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Numerical Analysis of Combustion and Film Cooling in LOx-Methane Rocket Engine

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Abstract. Nowadays methane is considered as an alternative fuel for rocket engine application. It is abundant in the outer solar system that can be harvested from mars, Titan, Jupiter, many other planets and moon. The properties of cryogenic methane like higher density, higher vaporization temperature, less challenge in storage requirement compared to cryogenic hydrogen and its higher specific impulse, superior cooling capability and higher coking limit compared to kerosene created a renewed interest on space scientist in choosing methane as a propellant. Combustion and film cooling involving supercritical and transcritical regimes in a high pressure rocket engine combustion chamber is a challenging problem for the rocket engine designers. At supercritical and transcritical conditions, the thermophysical and transport properties exhibit tremendous variations which makes modeling combustion and cooling a demanding task. The present study is based on the oxygen methane thrust chamber model B at DLR P6.1 test bench. The finite volume based software ANSYS Fluent 15.0 is used for the numerical studies. The combustion is simulated using single step eddy dissipation approach. The real gas effects are incorporated using Soave Redlich Kwong cubic equation of state. The numerical scheme is validated by comparing the results obtained with the experimental results reported in literature. Investigations have been carried out to study the effect of film cooling on chamber wall temperature and wall heat flux for different oxidizer inlet temperature. The result revel that the hot gas temperature near the wall is higher in the transcritical combustion compared to supercritical combustion.

1. Introduction

Liquid propulsion rocket (LPR) engine uses liquid propellants to react and form products at high pressure and temperature. These products are then accelerated through a nozzle to get a high velocity jet. Liquid propellants are commonly used in higher stage engines due to many reasons. High density of Liquid propellants helps to reduce the volume of propellant tank which leads to have high thrust to mass ratio and high specific impulse. They can be started and stopped whenever required hence they give better overall control compared to solid propellant rockets. Recent advancements in rocket technology have introduced reusable rockets which also employ liquid propellants. Many of the launchers nowadays use Liquid Hydrogen (LH2) and liquid Oxygen (LOX) as propellants because of their high specific impulse. The combustion products are of low molecular weight and hence higher velocity for the jet. Also, the combustion products are non-toxic and eco-friendly. However, LH2 as a fuel has two significant disadvantages: (i) it is a very low density cryogenic fluid and therefore need larger tanks maintained at temperatures as low as 20K for storage (ii) the pumping power requirement is very high. There also exists a chance for resonance as the pumping frequency is close to the natural frequency. A proposed alternative propellant combination is methane and LOX. Methane being four times denser than LH2, requires smaller tanks for storage. Also the pumping effort required is significantly lower, compared to LH2. Methane does not require extremely low temperatures as LH2 for storage. Although specific impulse is slightly less, ease of handling and storage enables it to be used in future launchers. Methane (CH4) is abundant in the outer solar system. With fuel waiting at the destination, a rocket leaving Earth wouldn't have to carry so much propellant, reducing the cost of a mission. Also in comparison with kerosene, methane is easy to produce, having higher coking limit and better cooling capabilities. Since use of methane as propellant is an emerging technology, the different process parameters that control these systems has to be carefully investigated. Rocket engines uses different cooling method to cool the thrust chamber walls from the hot gases formed due to combustion. The primary cooling method used in LPR is regenerative cooling. Regenerative cooling is the method by which either cryogenic fuel or oxidizer is passed through the cooling channel around the thrust chamber. Film cooling is an auxiliary cooling method used to protect the chamber walls from localised heating. Even though a good number of research finding are there in the field of film cooling, a critical design and performance evaluation is required for thrust chambers with multi-element injectors functioning with LOX/CH4 propellant combination. The performance characteristics are deeply influenced by the chamber pressure, propellant inlet conditions and also configurations of the injectors. The near injector performance of the film coolant is deeply influenced by the combustion temperature, pressure and shape of the flame. Arnold et al. [1-4] has carried out a number of experimental studies on DLR P8 combustor B. Film cooling effects are tested with LOX-H2, and LOX-CH4 propellant combination. In their preliminary film cooling studies for a pressure range of 5-11.5MPa, a significant variation of circumferential wall temperature distribution and film cooling effectiveness was observed with GH2 as film coolant. Their results showed that tangential injection of film coolant in the vicinity of coaxial injector head has produced an effective reduction in wall temperature. They also tested the film cooling effectiveness in a nozzle segment of a subscale rocket combustion chamber and found that modified film cooling model is required to implement an additional correction factor for local Mach number. Their further investigation on DLR P8 combustor using LOx/GH2 showed that an increase of the film blowing rate results in a direct improvement of the local film cooling effectiveness. Their studies also confirmed the dominant influence of the film blowing rate on film cooling effectiveness. Suslov et al. [5] investigated the experimental results of film cooling performance in an Oxygen-Methane subscale combustion chamber. They explained the different complexities occurring due to film cooling like steep temperature gradients, turbulent mixing, and complexities in flow properties. Barbara Betti [6] studied numerically the effects of film cooling in hydrogen and methane thrust chambers. The capability of Reynolds Average Navier Stokes equation is verified using pseudo-injector approach. This approach well predicts the heat flux characterization, particularly at high chamber pressure. The data acquired and models developed during the initial period of In-Space Propulsion (ISP-1) programme were used by Ordonneau [7] to validate the models used in CFD simulations. Daimon et al. [8] experimentally and numerically investigated the film cooling aspects in a CH4/Oxygen subscale chamber with multi-injector elements. The effect of slot dimension on film cooling and the net heat flux reduction were analysed. Their results showed that mixing causes a low heat flux distribution near face plate along with high combustion efficiency.

