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A SIMPLE APPROACH FOR THERMAL ANALYSIS OF REGENERATIVE COOLING OF ROCKET ENGINES

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ABSTRACT

A simple model for thermal analysis of regenerative cooled rocket engines is developed. In this model the multi-dimensional heat conduction in the engine wall is analyzed using the fin effect of cooling channel side walls. A one-dimensional model for the cooling channel flow and heat transfer is used. The coolant properties are evaluated based on the NIST database. The present model is used to perform thermal analysis on two engines: the Space Shuttle Main Engine (a liquid hydrogen cooled engine) and a liquid oxygen cooled engine. Given the simplicity of the present approach its results compare well with other comprehensive models.

NOMENCLATURE

A_c	cooling channel cross-section area
ch	cooling channel height
cw	cooling channel width
d_h	cooling channel hydraulic diameter
D_i	chamber diameter at station i
e	cooling channel surface roughness
f	friction factor
h	coolant heat transfer coefficient
i	coolant enthalpy
k_w	wall conductivity
L_i	length of cooling channel at station
$\dot{m}_c$	coolant mass flow rate
N_{cc}	number of cooling channels
P	coolant pressure
Pr	Prantl number
q_{gas}	heat flux of combustion gases
R_{th}	thermal resistance
Re	Reynolds number
t_i	wall thickness at station i (distance between the bottom of cooling channels and hot gas-side wall)
V	coolant velocity

Greek symbols

δ_i	distance between two cooling channels at station i
ε	cooling channel roughness
η_f	fin efficiency

Subscripts

c	coolant
i	station i

o corresponds to stagnation properties
s corresponds to static properties
w wall
x corresponds to the film properties

1.0 INTRODUCTION

For high-pressure liquid rocket engines (LRE's), hot-gas in the throat area may reach temperatures as high as 7000 R. Therefore, it is essential to cool the engine ensuring that the wall material withstands the high temperatures. In addition, using the fuel/oxidizer as the coolant increases the enthalpy prior to combustion, resulting in a more efficient combustion. Single Circuit Regenerative cooling is a widely used method to reduce the wall temperatures and increase coolant enthalpy for high-pressure LRE's.

In the regenerative cooling rocket engine, the coolant which is either fuel or oxidizer enters cooling passages at the nozzle exit and travels through the passages machined in the engine wall (see Figure 1). This method serves two purposes: 1) keeps the engine walls cool and, 2) increases coolant enthalpy. In some engines, such as the Space Shuttle Main Engine (SSME) the coolant, which is Liquid Hydrogen (LH_2) coming out of cooling channels is used to run turbo-pumps. Also, the increased enthalpy of Liquid Hydrogen makes the combustion process more efficient.

The existing models for thermal analysis of regenerative cooled rocket engines are based on sophisticated computational models. The model discussed in [1] involves iterations between two large computer codes, TDK (Two Dimensional Kinetics) [2] and RTE (Rocket Thermal Evaluation) [3]. Another model discussed in the literature involves interactions among a CFD model for the hot gas-side calculation; a three dimensional wall conduction model and a coolant flow heat transfer model (see [4]). Most recently, a CFD model, based on the Fluent software [5], is combined with the RTE to perform a comprehensive thermal analysis of regenerative cooled rocket engines. These models, due to their complexity require long computational time to run, in some cases more than a day. Often, the design of regenerative cooling circuit involves many design iterations which require quick computation results. To address the need for a fast turnaround time a new computational model is developed. In this model the multi-dimensional heat conduction in engine wall is analyzed using the fin effect of cooling channel side walls. A one-dimensional model for cooling channel flow and heat transfer is used. The coolant properties are evaluated based on the NIST database [7].

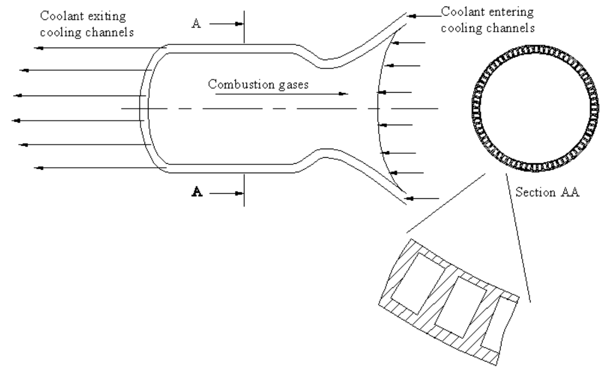

Figure 1: Configuration of the cooling circuit of regenerative cooled rocket engine

To examine the accuracy of the model its results are compared to the existing data for the Space Shuttle Main Engine (SSME), a hydrogen cooled engine, and an oxygen cooled engine. The results show good agreement between the results of the present approach and those of existing models.

