Comparison of Different Aspect Ratio Cooling Channel Designs for a Liquid Propellant Rocket Engine

M. E. Boysan[1], A. Ulas[2], K. A. Toker[1], B. Seckin[1]
[1]Roketsan Missile Industries Inc.,
P.K. 30 Elmadag, 06780, Ankara, TURKEY
[2]Middle East Technical University
Faculty of Engineering, Department of Mechanical Engineering,
06531, Ankara, TURKEY

Abstract*-High combustion temperatures and long operation durations require the use of cooling techniques in liquid propellant rocket engines. For high-pressure and high-thrust rocket engines with long operation times, regenerative cooling is the most preferred cooling method. In regenerative cooling, a coolant flows through passages formed either by constructing the chamber liner from tubes or by milling channels in a solid liner. Traditionally, approximately square cross sectional channels have been used. However, recent studies have shown that by increasing the coolant channel height-to-width aspect ratio, the rocket combustion chamber hot-gas-side wall temperature can be reduced significantly. In this study, the regenerative cooling of a liquid propellant rocket engine has been numerically simulated. The engine has been modeled to operate on a LOX/GH$_2$ mixture at a chamber pressure of 68 atm and LH$_2$ (liquid-hydrogen) is considered as the coolant. A numerical investigation was performed to determine the effect of different aspect ratio cooling channels and different coolant mass flow rates on hot-gas-side wall temperature and coolant pressure drop. The variables considered in the cooling channel design were the number of cooling channels and the cooling channel cross-sectional geometry along the length of the combustion chamber.

SYMBOLS

AR	Aspect ratio
Cp	Coolant heat capacity at constant pressure [J/kg-K]
d	Hydraulic diameter [mm]
f	Friction loss coefficient
g	Gravitational acceleration [m/s^2]
G	Coolant mass flow rate per unit area [kg/m^2-s]
h_C	Coolant-side heat transfer coefficient [W/m^2-K]
h	Cooling channel height [mm]
L	Cooling channel length [m]
$\dot{m}$	Total mass flow rate [kg/s]
$\dot{m}_{LH}$	Liquid hydrogen mass flow rate [kg/s]
n	Normal outward direction
N	Number of cooling channels
P	Pressure [atm]
P_{amb}	Ambient pressure [atm]
P_c	Chamber pressure [atm]
Pr	Prandtl number
T	Temperature [K]
T_f	Flame Temperature [K]
T_{LH}	Liquid hydrogen initial temperature [K]
T_{CO}	Coolant bulk temperature [K]
T_{WC}	Coolant-side wall temperature [K]
u	Velocity along x-direction [m/s]
v	Velocity along y-direction [m/s]
V_{CO}	Average coolant flow velocity [m/s]
w	Velocity along z-direction [m/s]
x	Cartesian coordinate x-axis
y	Cartesian coordinate y-axis
z	Cartesian coordinate z-axis
ω	Cooling channel width [mm]
ΔP	Coolant pressure drop [atm]
ρ	Average density of the coolant [kg/m^3]
μ	Coolant viscosity [kg/m-s]

I. INTRODUCTION

Wall cooling is an important phenomenon in liquid propellant rocket engines. Because of high combustion temperatures, long combustion durations and high heat transfer rates from the hot gas side to the chamber wall, thrust chamber cooling is a major design consideration. Regenerative cooling is one of the most common cooling methods used in rocket systems. In regenerative cooling, one of the propellants (mainly fuel) flows through cooling passages which are located within the combustion chamber wall itself. The aim of the cooling is to keep the temperature low on the hot-gas side chamber wall, to avoid failure because of melting or thermal stresses. For an effective cooling, it is desired to have a high heat transfer coefficient for the coolant, good rib efficiency, smaller

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thickness and high thermal conductivity for the chamber wall. The heat transfer coefficient of the coolant is given as [1];

$$h_c = \frac{85.37 Cp\mu^{0.2}}{\Pr^{2/3}}\left(\frac{G^{0.8}}{d^{0.2}}\right)\left(\frac{T_{co}}{T_{wc}}\right) \tag{1}$$

Increasing coolant flow rate per unit area or decreasing hydraulic diameter of the cooling channel increases heat transfer coefficient of the coolant. Both of these parameters can be changed with different numbers and aspect ratios (AR) of cooling channels (Fig. 1 and 2).