In the present study, the film cooling effects of a multi-element injector thrust chamber are numerically investigated. The influence of film coolant on chamber wall temperature and wall heat flux are explored. Computationally cheap Eddy Dissipation Model (ED) along with single step reaction is used for numerical simulation.

2. Theoretical Formulation and Numerical Approach

In the present numerical study, the commercially available numerical tool ANSYS FLUENT 15.0 [9-10] is used. The conservation equations of mass, momentum, energy, and species equations are solved in fluid phase sequentially. The turbulence is modelled using k-ε standard. The thermodynamic properties of the fluids are taken from the National Institute of Standards and Technology (NIST) database [11].

2.1. Computational Domain

The present study was conducted on DLR Lampoldshausen test facility P 6.1 combustor 'B' [8]. The schematic of the thrust chamber is shown in figure 1. The thrust chamber consists of 6 segments, each 50 mm long. The first five segments constitute the combustion chamber, while the last segment performs as the nozzle. The diameter of the combustion chamber is 50 mm. The nozzle has a contraction ratio of 3.2 with throat diameter 28 mm. The chamber and the nozzle are cooled by water circulated in the cylindrical channels provided on the periphery of the segments.

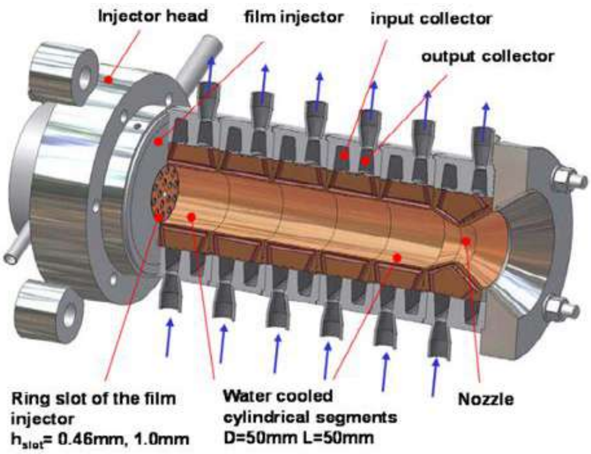


Figure 1. Schematic of the Thrust Chamber.