2.0 THE COMPUTATIONAL MODEL

In this numerical model, the rocket thrust chamber and nozzle are subdivided into a number of stations along the longitudinal direction, as shown in Figure 2. These stations do not have to be equally spaced; in fact, it is desirable to put more stations near the throat where the heat flux and temperature gradients are largest. The first station is located at the exit of the engine where coolant enters the cooling passages. At the first station the coolant properties, e.g. its temperature and pressure, are known. The fluid transport and thermodynamic properties at each station are evaluated using the NIST database (Refprop program) [7]. The coolant velocity, Reynolds and Nusselt numbers, and heat transfer coefficients are evaluated using appropriate correlations. To calculate the heat transfer to the coolant from hot-gases a simple one dimensional model using the fin effect of cooling channel side walls is used. The cooling channels of rocket engines are normally high aspect ratio, especially at the throat area of the engine (see Figure 3 for cross-section of a high-aspect-ratio cooling channel). Hence, treating the side walls of cooling channels as fin is a good assumption. To maximize the fin effect of cooling channels, when it is possible, they are made with a high aspect ratio, i.e. large ch/cw.

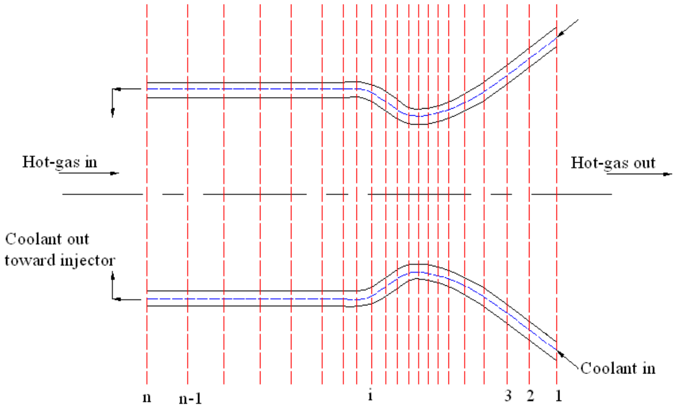

Figure 2: Schematics of an engine broken into a number of stations

For the analysis purpose consideration is given to the half cooling channel as shown in Figure 4. Because of the symmetry of the configuration, computations are performed for only one cell. Since no heat is transferred to the two sides of the cell, they are assumed insulated. It should be noted that all of the heat transferred to the coolant are from the side wall and the bottom of cooling channel. Hence, a simplified one-dimensional heat transfer model for regeneratively cooled rocket engine can be represented by a set of thermal resistances shown in Figure 5. One side is the temperature of wall exposed to combustion gases (T_w) and the other side is the coolant temperature (T_c). The overall thermal resistance between the wall and coolant is expressed by:

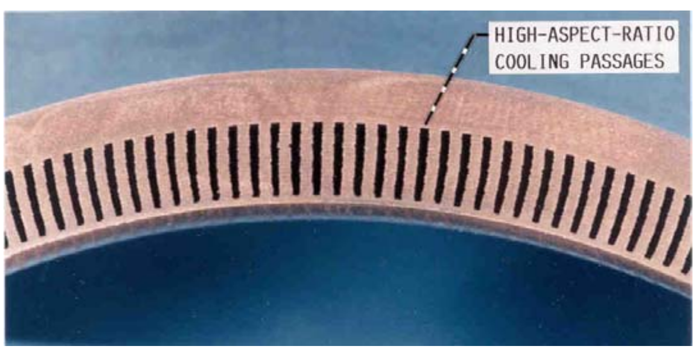

Figure 3: Cross-section of a high-aspect-ratio cooling passage

$$R_{th_i} = \frac{1}{h_i L_i (2\eta_{f_i} ch_i + cw_i)} + \frac{\ln\left(1 + \frac{2t_i}{D_i}\right) N_{cc}}{2\pi L_i k_w} \quad (1)$$

The fin efficiency in the above equation is given by [8]:

$$\eta_{f_i} = \frac{1}{m\,ch}\tanh(m\,ch) \quad (2)$$

Where $m = \sqrt{\frac{h_i}{k_w \delta_i}}$ and δ_i is the distance between two cooling channels at station i.

