Numerical Analysis of Regenerative Cooling in Liquid Propellant Rocket Engines for Ground Testing

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Abstract: ***In liquid propellant rocket engine must be required cooling techniques for high combustion temperature during long operation. In this study propellants of liquid rocket engine for ground testing are used liquid Kerosene and gaseous oxygen and they are exploded in combustion chamber of rocket engine. The combustion gases produce high temperature and pressure that effected to destroy the inner side wall or gas-side wall and the rocket engine is burned and explosion. So, cooling system is essential for liquid propellant rocket engine. For my numerical analysis study coolant is used as water for reduced temperature of liquid propellant rocket engine walls and cooling method is regenerative cooling method. To determine the cooling efficiency or cooling temperature on gas-side wall, coolant-side wall and coolant are performed by numerical calculation.***

Keywords: Liquid propellant rocket engines, Kerosene, Gaseous Oxygen, Regenerative cooling, Temperature

1. Introduction

All liquid propellant rocket engines are high energy released by combusted gases. The high combustion temperature, high heat transfer rates are requires special cooling techniques for the rocket engine. Cooling techniques are regenerative cooling, radiation cooling, film or transpiration cooling, ablation, arid inert or endothermic heat sinks. The selection of cooling technique should be considered by mission requirements, environmental requirements and operational requirements should be considered.

In liquid propellant rocket engines regenerative cooling is one of the most widely applied cooling techniques used. In regenerative cooling of rocket engine there are three domains gas domain (combusted gases), liquid domain (coolant) and the solid domain (thrust chamber wall). The heat transfer analysis are simply based on convection and radiation heat transfer for gas domain, conduction heat transfer for solid domain and convection heat transfer for liquid domain. Assuming that the outer surface wall is adiabatic an heat transfer from the outer surface of thrust chamber to the environment can be neglected. To simplify the gas side and coolant side heat transfer analysis, many correlations are developed to calculate the heat transfer coefficients.

In this study, the combustion chamber and nozzle of rocket engine geometry is obtained preliminary according to the design parameters. The effects of hot combustion gas on cooling efficiency are investigated in terms of the maximum temperature of thrust chamber wall and coolant. Thermal properties of combustion of Kerosene and gaseous Oxygen are calculated with thermochemical equilibrium code. Heat transfer calculation from gas side domain (combustion gases) to the solid domain (thrust chamber) and the solid domain to coolant are calculated with M.B Dobrovolskee method. Moreover the resulting data are shown in tables and graphs.

2. Regenerative Cooling

In 1938 Regenerative cooling is first demonstrated by James H. Wyld in United States and today it is one of the most widely applied cooling technique used in liquid propellant rocket engines. In this process, the coolant generally enters at nozzle exit of the thrust chamber, passes in passages by the throat region and exits near the injector face. Cross-sectional view of a regenerative cooling of liquid propellant rocket engine is shown in Figure 1.

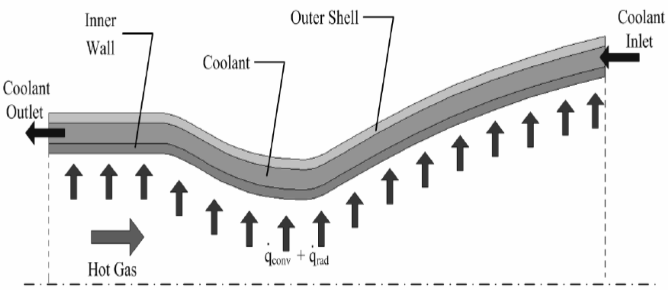

Figure 1: Cross-Section of liquid propellant rocket engine regenerative cooling system

The highest heat flux is occurred at the nozzle throat region and the most difficult to cool in it. The cooling passage in this reason is often designed and the coolant velocity is highest at regions by restricting the coolant passage cross-section. There are various types of regenerative cooling and in some cases to increase the cooling efficiency; coolant can enter the coolant passages either from the nozzle exit and nozzle throat or directly from the nozzle throat.

A typical heat flux distribution along the thrust chamber wall is shown in Figure 2. The lowest heat flux is usually near the nozzle exit and the maximum heat flux is always at the nozzle throat and.