Ribs are the chamber wall volumes between the cooling channels. They increase the strength of the chamber wall. To increase the total heat transfer from the hot-gas side chamber wall, total heat transfer area on the cross-section should be increased. This is possible by placing closely spaced and thin ribs.

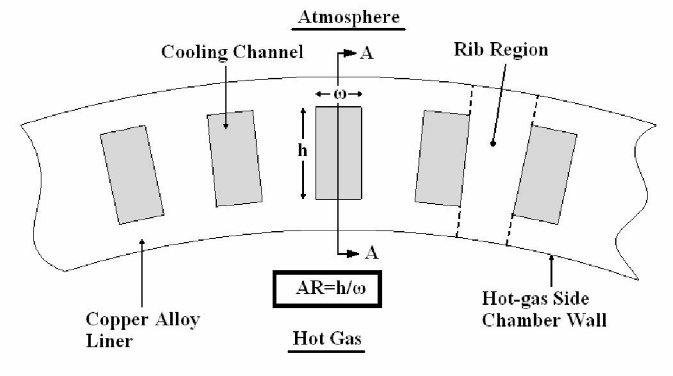

Fig. 1. Cross-sectional View of Combustion Chamber

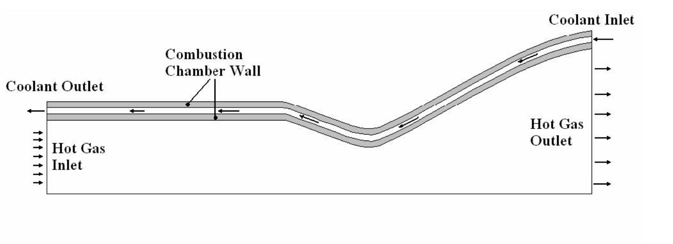

Fig. 2. View of Combustion Chamber from Section A-A

It is essential to design a cooling channel with minimum pressure drop during cooling. The pressure drop in the channel is estimated as [1];

$$\Delta P = f\frac{L}{d}\frac{\rho V_{CO}^2}{2g} \tag{2}$$

Increasing aspect ratio will decrease the average diameter of the channel and increase the pressure drop for the same coolant flow velocities. Therefore to use the high aspect ratio just for the critical regions instead of the whole region, will decrease the pressure drop relatively.

In this study maximum wall temperature and pressure drop parameters are investigated by changing the channel geometries for an engine to operate on a LOX/GH_2 mixture at a chamber pressure of 68 atm. Mass flow rate of the coolant is fixed as the total mass flow rate of the liquid hydrogen.

II. Validation

The geometry of the trust chamber is designed by using isentropic gas dynamic equations. The numerical simulation was carried out by FLUENT®. The comparison of the solution was made with the baseline solution of Wadel [2] for an 89 kN LOX/GH_2 liquid propellant rocket engine.

In Wadel's study Rocket Thermal Evaluation code (RTE) and Two-Dimensional Kinetics nozzle performance code (TDK) are used. RTE is a three-dimensional thermal analysis code and uses a three-dimensional finite differencing method. A Gauss-Siedel iterative method is used at each axial location to determine the wall temperature distributions. GAS Properties (GASP) and complex Chemical Equilibrium and Transport properties (CAT) are the two subroutines used in this code to determine the coolant and hot-gas side thermal properties. TDK code evaluates the heat fluxes on hot-gas side chamber walls with the wall temperature distribution from RTE. Chamber pressure, coolant temperature, mass flow rates and coolant inlet pressure are given as input parameters, pressure drop, hot-gas side chamber wall temperature and coolant exit pressure are the results of the solution.

Four different channel geometries were investigated according to their maximum hot-gas side chamber wall temperatures and pressure drops in cooling channels. Since the chemical combustion reactions in the chamber near injector were not modeled in this study, the wall temperatures on the combustion wall for this region are not considered.