The coolant is assumed well mixed at a given cooling channel cross-section. Several correlations are reported in the literature for calculating the convective heat transfer coefficient h_i. These correlations are based on the type of coolant. For liquid hydrogen Diprey and Sabersky [4] and Hendrick [9] correlations can be used. Diprey and Sabersky's correlation, used in this work, is given by:

$$Nu = \frac{(f/8)\mathrm{Re}_{CS}\,\mathrm{Pr}_{CS}\left(\frac{T_{CS}}{T_{CW}}\right)^{0.55}}{\left(1 + (f/8)^{0.5}\left[B(\varepsilon^*) - 8.48\right]\right)} \quad (3)$$

Where

$$\varepsilon^* = \mathrm{Re}_{CS}.(e/d_h).(f/8)^{0.5}$$

$$B(\varepsilon^*) = 4.7(\varepsilon^*)^{0.2} \qquad \text{for } \varepsilon^* \geq 7$$

$$B(\varepsilon^*) = 4.5 + 0.57(\varepsilon^*)^{0.75} \qquad \text{for } \varepsilon^* < 7$$

The friction factor f in equation (3) is evaluated using the Colebrook correlation:

$$\frac{1}{\sqrt{f}} = -2\log\left(\frac{e}{3.7065 d_h} + \frac{2.5226}{\mathrm{Re}_{CX}\sqrt{f}}\right) \quad (4)$$

The friction factor evaluated based on the above equation is also used to evaluate the viscous pressure drop in the cooling channels. When the coolant is liquid oxygen the correlation given Spenser and Rousar [10] can be used. To account for the curvature and entrance effects correction factors for Nusselt number given in [9] are used.

Once the wall thermal resistances are determined the hot-gas-side wall temperature at station *i* can be determined via:

$$T_{w_i} = R_{th_i} . q_{gas_i} + T_{c_i} \quad (5)$$

where q_{gas_i} is the heat flux from hot-gases (combustion gases) to the engine wall. There are a number of approaches that can be used to evaluate the wall heat flux. The simplest approach is the use of the public-domain NASA software CEA (Chemical Equilibrium with Applications) [11] along with heat transfer correlations given in [12]. Other approaches involved use of software, such as TDK (Two Dimensional Kinetics) [2], or CFD programs. Most comprehensive thermal analysis models for regeneratively cooled rocket engines use an interactive approach to iterate between the wall conduction-cooling-channel-convection and the hot-gas heat flux programs. The process can take a long time to converge. In the present approach a look-up table of heat fluxes for all axial position and a number of temperatures, starting with the lowest to the highest wall temperatures is generated. Figure 6 shows the plot of the wall heat flux for the SSME engine for all axial locations and temperatures from 540 R (smallest temperature) to 1500 R (largest temperature). It should be noted that the negative axial location in Figure 6 correspond to upstream of the throat and positive values correspond to the upstream of the throat. The most critical location for cooling is the downstream of the throat (axial location ≈ -1 in), where the wall heat fluxes are the largest.

