8], showed a comparison between spray combustion in two different coaxially injected CH4/LOx and H2/Lox chambers, at similar injection conditions was performed keeping Weber number and the momentum flux ratio at a varying rate. Introducing a coolant along the inside surface of the nozzle entrance in a convergent divergent rocket nozzle a was done experimentally by [J.L.Sloop and George.R.Kinney in (1948)] [4]. Characterization of different coolant rates in liquid rocket engine injectors has been studied by Richard Farmer, Gary Cheng &Yen-Sen Chen in a 2nd International Workshop on Rocket Combustion Modeling Germany 25-27 Mar 2001 [6]. A clear cut study of critical fundamental processes which are directly involved in the combustion of cryogenic propellants, namely liquid oxygen (LOX) and gaseous hydrogen (GH2) have been clearly depicted in simulation and development of MASCOTTE cryogenic combustion test facility by ONERA by [Habiballah, 1991] [2]. Three dimensional simulation of liquid oxygen gaseous hydrogen studying combustion , conjugate heat transfer had been done using LOX/GH_2 in a 10 bar Mascotte Test case by Aurelie nicole, Gerard Ordonneau & Mary Theron [1]

MODELING AND COMBUSTION

A computational model has been developed to analyse the combustion characteristics, conjugate heat transfer and cooling inside a hydrogen- oxygen combustion chamber of a cryogenic rocket engine. The selected geometry gave well accurate results compared with the former works.The combustor is properly converted to a grid and the governing equations and boundary conditions are solved using a CFD solver.Computational fluid dynamics analysis is successfully adopted to carry out the study of film cooling and regenerative cooling which can very effectively be considered to be most effective cooling techniques now a days in cryogenic engines. Here the injector head consists of a single coaxial injector in which liquid oxygen flows through the centre as oxidizer and the fuel as gaseous hydrogen flows through the annular space. The validation of the code is performed using a three dimensional analysis. Ansys Work Bench package is used for modeling and analysis.Then the two dimensional analysis of the chamber without film cooling is performed first and three cases namely 2.5%,3.5% and 4% with cooling are separately done. Two film coolant inlets where one is near the injector hole and the other near the throat of the nozzle is primarily used for present analysis and a third one may be taken to bring down the temperature to 560K to avoid melting of chamber wall..The above analysis is treated as an axisymmetric case to reduce ambiguity and time for massive computation. Because of attractive combustion characteristics and burning properties, hydrogen itself is serving as the fuel and the coolant. In the present case the adiabatic flame temperature , optimum coolant flow rate are having equal importance in maintaining necessary thrust in the rocket engine. Hydrogen being stored at cryogenic temperatures in the fuel tank generally pumped through the cooling channels before it is injected into the combustion chamber. Here the solid domain consists of copper as the material which cannot withstand temperatures above 560K. The results which are obtained proved to be the most effective one when compared with the results of previous works. From the analysis side, fine mesh with high smoothing is adapted for the geometry. Tetrahedral, hybrid type elements are selected for volume meshing the number of cells created in the entire computational domain 242387 after a detailed grid independence study is performed.The boundary conditions are defined (inlet, outlet and wall) along with the solid and fluid volumes. The meshed file is then analyzed using Fluent 6.3.26.The two dimensional geometry used for the analysis is shown below. Segregated solver is the type of solver used for the current work. The governing equations associated with the analysis are thereby solved sequentially in the segregated solver. k- ε model and P-1 radiation model are the turbulence models used and absorption coefficient based on weighted gray gas sum model is selected.. The Non-Premixed option is selected for combustion modeling available with the FLUENT. Mass flow rate is adopted at the inlet where pressure outlet is at outlet. The temperature of the outer wall for both the cases is assumed to be 560 to avoid melting of the wall.Fuel which (hydrogen) is entering through the inlet at 287K and the oxidizer enters the inlet at a temperature of 85K.

RESULTS AND DISCUSSION

The optimum and relevant results obtained from the analysis of the cryogenic thrust chamber with and without cooling techniques namely film and regenerative cooling are noted below using plots, contours and graphs. In addition to it, analysis showing optimal coolant flow rate which could maintain adequate thrust for the rocket is also narrated.Various flame behavior near the throat, injector side is also analysed. The analysis also noticed combustion instabilities during combustion.