III. Combustion Chamber Design

Chemical Equilibrium Code [3, 4] was used to obtain the thermo-chemical properties of the combustion gases, flame temperature of the combustion and the specific impulse for the optimum expansion ratio for the sea-level conditions. The operating parameters of the liquid propellant rocket engine used while designing the regenerative cooling model are given in TABLE I.

TABLE I
Operating Parameters of the Engine

Propellant	LOX/GH_2
Coolant	LH_2
Mixture ratio (O/F)	5.0
Chamber pressure	68 atm (1000 psia)
Total mass flow rate of the propellant	13.0 kg/s

The rocket engine model is given in Fig. 3. Bell-shaped nozzle with parabolic approximation is used. The inner liner of the engine is a copper alloy.

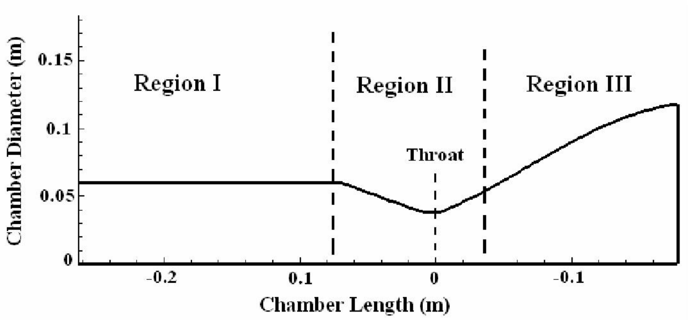

Fig. 3. Model of the Combustion Chamber

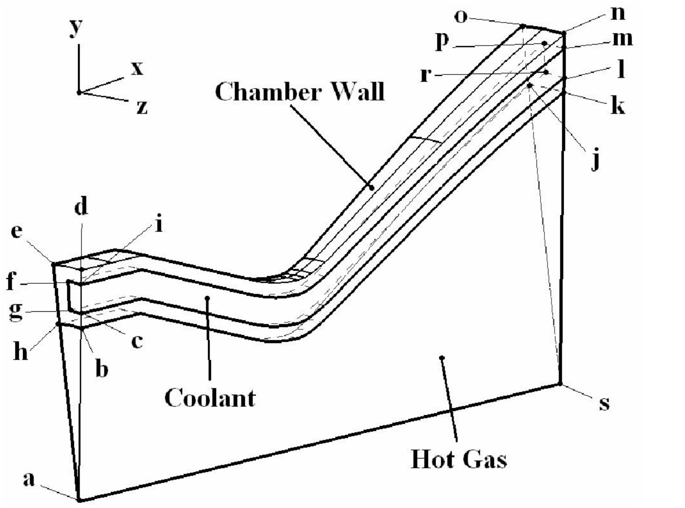

Fig. 4. Solution Domain of the Geometry

IV. Cooling Channel Design

Three regions are considered for each cooling channel as shown in Fig. 3. For most rocket systems, throat region is the most critical part because of high heat fluxes. In this study, to compare cooling efficiencies of different cooling channel designs, Region I and Region III inner geometries are held constant, which provided the same flow field inside the engine for all the simulations. Comparison matrix for the cooling channel geometries are given in TABLE II.

TABLE II
COMPARISON MATRIX FOR COOLING CHANNELS

Alternatives	# of Cooling Channels	Mass Flow Rate per Channel (kg/s)	Geometry of Cooling Channel (h x w) in mm		
			Region I	Region II	Region III
1	100	0.022	3 x 1.2	3 x 1.2	3 x 1.2
2	200	0.011	3 x 0.6	3 x 0.6	3 x 0.6
3	100	0.022	3 x 1.2	3 x 0.6	3 x 1.2
4	100	0.022	3 x 1.2	4 x 0.9	3 x 1.2

V. Computational Method

The computational solution is based on FLUENT® CFD code for steady-state conditions. For turbulence, k-ε renormalization group theory (RNG) turbulence model with non-equilibrium wall functions is used and for the convective fluxes Roe flux-difference splitting scheme is used which is for compressible flows. Grid system is generated with GAMBIT®. Computational domain is divided into three parts: hot gas domain, coolant domain and chamber wall domain (see Fig. 4) but the inlet and outlet of the solution domain defined including all regions defined in Fig. 3.