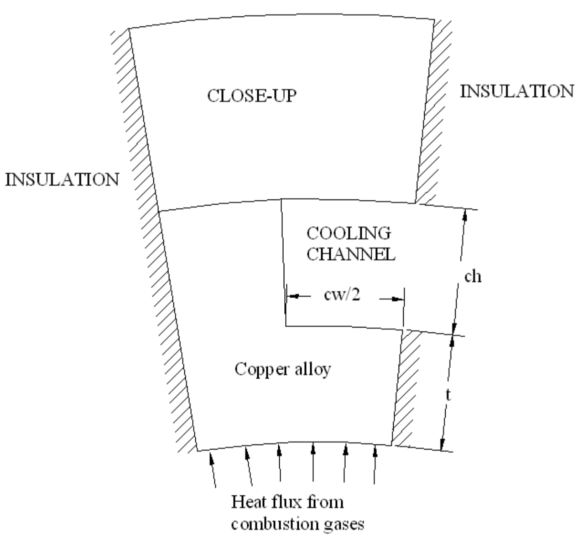

Figure 4: Configuration of one cell cooling channel

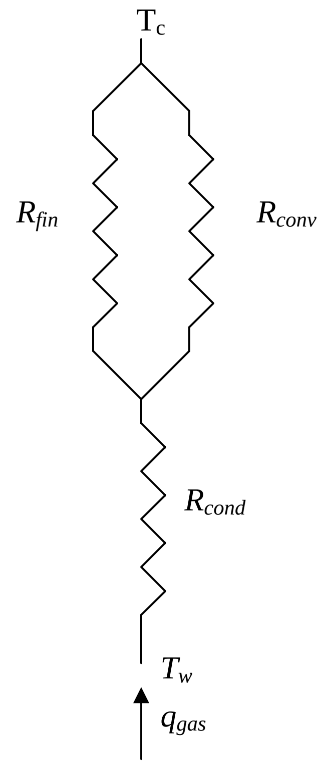

Figure 5: An equivalent thermal resistances for heat transfer in a regeneratively cooled rocket wall

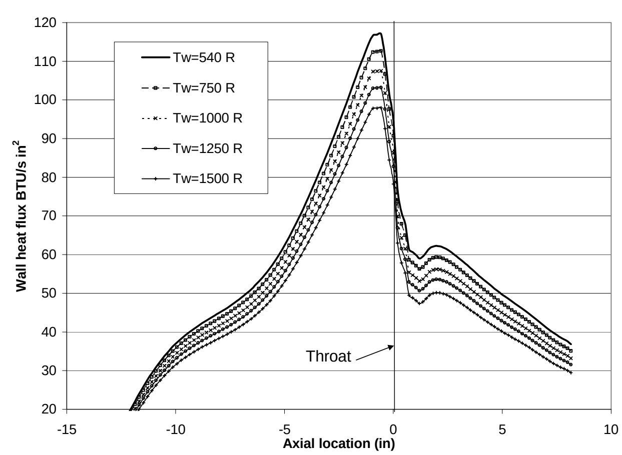

Figure 6: Wall heat flux for the SSME for different wall temperatures at all axial locations

The coolant stagnation enthalpy at the next station (station $i+1$) is evaluated by writing the coolant energy balance equations between stations i and $i+1$, as given by:

$$i_{co_{i+1}} = i_{co_i} + \frac{q_{gas_i} \Delta A_{i,i+1}}{\dot{m}_c} \quad (6)$$

The value of heat flux, in the above equation, is assumed to be constant and equal to that of station i . This is a good assumption as long as the spacing between stations, especially at location with large heat flux variation, is kept small. $\Delta A_{i,i+1}$ in equation (6) is the nozzle wall area between stations i and $i+1$, and $\dot{m}_c$ is the coolant mass flow rate.

The pressure drop between stations i and $i+1$ consists of three terms, friction (viscous), momentum and area change (contraction and expansion) pressure drops, as given by the following equation:

$$\Delta P_{i,i+1} = \Delta P_{v_{i,i+1}} + \Delta P_{c-e_{i,i+1}} + \Delta P_{m_{i,i+1}} \quad (7)$$

The viscous pressure drop can be evaluated using the Darcy's law

$$\Delta P_{v_{i,i+1}} = \frac{f_i \rho_i V_i^2 \Delta L_{i,i+1}}{2 d_{h_i}} \quad (8)$$

The expansion and contraction pressure drop is calculated via:

$$\Delta P_{c-e_{i,i+1}} = \frac{K \rho_i V_i^2}{2}$$

For the expansion $K = \left[\left(\frac{d_{h_i}}{d_{h_{i+1}}} \right)^2 - 1 \right]^2$

For the contraction:

$$K = 0.5 - 0.167 \frac{d_{h_{i+1}}}{d_{h_i}} - 0.125 \left(\frac{d_{h_{i+1}}}{d_{h_i}} \right)^2 - 0.208 \left(\frac{d_{h_{i+1}}}{d_{h_i}} \right)^3$$

The momentum pressure drop results from fluid acceleration and is given by:

$$\Delta P_{m_{i,i+1}} = \left[\frac{2}{(A_C.N)_i + (A_C.N)_{i+1}} \right] \dot{m}_C^2 \left[\frac{1}{(\rho_{CS}.A_C.N)_i} - \frac{1}{(\rho_{CS}.A_C.N)_{i+1}} \right] \quad (9)$$

Knowing the pressure drops between stations $i+1$ and i the next station's pressure can evaluated via:

$$P_{i+1} = P_i - (\Delta P_{v_{i,i+1}} + \Delta P_{c-e_{i,i+1}} + \Delta P_{m_{i,i+1}}) \quad (10)$$

After evaluation the coolant enthalpy and pressure (using equations (6) and (10), respectively) other coolant properties at station $i+1$ can be evaluated from the NIST database [7]. By marching from the first station (entrance of the cooling channels) to the last station (exit of the cooling channels) and implementing the procedure described here all thermodynamics and transport properties of coolants, as well as wall temperature, are evaluated.