The injector plate has 15 coaxial injector elements. They are shown in figure 2. Oxygen is injected through the inner channel while fuel is injected through the outer ones. The injectors are placed in the face plate without recess. Among the 15 injectors, 5 injectors are placed on a pitch circle of diameter 19 mm and remaining are positioned on 38 mm pitch circle. The injectors are positioned uniformly on the pitch circle such that the elements form 5 identical triangles.

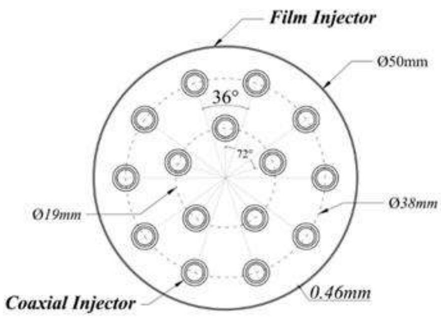


Figure 2. Injector face plate.

The film is injected through a continuous slot of 0.46 mm width. From the literature [5] it was found that the water flow rate was arranged such that a uniform wall temperature of 400 K is maintained on chamber lateral walls.

The computational domain was modelled using Design Modeler. The inlet for oxygen, fuel, and film are shown in figure 3. In order to reduce the size of the computational domain, only 72° of the thrust chamber was modelled. Periodic boundary condition was imposed on two sides of the domain. Periodic boundary condition was found most effective in modelling fluxes across the boundaries.

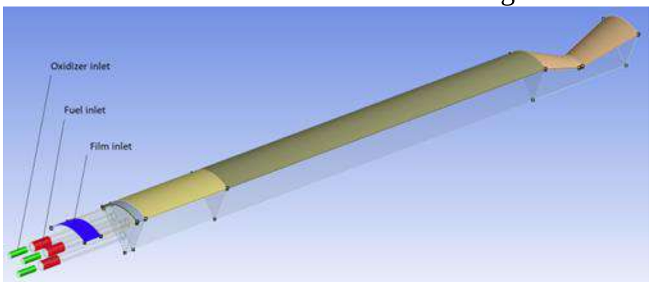


Figure 3. Computational Domain.

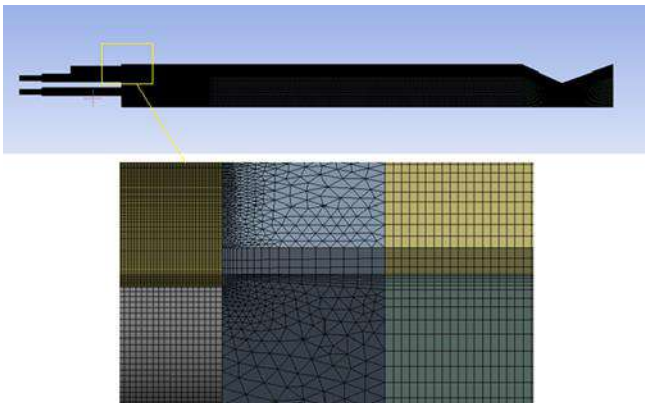


Figure 4. Generated Mesh.

The hybrid mesh was generated composing of hexahedral and tetrahedral elements. Fine elements were concentrated near the region of mixing. Tetrahedral elements were located up to a thickness of 5 mm from the faceplate. Inflation layers were created near the fluid-solid boundary for accuracy. A mesh having 1947112 elements is selected for the simulation after the grid independence study and the same is shown in figure 4.

3. Validation and Grid Independence Study

The results obtained from experimental studies by Daimon et al. [8] was chosen for validation of numerical scheme. The boundary conditions used for the validation study are shown in table 1.