Temperature Distribution in the thrust chamber without cooling

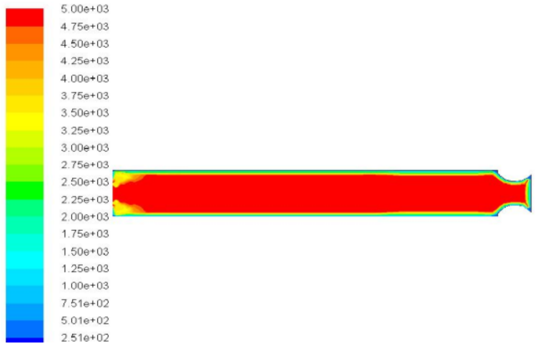
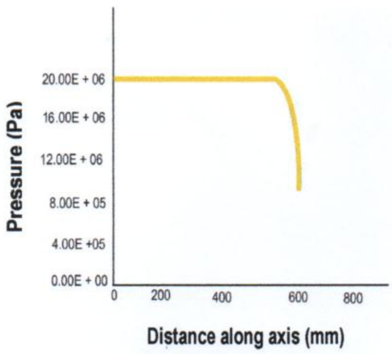


Fig 1

The temperature distribution inside the combustion chamber for 20 bar case without cooling is shown in fig 1. From the contour obtained , it is understood that combustion chamber temperature is getting boosted ,but it gets reduced due to expansion of the gases when approaches near the nozzle.The maximum temperature found, if coolant is not used for about 5000K is shown. For the case with 2.5% cooling ,the temperature gets reduced and found to be 3824 K and for 3.5% & 4% cases the subsequent temperature value after cooling is effected are found to be 3560K and 3540K. without film cooling, since we assumed outer wall temperature to be 560K, so as to avoid the melting of chamber wall, we can see temperature is decreasing when coolant rate is increased. If film cooling is imparted , the subsequent reduction in temperatures would be 3560K & 3540k respectively. The different plots showing the variation of various thermodynamic properties along the axial length of the cryogenic thrust chamber with and without cooling is shown below.