3.0 RESULTS AND DISCUSSIONS

To examine the accuracy of the present approach it is used to analyze the regenerative cooling circuit of the SSME (Space Shuttle Main Engine). This engine is a public domain engine and has been analyzed by a number of models (see [1] and [4]). The SSME engines were that studied in the previous works have the same nozzle and chamber diameters, but there are slight differences in their cooling circuit designs. In the present work the cooling circuit design of reference 1 is used. The resulting maximum wall temperature distributions (wall temperature at the hot-gas-side of the wall) for the SSME based on the present approach and other models are shown in Figure 7. The maximum wall temperature in design of liquid rocket engines is an important parameter since it must be kept below the wall material limit. The wall material for the SSME is NARloy-Z (a copper alloy), which has a thermal limit of approximately 1600 R. As shown in this figure the results of the present approach compare well with the results of RTE-TDK method based on Diprey and Sabersky's (D & S) correlation. The results based on the Wang's model [4] over-predicts the wall temperature at the chamber section of the engine (x<0) which show a temperature very close to the wall material limits.

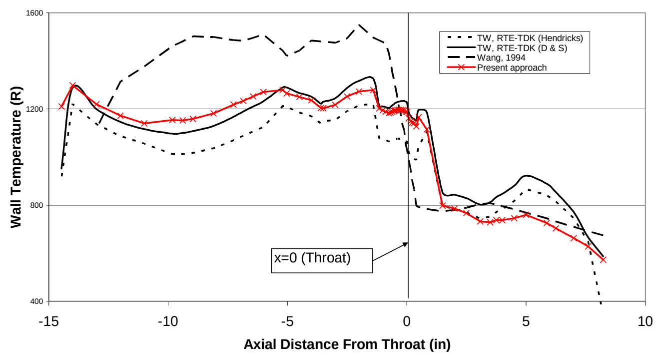

Figure 7: Hot-gas wall temperature comparison between the present approach and those of other methods for the SSME engine

Figure 8 shows the comparison between the stagnation coolant temperature of the SSME along the axial locations based on the present approach and that of RTE-TDK method. This figure shows that the present approach over-predicts the coolant temperature. The results based on the two methods are almost identical at the entrance section of the cooling channel (x>5 in). The difference between the two methods grows as the coolant travels downstream of the cooling channel. The largest difference is at the exit of the cooling channel. The reason for this increase in the difference between the coolant temperatures based on these two methods is due to the fact that at each station the present approach slightly over-predicts the coolant temperature. Since the properties of the next station depend on the properties of previous station this temperature over-prediction grows to a larger value at the exit of the coolant channel. A similar trend can be observed in the coolant pressure distribution as shown in Figure 9. The present approach over-predicts the coolant stagnation pressure (under-predicts the pressure drop). The results of pressure distribution based on the present approach and that of RTE-TDK are in excellent agreement at diverging section of the engine. The results, however, divert in the thrust chamber section.

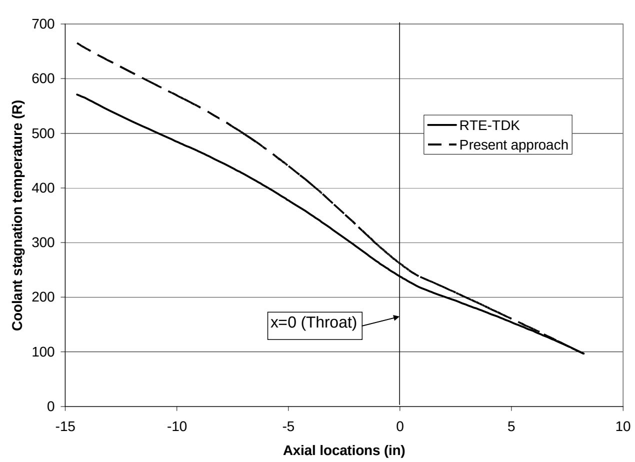

Figure 8: Comparison coolant temperature of the SSME based on the present approach and that of RTE-TDK