Table 1. Boundary Conditions

Flow Parameters	Value
Fuel mass flow rate, kg/s	0.083
Fuel temperature, K	267
Oxidizer mass flow rate, kg/s	0.28
Oxidizer temperature, K	272
Film mass flow rate, kg/s	0.073
Film temperature, K	281
Velocity ratio (Vfilm/Vfuel)	1.08

A mixture ratio of 3.4 is used for the simulation. The real gas effects are predicted by using Soave-Redlich-Kwong equation of state. Pressure based solver was used for the simulation. Computationally cheap Eddy dissipation (ED) was chosen as the reaction mechanism. The standard k-ε model was used for modelling turbulence. Constant wall temperature of 400 K was imposed on the outer wall of the thrust chamber.
The grid independence was checked with element sizes of 501273,593277, 1244477, 1947112, 2550000. The results of chamber pressure and wall heat flux were chosen to predict the grid independence. Among the tested grids a mesh with 1947112 elements showed good agreement with experiment results with maximum error of 18% in the case of wall heat flux. The details are illustrated in Figure 5.

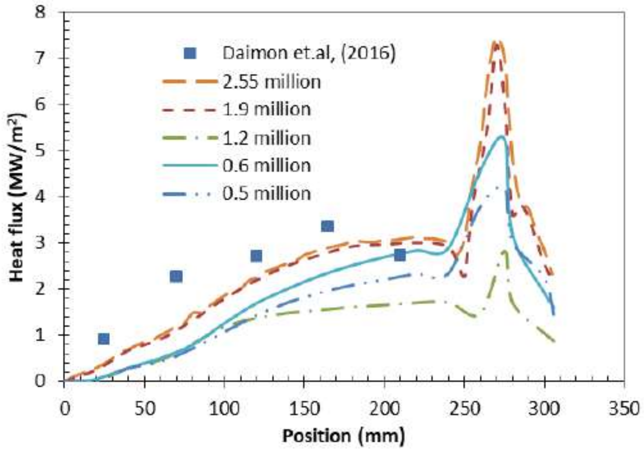


Figure 5. Details of Grid Independence Study

4. Results and Discussion
Most of the high-pressure rocket engines are provided with film cooling in addition to regenerative cooling to protect the wall surfaces from sudden heating. In the present study, efforts are taken to understand the influence of film coolant on wall heat flux and chamber temperature on PLR D6.1 subscale rocket thrust chamber at supercritical pressure.
The mass flow rates of oxygen and methane are 1.683 kg/s with 0.495 kg/s respectively. The film coolant methane is injected at a mass flow rate of 0.073 kg/s. Here the parametric studies were carried out by varying the oxygen inlet temperature. The chamber pressure lies in the near critical pressures of methane and oxygen where drastic variations in properties are observed. At 120K and 6 MPa density oxygen is 998 kg/m^3 and is approximately equal to that of water at 1 atmospheric pressure. Studies are conducted at three oxidizer inlet temperatures of 272K, 150K and 120K. In the second and third cases the injected oxygen is undergoing transcritical to supercritical transformation.

4.1. Effect of Inlet Temperature of Oxygen on Wall Temperature
Thrust chamber wall temperature increases gradually and shows a peak at the throat and then reduces due to thermodynamic expansion. In all the three cases the outer surface of the thrust chamber wall is imposed with a wall temperature of 400K assuming uniform regenerative cooling. In all the cases the wall temperatures are within safe limit. The maximum wall temperature obtained for 272 K is 509K and that for the 120K is 537K. This result indicates that regenerative cooling plays a vital role in maintaining the wall temperature under safer operating condition. When oxygen is injected at supercritical state, combustion occurs faster this is reflected in the graph. In the case of transcritical injection combustion occurs in the shear layer between the dense core of oxidizer and lighter fuel core. A slight delay in combustion is happening and due to this delayed combustion, the wall temperature increases slower in the case of transcritical combustion. The variation in wall temperature are illustrated in figure 6.