Hot Gas Domain: Flow in the hot gas domain is assigned as compressible fluid with a total of seven conservation equations. For the convective fluxes Roe flux-difference splitting scheme is used as mentioned above. Chemical equilibrium code is used to calculate the temperature dependent thermal properties of the fluid. Boundary conditions of hot gas domain are given in TABLE III.

TABLE III
BOUNDARY CONDITIONS FOR HOT GAS DOMAIN

Plane (abh)	$P = P_c,\ T = T_f$
Plane(jks)	$P = P_{amb}$
Plane (abks)	$\frac{\partial u}{\partial n} = \frac{\partial v}{\partial n} = \frac{\partial w}{\partial n} = \frac{\partial T}{\partial n} = 0$
Plane (ahjs)	$\frac{\partial u}{\partial n} = \frac{\partial v}{\partial n} = \frac{\partial w}{\partial n} = \frac{\partial T}{\partial n} = 0$

Coolant Domain: Flow in the coolant domain is assigned as incompressible flow with constant thermal properties [5]. Seven conservation equations are solved but since the flow is incompressible, to solve low Mach number flows Fluent uses a time derivative preconditioning method. Boundary conditions for coolant domain are given in TABLE IV.

TABLE IV
BOUNDARY CONDITIONS FOR COOLANT DOMAIN

Plane (cfgi)	$\dot{m} = \frac{\dot{m}_{LH_2}}{2 \times N}, \; T = T_{LH_2}$
Plane(lmpr)	$P = P_c$
Plane (cilm)	$\frac{\partial u}{\partial n} = \frac{\partial v}{\partial n} = \frac{\partial w}{\partial n} = \frac{\partial T}{\partial n} = 0$

Chamber Wall Domain: The chamber wall material is assigned as copper alloy (NARloy-Z) with temperature dependent thermal properties [6] and the roughness on the inner wall is assigned as 6.5 μm [7]. Only heat conduction equation is solved. Boundary conditions for chamber wall domain are given in TABLE V.

TABLE V
BOUNDARY CONDITIONS FOR CHAMBER WALL DOMAIN

Plane (bcgfideh)	$\frac{\partial T}{\partial n} = 0$
Plane (jklrpmno)	$\frac{\partial T}{\partial n} = 0$
Plane (bck)l	$\frac{\partial T}{\partial n} = 0$
Plane (idmn)	$\frac{\partial T}{\partial n} = 0$
Plane (ehjo)	$\frac{\partial T}{\partial n} = 0$

VI. RESULTS

The purpose of this study is to investigate the efficiency of different channel geometries on cooling. First, the numerical method used in this study is validated with Wadel's study [2]. Test case chosen for the validation is the baseline cooling channel design and the results are given in Fig. 5. Chemical droplet combustion reactions near injector in the chamber were not modeled in this study. Instead, the fluid at the inlet of the chamber is modeled as a single gas, which is a mixture of combustion gas products. It means that gas is already in thermal equilibrium at the inlet section of combustion chamber. Because of this assumption the results are not sufficiently accurate in Region I. Therefore only the solutions in Regions II and III (Fig. 3) are evaluated in this study. Although the general nature of temperature distribution is similar to Wadel's solution, there are differences in the Regions II and III. The difference between Wadel's and the results of this study can be explained as a result of different material properties and the assumption of no droplet combustion.