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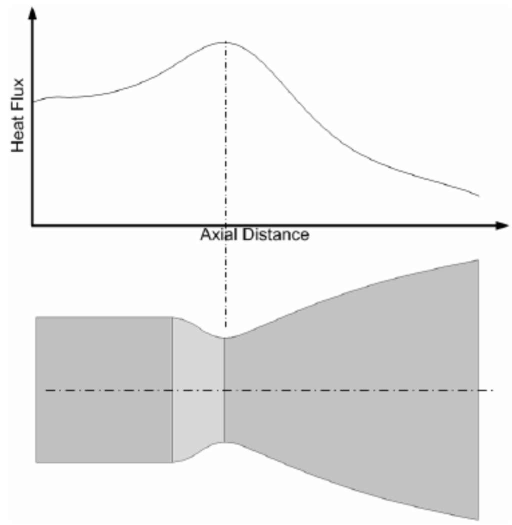

Figure 2: Typical heat flux of liquid propellant rocket engine

3. Selection of Materials for Thrust Chambers

Selection of inner wall is used copper. Cooper is does not oxidize in fuel rich non-corrosive gas mixtures and it is an excellent conductor. Copper alloys with small additions of zirconium, silver or silicon can be used for thrust chambers to increase the strength of material. The zirconium copper is thermal conductivity with good strength at high temperature. The silver zirconium copper alloy is thermal conductivity with moderate strength retention at high temperature. These materials have better strength retention, but they have lower conductivity than oxygen free high conductivity copper. The outer wall material is selected as stainless steel.

4. Specification Data of Liquid Rocket Engine

Pressure in combustion chamber; p_c = 2.068427 MPa
Temperature in combustion chamber; T_c = 3513 K
Specific hear ratio; γ =1.219
Thrust force; F = 300 N
Component of propellant; Gaseous oxygen (O_2) and Kerosene ($C_{7.2107}H_{13.2936}$)
Coolant (Water) flow rate; $\dot{m}_{co} = 0.0357 \quad \text{kg/sec}$
Coolant inlet temperature; $T_{co.in}$ = 288 K

The liquid propellant rocket engine for ground testing with cooling cover is designed at shown in Figure 3.

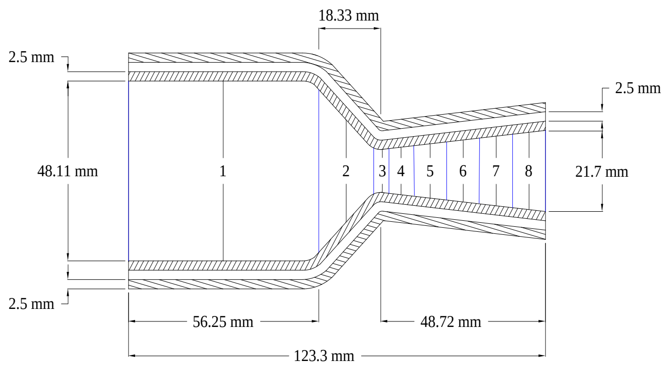

Figure 3: Profile of liquid propellant rocket engine

For numerical calculation of gas-side wall temperature is needed assumption firstly. So, initial gas-side wall temperature is assumed along the chamber and nozzle length which is shown in Figure 4.

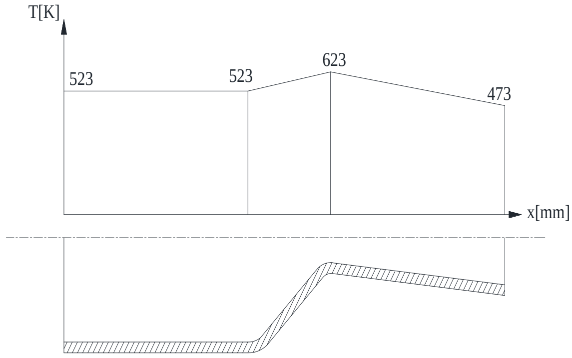

Figure 4: Initial assuming gas-side wall temperature along liquid propellant rocket engine

5. Numerical Analysis of Regenerative Cooling

Convection heat transfer coefficient H_g of combustion gas

$$H_g = 0.0206 C_{p.g} \eta_g^{0.18} \frac{\dot{m}_{co}^{0.82}}{d_m^{1.82}} \left(\frac{T_{aw}}{T_{wg}} \right)^{0.35}$$

Convection heat flux of combustion gas

$$q_{conv} = H_g (T_{aw} - T_{wg})$$

Radiation heat flux of combustion gas

$$q_{rad.cc} = \varepsilon_{wall} . \varepsilon_g . C_o \left(\frac{T_{aw}}{100} \right)^4$$

Starting distance 50 mm of combustion chamber

$q_{rad} = q_{rad.\,cc}$

At section № 2

$q_{rad} = 0.75 \cdot q_{rad.\,cc}$

At throat section (Section № 3)

$q_{rad} = 0.5 \cdot q_{rad.\,cc}$

At divergent section

$$q_{rad} = \frac{q_{rad.cc}}{2 \cdot \bar{d}^2}$$

Total Heat flux in the combustion chamber

$q_{\Sigma} = q_{conv} + q_{rad}$

Heat flux along Liquid Propellant Rocket Engine is shown in Fig. 5.