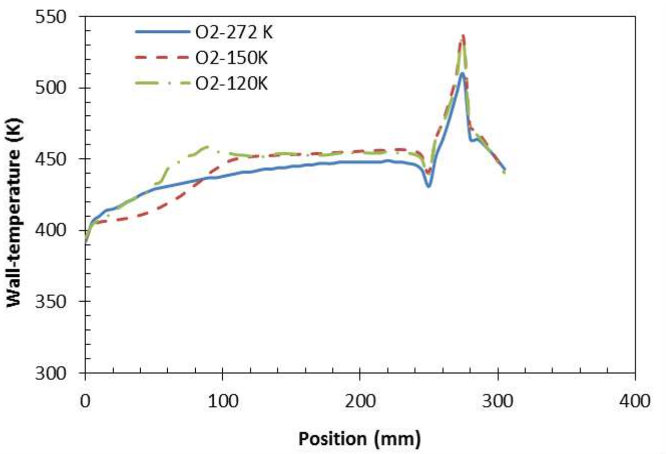


Figure 6. Variation in wall temperature for Isothermal wall Condition.

4.2. Effect of Inlet Temperature of Oxygen on Film Deterioration

Film cooling is an auxiliary cooling method used to protect the thrust chamber wall surfaces from localized heating caused due to sudden heat flux variations. Here the film coolant is injected at an inlet temperature of 281K. Figure 7 shows the consumption of film coolant methane along the axial direction measured 1mm below the thrust chamber. It can be clearly seen that in the case of cryogenic injection a stable film is maintained till the end of the thrust chamber. Primary recirculation occurring between the shear layers of propellants near the injector outlet, directs a part of the fuel towards the chamber wall. At the nozzle exit the mass fraction of methane is 0.1 for cryogenic injection cases; and for 272K injection case it is 0.03 only. Similar trend is obtained for mass fraction of methane at 2mm below chamber wall in figure 8.

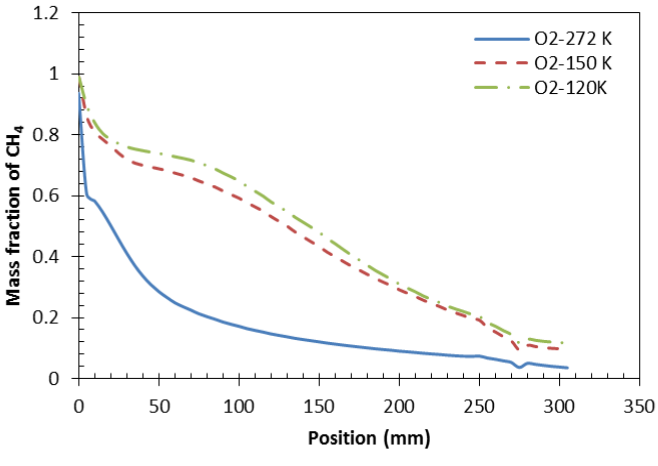


Figure 7. Consumption of methane in axial direction 1mm below chamber wall.

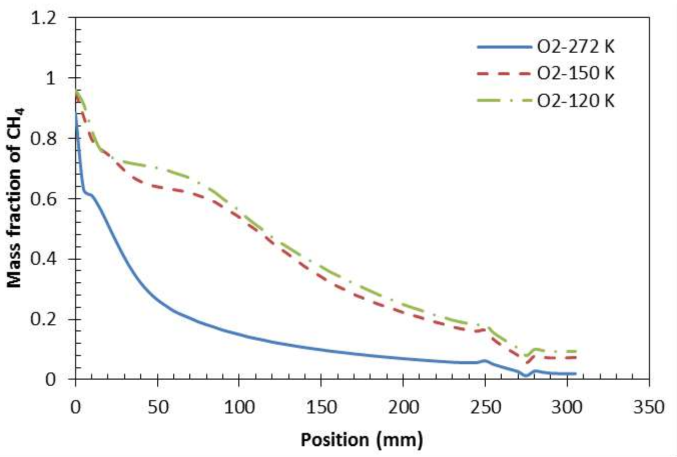


Figure 8. Consumption of methane in axial direction 2mm below chamber wall.