To further examine the accuracy of the present approach consideration is given to a hydrocarbon fuel engine with liquid Oxygen as coolant. The specifications of this engine are:

Chamber pressure	2000 psi
O/F (mixture ratio)	1.8
Contraction ratio	3.4
Expansion ratio	7.20
Throat diameter	2.6 inch
Propellant	RP1-LO_2 ($C_{13}H_{23}$-LO_2)
Coolant	LO_2
Total coolant flow rate	32.893 lb/s
Coolant inlet temperature	160 R
Coolant inlet pressure	3000 psi
Number of cooling channels	100
Throat region channel aspect ratio	2.5

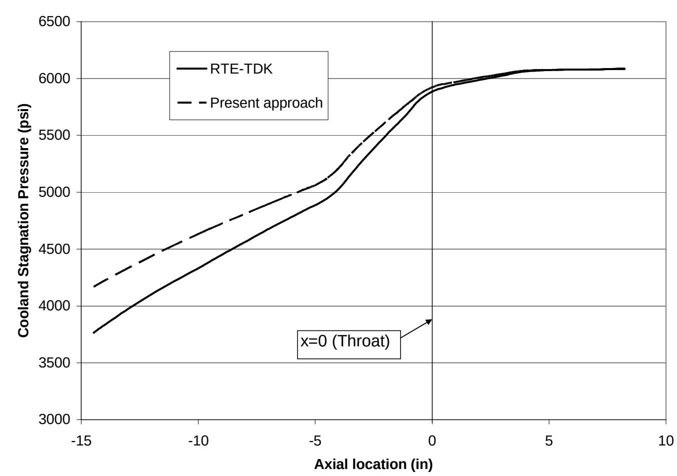

Figure 9: Comparison coolant pressure of the SSME based on the present approach and that of RTE-TDK

Figure 10 shows the contour of this thrust chamber and nozzle. The resulting maximum wall temperatures for this engine based on the present approach and those of RTE-TDK are shown in Figure 11. As shown in this figure the results of the present approach compares excellently to those of RTE-TDK for the high temperature area of the engine (upstream of the throat). For the other parts, the present approach slightly under-predicts the maximum wall temperature. For the coolant temperature variation along the cooling channels, as shown in Figure 12, the results of the present approach is in excellent agreement for those of RTE-TDK.

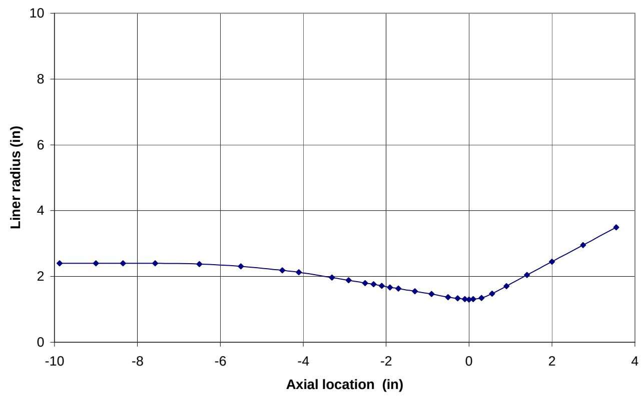

Figure 10: RP1-LO_2 rocket thrust chamber and nozzle contour showing station locations

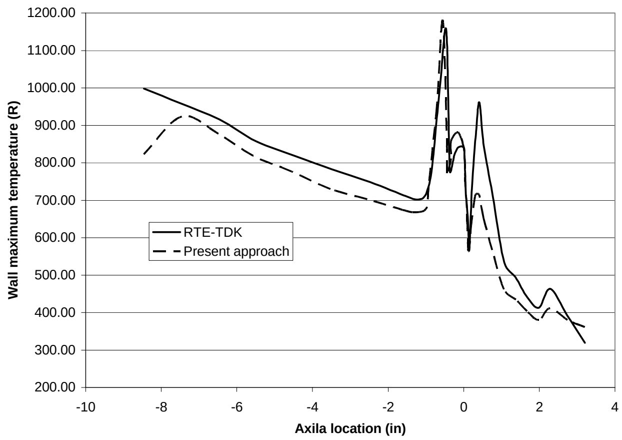

Figure 11: Comparison between the maximum wall temperature distributions of an oxygen cooled engine based on RTE-TDK and the present approach.