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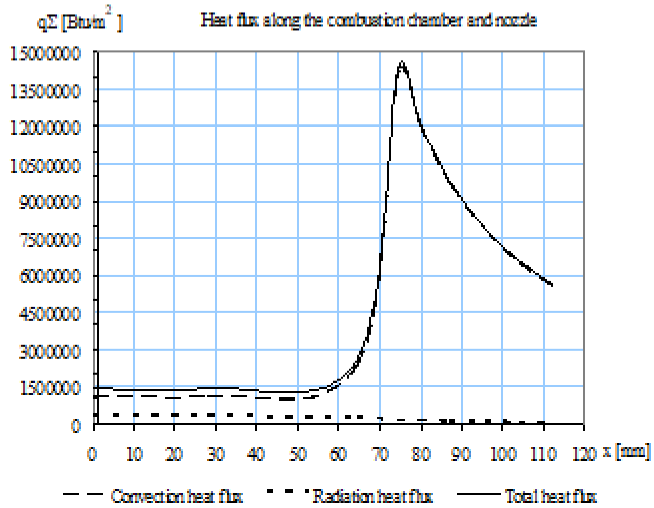

Figure 5: Heat flux along liquid propellant rocket engine

Coolant temperature at section № 8

$$T_{co.out.i} = \frac{q_i}{\dot{m}_{co} C_{p.co}} + T_{co.in.i}$$

Where; $C_{p.co} = -2.82 \times 10^{-8} T_{co.in}^2 - 2.95 \times 10^{-4} T_{co.in} + 0.261$

Coolant temperature at section № 8

$$T_{co.i} = \frac{T_{co.in.i} + T_{co.out.i}}{2}$$

Thermo-physics parameter at section № 1

$$K_i = \frac{C_{p.co.i}^{0.4} . \lambda_{co.i}^{0.6}}{\eta_{co.i}^{0.4}}$$

Where; $\lambda_{co} = 9.64 \times 10^{-8} T_{co}^2 - 2.95 \times 10^{-4} T_{co} + 0.261$

$$\eta_{co} = -1.46 \times 10^{-11} T_{co}^3 + 3.22 \times 10^{-8} T_{co}^2 - 2.39 \times 10^{-5} T_{co} + 0.006$$

Heat transfer coefficient of coolant H_{co} at section № 1

$$H_{co.i} = \frac{0.023 \cdot (\rho_{co} v_{co})^{0.8} \cdot K_i}{D_{h.i}^{0.2}}$$

Heat transfer coefficient of coolant H_{co} is calculated for every section and then coolant-side wall temperature can be calculated at every section.

$$T_{wc} = \frac{q_{\Sigma.i}}{H_{co.i}} + T_{co.i}$$

Mean temperature of wall; $T_{w.m} = \dfrac{T_{wc} + T_{wg}}{2}$

Result of gas-side wall temperature $T_{wg.i}$ at section № 1

$$T_{wg.i} = \frac{q_{\Sigma.i} \cdot t}{\lambda_{wall}} + T_{wc.i}$$

Where; $\lambda_{wall} = 6.66 \times 10^{-4} T_{w.m}^2 - 5.91 \times 10^{-1} T_{w.m} + 520$

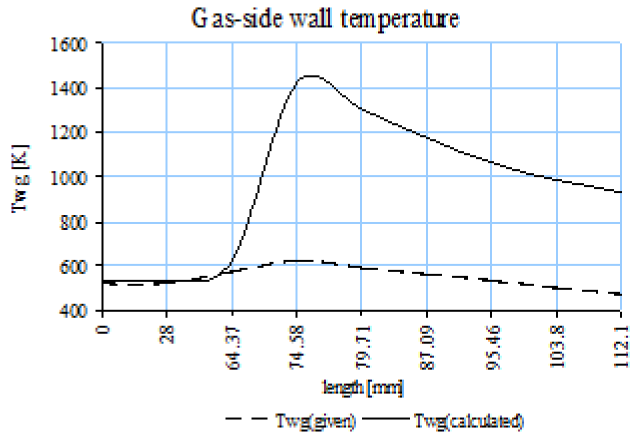

Figure 6: First calculation result of gas-side wall temperature

Error presence at section № 1

$$E_i = \left| \frac{T_{wg(calculated)} - T_{wg(given)}}{T_{wg(calculated)}} \right| \times 100$$