Studies indicate that due to the fuel injection pattern there is a layer of methane near the inner surface of the thrust chamber even without film cooling (figure 9). When oxygen is injected at 120 K and without film coolant, a significant amount of methane present near the wall is contributed by the upper injector elements. The mass fraction of methane near the wall increases with film cooling.

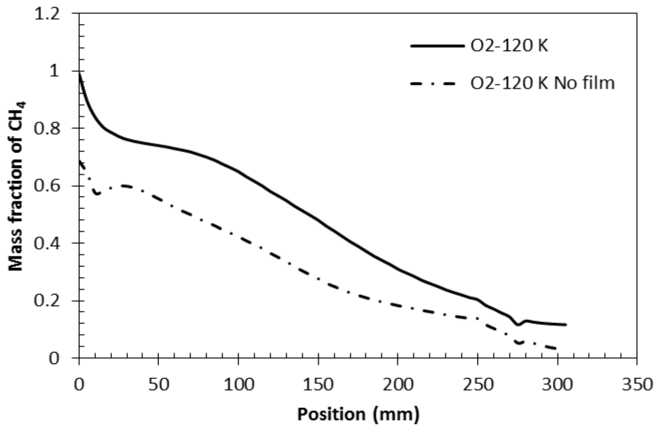
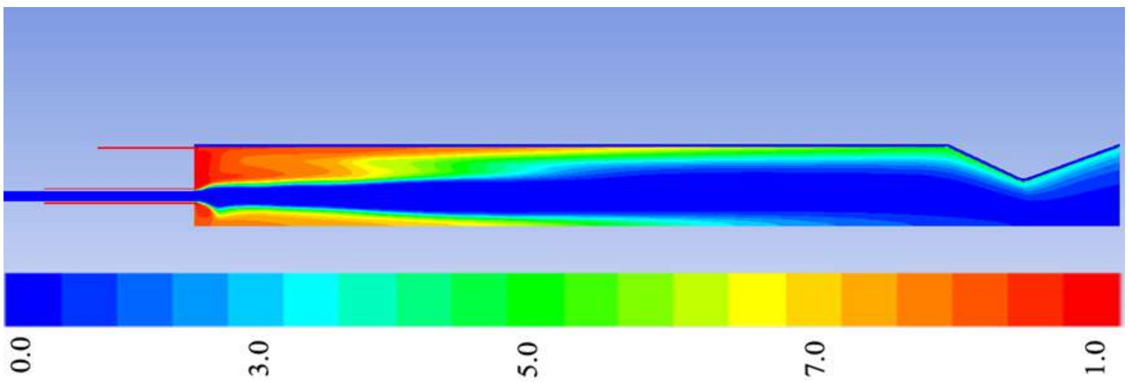
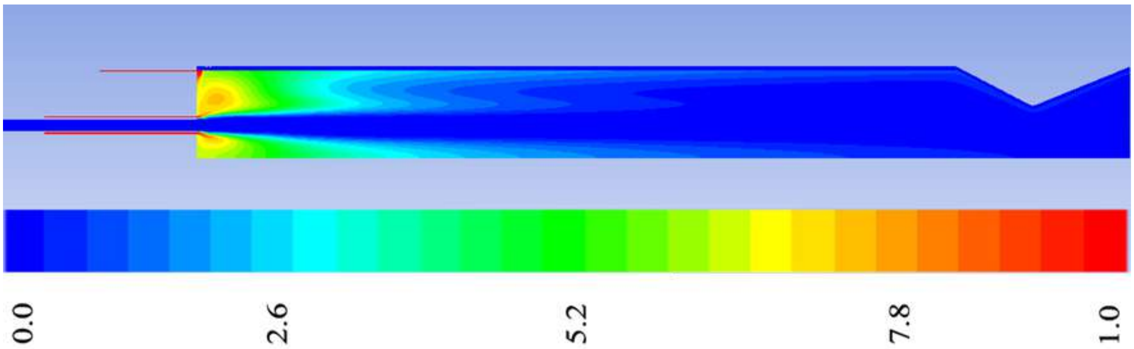


Figure 9. Comparison of methane mass fraction near wall with and without film cooling, oxygen temperature 120 K.



a)



b)

Figure 10. Contour of mass fraction of CH4, (a) oxygen temperature 120 K (b) oxygen temperature 272 K.