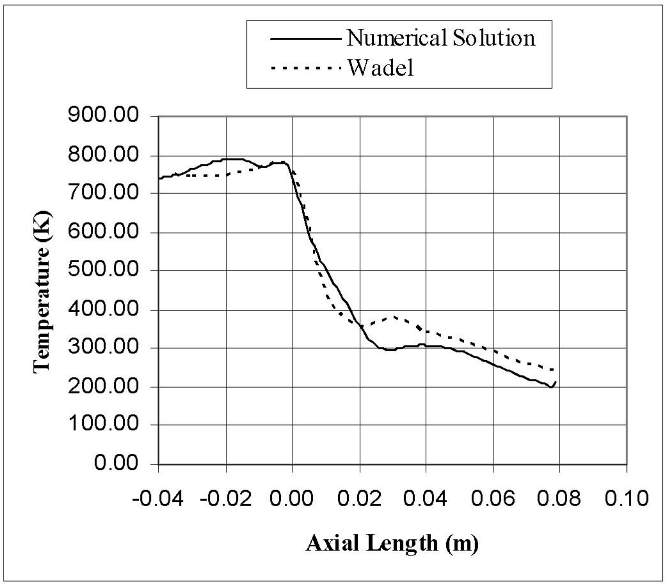

Fig. 5. Hot-Gas Side Chamber Wall Temperature Comparison of Wadel's Solution and Numerical Solution

Four different channel geometries are examined in this study. The objective of the study is to get the lowest temperature on the hot gas side chamber wall to have a maximum life cycle and to get the possible minimum pressure drop in the cooling channel. As can be seen from Equation 1 for the same mass flow rate per unit area, hydraulic diameter should have an effect on hot-gas side wall temperatures. For Alt. 1 and Alt. 2 mass flow rates per unit area are the same and the hydraulic diameter of Alt. 2 is smaller than Alt.1. Therefore it is expected to have a higher coolant heat transfer coefficient and relatively low wall temperatures for Alt. 2 according to Alt. 1 (Fig. 6 and 7).

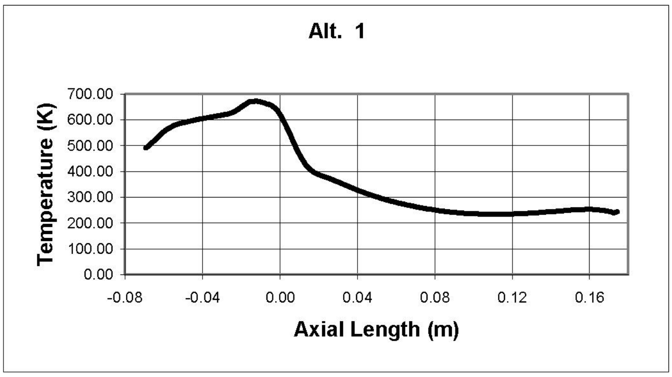

Fig. 6. Hot-Gas Side Chamber Wall Temperature Distribution of Alt. 1

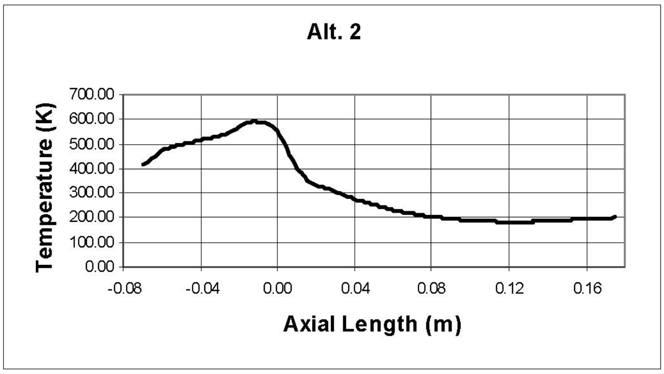

Fig. 7. Hot-Gas Side Chamber Wall Temperature Distribution of Alt. 2

Alt. 3 has a higher mass flow rate per area and lower hydraulic diameter according to Alt. 1 for the Region II. Therefore it is expected to have better coolant efficiency at this area. From Fig. 8**,** for the Region II, Alt. 3 has a better efficiency. When the Regions III of Alt. 2 and Alt. 3 are compared, it can be seen that Alt. 2 has a better efficiency. This can be explained as the rib effect of cooling channels. In Alt. 2 coolant channels are placed thinner and the number of channels is doubled. Therefore coolant efficiency at Region III is better in Alt. 2.