Table 1: First calculation of gas-side wall temperature

№	1	2	3	4	5	6	7	8
T_{wg} (calculated)	532	621	1418	1303	1171	1062	983	925
T_{wg}(given)	523	573	623	593	563	533	503	473
Error (%)	1.81	8.46	127.6	119.8	108.1	99.3	95.5	95.6

Figure 6 and table 1 shown the difference between calculated and given gas-side wall temperature is not equal, so the calculation is again by try and error method. If the calculated gas-side wall temperature T_{wg} is not 5% more than given gas-side wall temperature, calculation should be stopped. Final calculation results are shown by the following.

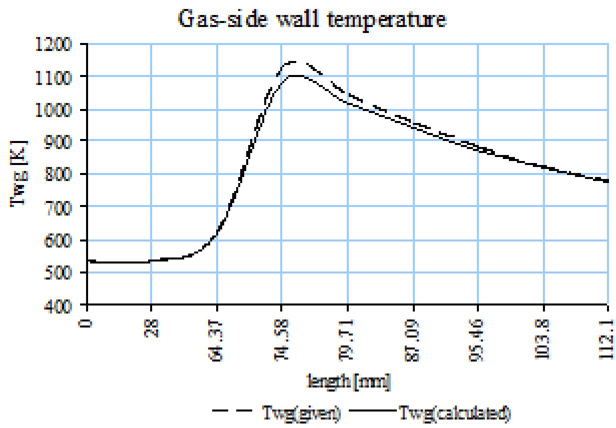

Figure 7: Final calculation result of gas-side wall temperature

Table 2: Final calculation of gas-side wall temperature

№	1	2	3	4	5	6	7	8
T_{wg} (calculated)	534.4	617.0	1079.3	1016.2	940.61	872.04	817.47	773.13
T_{wg}(given)	534.0	616.1	1120.9	1042.4	955.33	880.79	823.37	777.51
Error (%)	0.07	0.15	3.71	2.52	1.54	0.99	0.72	0.56

In final numerical calculation resulting given and calculated temperature are difference below 5% error at shown in Fig. 7 and table. 2, and the calculated should be stopped.

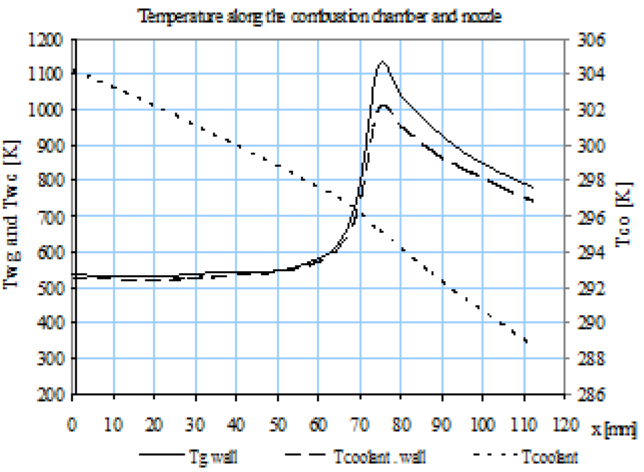

Figure 8: Final numerical calculation resulting temperature

6. Result and Discussion

The final numerical calculation resulting temperature in Fig. 8 is satisfied theoretically for ground testing of liquid propellant rocket engine cooling system because the temperature of wall and coolant temperature are not more than their melting temperature and boiling temperature.

In this study, the maximum heat flux is occurred at the throat of engine and the wall temperature is the highest at this section. So, the throat section of liquid propellant rocket engine is essentially needed for cooling. From the numerical calculation results data the wall temperature is not only less than the melting temperature of selected material, but coolant is also less than the boiling temperature. So, this calculation result data are satisfied for experimental ground testing. On cooling efficiency geometry effect and rectangular cooling channels should be investigated in terms of the maximum temperature of thrust chamber wall and coolant.

This paper gives the numerical analysis of regenerative cooling for a ground test preliminary designed thrust chamber and nozzle. As a future work, the parameters affecting the cooling efficiency can be optimized for given conditions. User defined function used for heat flux on gas side wall can be improved in consideration of turbulence effect in combustion region of thrust chamber.

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