The figure 10 shows the contour of mass fraction of CH4 on the plane passing through the centre of an inner injector element. In the case of cryogenic injection a significant amount of methane mass fraction is observed near the wall and this may affect the combustion efficiency.

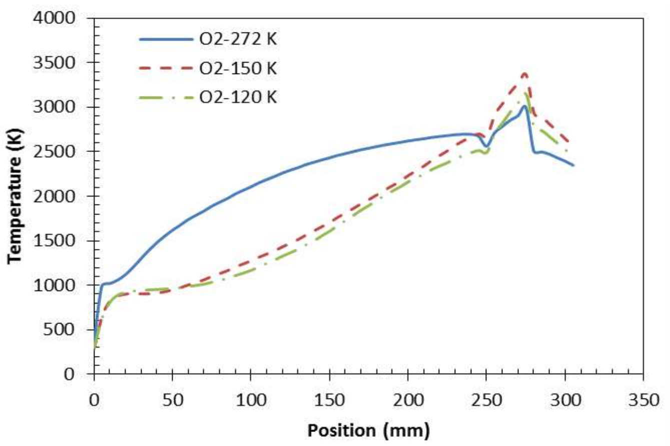


Figure 11. Variation of hot gas temperature in axial direction 1mm from the wall.

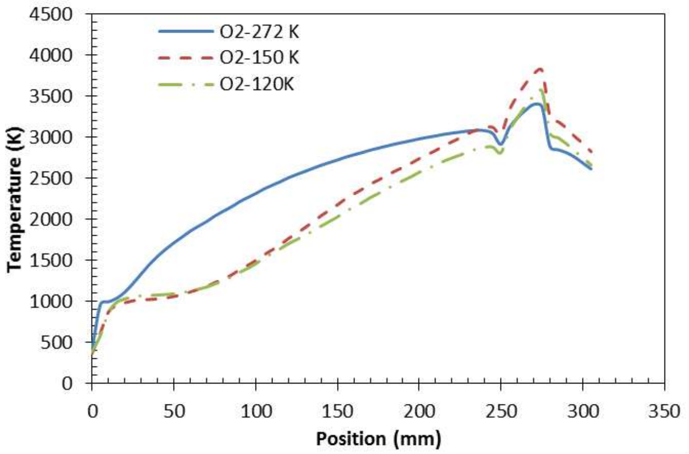


Figure 12. Variation of hot gas temperature in axial direction 2mm from the wall.

Hot gas temperature at distances of 1mm and 2mm from the wall is depicted in figure 11 and figure 12 respectively. In all the cases the hot gas temperature increases up to the throat then it decreases further downstream due to the thermodynamic expansion of gases. Hot gas temperature near the wall in the combustion chamber is high at supercritical combustion. In the case of transcritical injection due to delayed combustion, the hot gas temperature reaches the maximum value at a slower rate than supercritical injection.