4.0 CONCLUSION

A simple approach for design of cooling circuits of regenerative cooled rocket engines is developed. Comparison of the results of this approach to those of other models shows that the present model makes an accurate prediction of the maximum wall temperatures at throat area of the engine. Determination of the cooling channel dimensions at throat area is a critical part of the cooling circuit design. This is due to the fact that the largest heat flux is at the throat area and the available area is small to machine cooling channels.

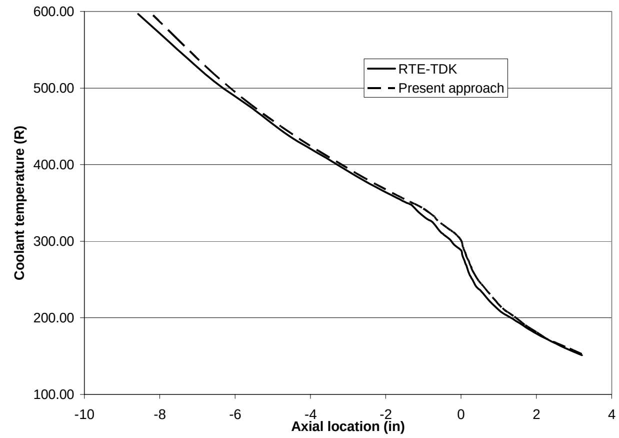

Figure 12: Comparison between the coolant temperature variations of an oxygen cooled engine based on RTE-TDK and the present approach

The present model can be easily programmed using spreadsheets or other programming languages. The results presented in this paper were performed using Microsoft Excel software. The computations for the cases studied in this work were almost instantaneously. The short turnaround time of this model makes it ideal for the initial design of cooling circuit. Further refinements of the design can be made using more comprehensive multidimensional models.

5.0 REFERENCES

1. Naraghi, M.H., Dunn, S. and Coats, D., "A Model for Design and Analysis of Regeneratively Cooled Rocket Engines," AIAA-2004-3852, presented at the *Joint Propulsion Conference*, Fort Lauderdale 2004.

2. Dunn, S.S., Coats, D.E., and French, J.C., "TDK'02™ Two-Dimensional Kinetics (TDK) Nozzle Performance Computer Program", User's Manual, prepared by Software & Engineering Associates, Inc., Dec 2002.

3. Naraghi, M.H., "RTE - A Computer Code for Three-Dimensional Rocket Thermal Evaluation", User Manual, Tara Technologies, LLC, Yorktown Heights, NY 2002.

4. Wang, T.S. and Luong, V. "Hot-Gas-Side and Coolant-Side Heat Transfer in Liquid Rocket Engine Combustor", AIAA Journal of Thermo-physics and Heat Transfer, 1994, Vol. 8, No. 3, pp 524- 530.

5. Fluent Reference Manual, Software Release Version 6.3, Fluent Inc. 2007.

6. Jokhakar, J. and Naraghi, M.H., "A CFD-RTE Model for Thermal Analysis of Regeneratively Cooled Rocket Engines", to be presented at the AIAA Joint Propulsion Conference, Hartford, Connecticut, July 27-30, 2008.

7. Eric W. Lemmon, "NIST Reference Fluid Thermodynamic and Transport Properties-REFPROP", Physical and Chemical Properties Division, National Institute of Standards and Technology, Boulder, Colorado 80305, Version 7.0, 2002

8. F.P. Incropera and D.P. DeWitt, "Introduction to Heat Transfer", Wiley, Third Edition, p.125, 1996.

9. Hendricks, R. C., Niino, M., Kumakawa, A., Yernshenko, V. M., Yaski, L. A.,Majumdar, L. A., and Mukerjee, J., "Friction Factors and Heat Transfer Coefficients for Hydrogen Systems Operating at Supercritical Pressures", Proceeding of Beijing International Symposium on Hydrogen Systems, Beijing, China, May 7-11, 1985.

10. Spencer, R.G. and Rousar, D.C., "Supercritical Oxygen Heat Transfer," NASA, CR-135339, 1977.

11. McBride, B. J. and Gordon, G., NASA Computer program CEA, Chemical Equilibrium with Applications, http://www.grc.nasa.gov/WWW/CEAWeb/.

12. Bartz, D. R., "Turbulent Boundary-Layer Heat Transfer from Rapidly Accelerating Flow of Rocket Combustion Gases and of Heated Air," Advances in Heat Transfer, pp. 2-108, 1965.