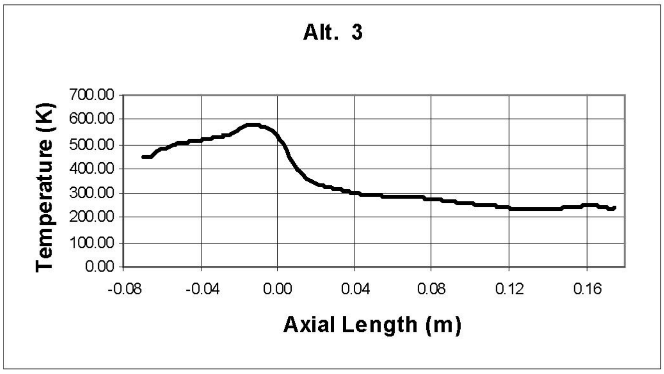

Fig. 8. Hot-Gas Side Chamber Wall Temperature Distribution of Alt. 3

Alt. 4 has the same mass flow rate per area with Alt. 1. Hydraulic diameter of Alt. 4 for the Region II is smaller, therefore has a lower maximum temperature according to Alt. 1 (Fig. 9).

Pressure drops are also critical for cooling channel design. Pressure drops in the cooling channels affects the design of all feeding system in the rocket motor. According to Equation 2 there are two significant parameters which affect the pressure drop; average velocity of coolant along the channel and average diameter of the coolant channel. As expected, the highest pressure drop came from Alt. 3 because of high aspect ratio (low average diameter) and high average velocity in Region II. Alt. 2 also have high pressure drop because of relatively high aspect ratio. Alt. 1 and Alt. 4 have nearly same average velocities and average diameter. Therefore pressure drop values are close to each other. The pressure drops and maximum temperatures on the hot-gas side wall for each design given in TABLE VI.

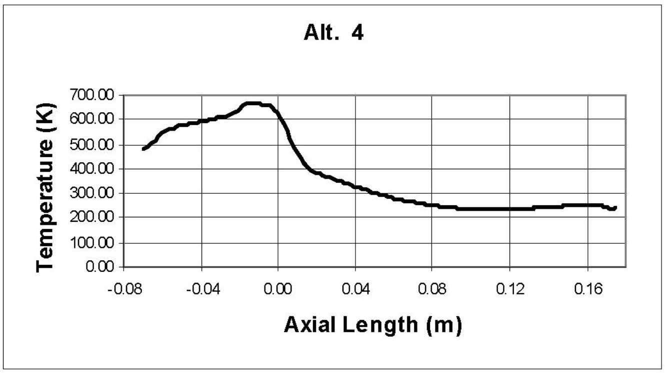

Fig. 9. Hot-Gas Side Chamber Wall Temperature Distribution of Alt. 4

TABLE VI
MAXIMUM HOT-GAS SIDE CHAMBER WALL TEMPERATURE AND TOTAL PRESSURE DROP FOR EACH DESIGN

Alt.	Maximum Hot-Gas Side Chamber Wall Temperature (K)	Pressure Drop along the Channel (atm)
1	673	16.6
2	592	32.6
3	579	51.6
4	668	17.8

From the initial calculations Alt. 3 with 100 cooling channels having 3 x 0.6 mm^2 cross-section area gives the best result in terms of lower wall temperature (579 K) but gives the highest pressure drop (51.6 atm). As a result, to use high aspect ratio channels just for the critical regions instead of the whole region will decrease the hot-gas side chamber wall temperature on this region and pressure drop relatively.

VII. DISCUSSION AND CONCLUSION

In this study, the effect of channel geometry on regenerative cooling effectiveness is studied and an analysis method was developed and validated. In this method, the following design rules for cooling channels are observed:

- An increase in the number of cooling channels increases the cooling efficiency.
- An increase in the aspect ratio with constant height increases the cooling efficiency.
- An increase in the aspect ratio with constant height increases the pressure drop.

The primary objective of this research is to investigate possible cooling techniques for rocket combustion chamber and nozzles. Regenerative cooling is a well-established method from the heat transfer point of view. However selection of structure and channel materials in the vicinity of high temperature gradients is still being investigated.

This article describes the initial work of such a comprehensive investigation. A preliminary design methodology will be validated for regenerative cooling system when this research is finished.

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