Variation of pressure along thrust chamber without cooling

Fig 2

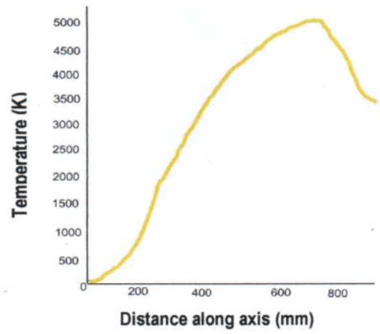


Fig 3

Temperature variation in the axial direction of chamber

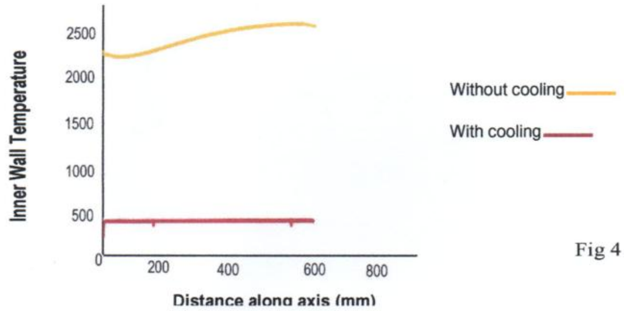


Fig 4

Temperature distribution along inner surface of thrust chamber

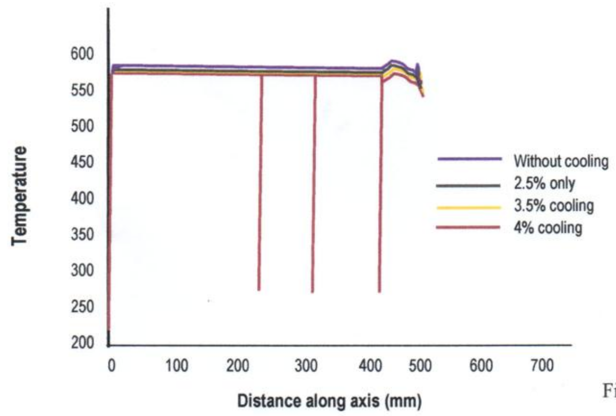


Fig 5

Comparison of temperature with & without cooling chamber

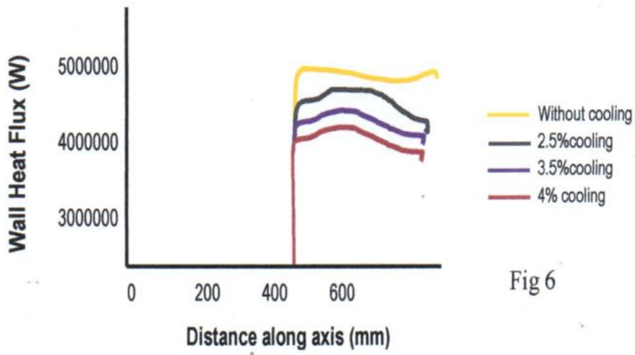


Fig 6

Comparison of temperature with & without cooling