4.3. Effect of Inlet Temperature of Oxygen on Wall Heat Flux

Once the oxygen is injected at 272K the wall heat flux increase gradually and reaches maximum at throat and decreases in the divergent section due to expansion. Figure 13 shows the average wall heat flux distribution along the axial direction. When oxidizer is injected at 272 K, the chamber pressure developed is 5.33MPa and the maximum value of heat flux at throat is 38 MW/m^2. In the cases of transcritical injection, upto an axial distance of 70mm the wall heat flux is lower compared to that at 272K. This reduction in wall heat flux is due to the influence of strong primary recirculation at cryogenic injection. For the inlet temperature of 120 K the density of oxygen is approximately 1000 kg/m^3 and due to this high density the combustion occurs in the shear layer between dense core of oxygen and methane. Once the propellant oxygen is injected at cryogenic state the combustion chamber pressure is raised to 6MPa. Similar wall heat flux conditions are observed for the injection temperature of 150K.

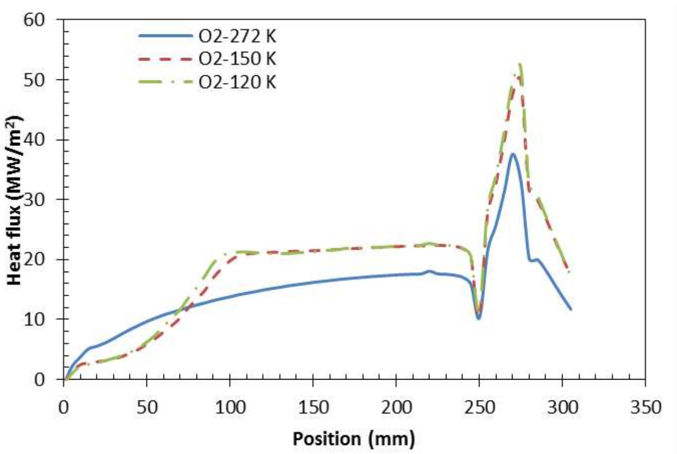


Figure 13. Variation of wall heat flux with oxidizer temperature.

5. Conclusions

Numerical studies on film cooled subscale thrust chamber was carried out to study the effect of inlet temperature of oxygen on wall temperature and wall heat flux. The numerical scheme is validated comparing the results with experimental results reported in literature. The combustion model used is single step eddy dissipation model and the turbulent model used is standard k-ε. The following conclusions are derived from the study.

For combustion where oxygen is undergoing transcritical to supercritical transformation, the methane concentration near the wall increases. A significant amount of the methane mass fraction near the chamber wall is contributed by the upper injector elements. The hot gas temperature near the wall is higher in the transcritical combustion compared to supercritical combustion. In the case of cryogenic injection a stable coolant film is maintained till the end of the thrust chamber. In the cases of transcritical injection, up to an axial distance of 70mm the wall heat flux is lower compared to that of supercritical injection.

References

[1] Arnold R, Suslov D and Haidn O J 2008 *proc. 44th AIAA/ASME/SAE/ASEE Joint Propulsion Conference & Exhibit* (Hartford) p 5242.
[2] Arnold R, Suslov D and Haidn O J 2009 *Journal of propulsion and power* **25** p 443.
[3] Arnold R, Suslov D I , and Haidn O J 2009 *Transactions of the Japan society for aeronautical and space sciences* (Japan) **7** p 81.
[4] Arnold R, Suslov D I, and Haidn O J 2010 *journal of propulsion and power* **26** p 428
[5] Suslov D, Betti B, Aichner T, Soller S, Nasuti F, Haidn O J 2012 *In Space Propulsion Conference*.
[6] Betti B, Onofri M, and Nasuti F 2012 *proc. 48th AIAA/ASME/SAE/ASEE Joint Propulsion Conference & Exhibit* (Atlanta: Georgia) p 3907.
[7] Ordonneau G 2013 *proc. 5th European conference for aeronautics and space sciences* p 11.
[8] Daimon Y, Negishi H, Koshi M, and Suslov D 2016 *Progress in Propulsion Physics* **8** p 129.
[9] Ansys Fluent 15.0 User's Guide 2013 (Ansys Inc: Canonsburg USA).
[10] Ansys Meshing 15.0 User's Guide 2013 (Ansys, Inc., Canonsburg USA).
[11] NIST Standard Reference Database 69.