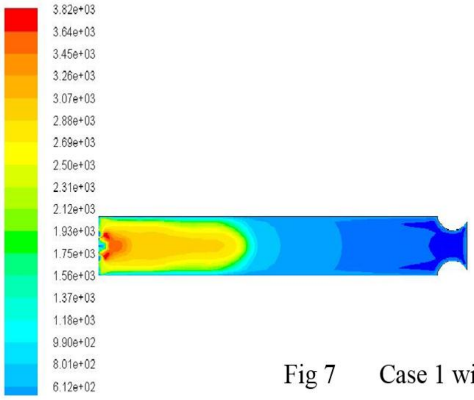


Fig 7 Case 1 with 2.5% cooling

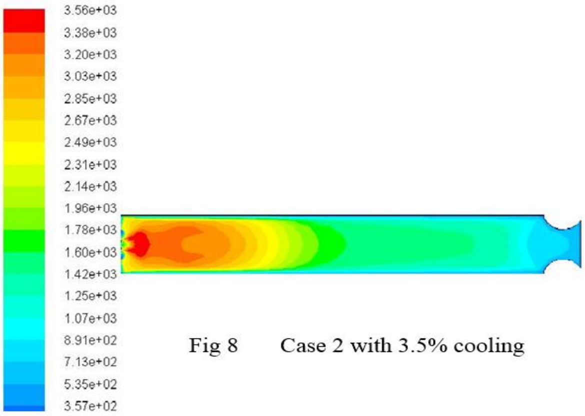


Fig 8 Case 2 with 3.5% cooling

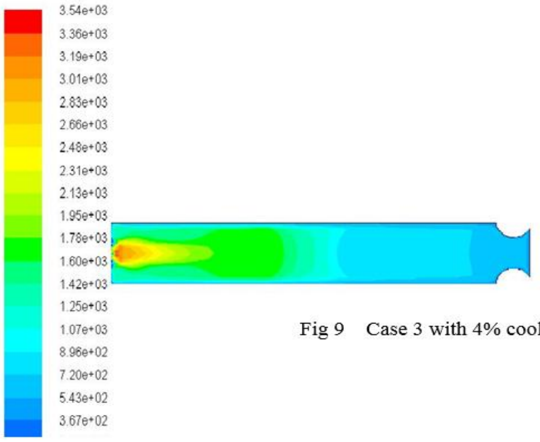


Fig 9 Case 3 with 4% cooling

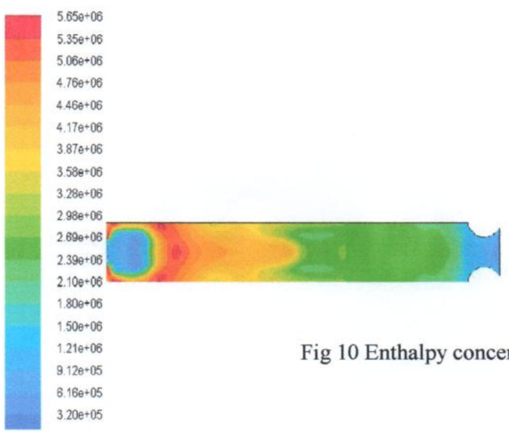


Fig 10 Enthalpy concentration of O_2

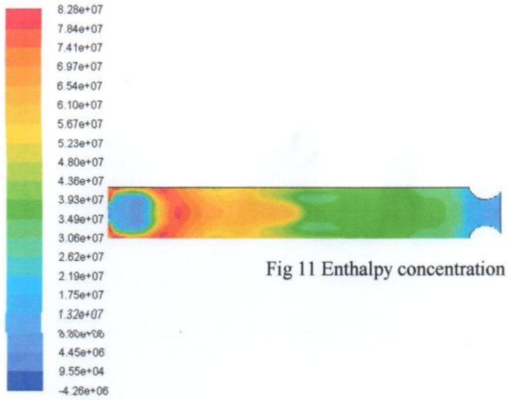


Fig 11 Enthalpy concentration of H_2

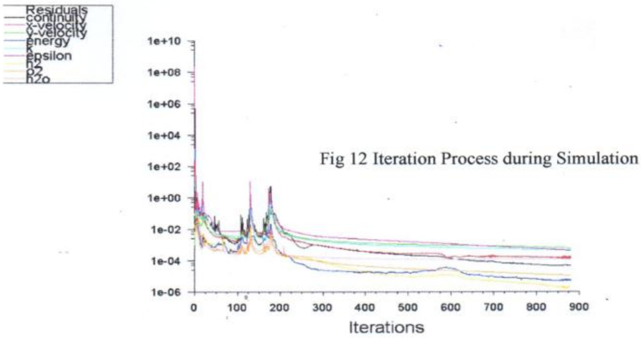
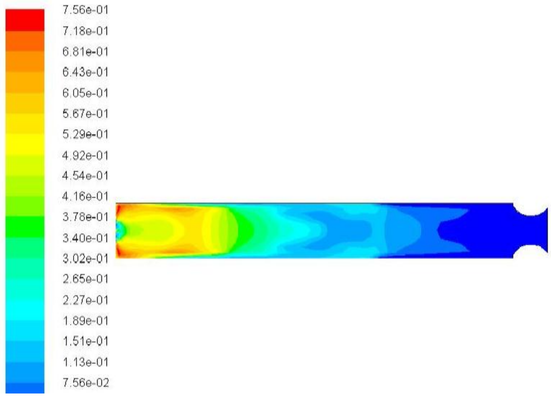


Fig 12 Iteration Process during Simulation

Fig 13 Contours of mass fraction of H_2O



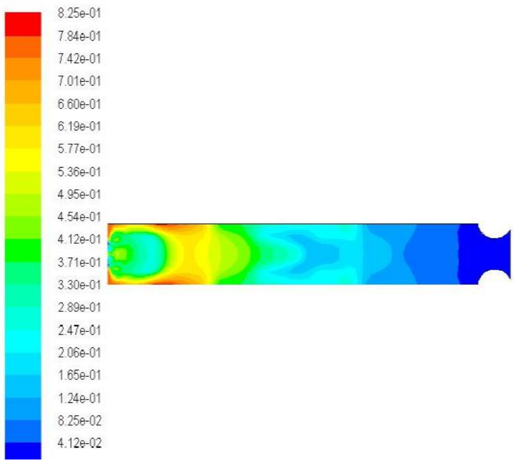


Fig 14 Contours of mole fraction of H_2O

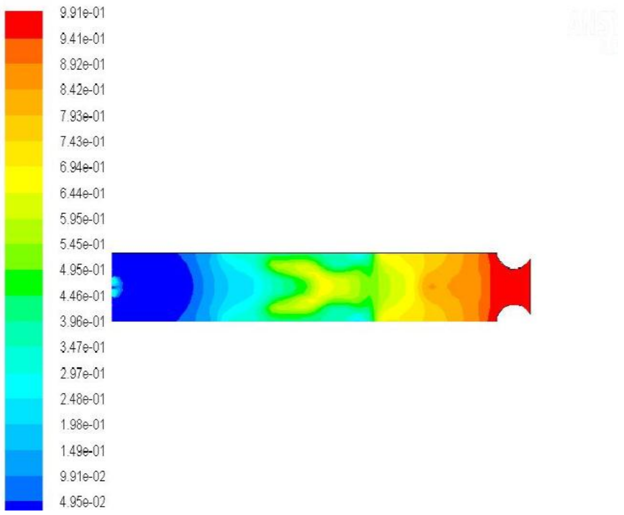


Fig 15 Contours of mole fraction of O_2

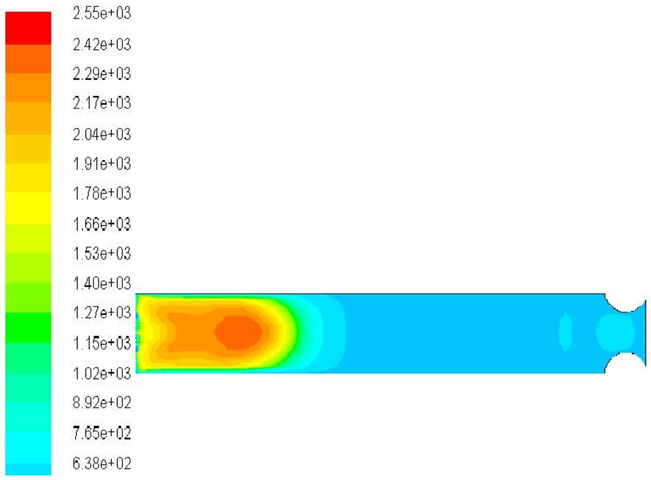


Fig. 16 combustion progress near the injector

CONCLUSION

The detailed evaluation of the combustion process, conjugate heat transfer and cooling is performed using an axisymmetric model of a typical cryogenic rocket engine. The studies clearly depicts the importance of inducing proper coolant so as to avoid fuel loss which itself is used for attaining cooling requirement. Combustion behavior of the chamber with and without cooling is studied thoroughly out of which a cooling percentage of 4% was found to be very effective for the chamber. It was successfully studied using the standard CFD package called fluent which is particularly used everywhere in cryogenic studies. On behalf of the analysis, it is found that the maximum thrust chamber pressure and temperature achieved was 20.32 bar and 3824K for case 1 with 2.5% cooling and 17.85bar for the uncooled study having an elevated temperature of above 5000K respectively. Heat fluxes achieved are over the range of 5.1 MW/m^2 which is quite common and up to the mark. Subsequently the following temperatures namely 3559 k for 3.5% cooling and 3539 for 4% cooling are worth noting in the analysis. It is clearly seen from case 3 that the heat flux got reduced by 21% when combination of film and regenerative cooling is adopted. Earlier the intensity of heat flux was maximum for the uncooled case.. During the analysis, a certain amount of fuel loss is noted in all the three cases including uncooled case is shown by light green shade due to the usage of single coaxial injector. This may lead to incomplete combustion and by providing suitable number of injectors, the above issue may be avoided. Due to the use of single injector, the fuel loss may get severly increased when coolant rate is increased. It is very clear from the flame behavior that heat flux is getting reduced when coolant rate is increased thereby results in fuel loss leading to improper combustion.

It has been mentioned in the figure 5 that the inner wall temperature is the maximum for the case without film cooling. When film cooling is imparted, the inner temperature gets reduced. The three dip in temperature when coolant is effected is also seen and the extreme reduction in temperature is noted in the entry of the third section of the film coolant, which is very much closer to the nozzle portion.

RESULTS OF THE PRESENT ANALYSIS

Chamber properties	Uncooled case	When 2.5% coolant used	When 3.5% coolant used	When 4% coolant used
Maximum Pressure in the chamber in bar	17.85	20.32	20.38	20.43
Maximum Inner Wall Temperature in Kelvin	5000.61	3824.52	3559.32	3539.58
Exit Temperaure in Kelvin	2247.54	1851.31	1845.65	1739.66
Exit velocity in m/s	5114.10	3971.43	3967.35	3884.39

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