Thermal Analysis of Cryogenic Rocket Engine with One, Two and Three Dimensional Approaches

Kuzhiveli B.T.[*], Ghosh S.C., Kuruvila G. K. and Gandhi V.G.

Liquid Propulsion Systems Centre, Indian Space Research Organisation,
Valiamala, Trivandrum, India - 695 547
[*]Presently at DCI, 112 Nila, Techno Park, Trivandrum, India - 695 581, Email: btkuzhiveli@yahoo.com

Prediction of heat transfer characteristics in a regenerative cooled cryogenic rocket engine, which uses liquid oxygen (LOX) and liquid hydrogen (LH2) as propellants provides one of the major inputs while designing a rocket engine. The objective of this paper is to present thermal analysis with one, two and three-dimensional approaches. Details of heat transfer analysis program that can predict temperature distribution on the thrust chamber, hot side and coolant side heat transfer coefficients, coolant temperature rise and heat flux have been presented. The thermal analysis program thus developed can be well suited for parametric studies to recommend stable operation during actual hot test conditions.

INTRODUCTION

Modern cryogenic rocket engine thrust chambers are exposed to high pressure and high heat flux environments due to high energy combustion. This has presented some of the most challenging engineering problems due to the presence of high heat flux and high pressure. These extreme conditions are further complicated by the fact that an engine should achieve a bare minimum weight and deliver a high specific impulse. A commonly employed method to overcome the problem due to high temperature arising out of high energy combustion is by providing regenerative cooling by passing LH2 through channel passages in the thrust chamber wall.

The design methodology of rocket engine heavily relies on an accurate prediction of temperature and pressure profile on the engine. The physical phenomenon involved in the operation of cryo engine is complicated. The flow in the engine is turbulent, reactive and undergoes both subsonic and supersonic flow regimes. As the hot test of engineering hardware can be prohibitively expensive, numerical simulations of the conditions are necessary. Hence it was essential to develop a program, which could accurately predict the thermal characteristics of the thrust chamber.

COMPUTATIONAL MODEL

To make the analysis computationally traceable it is assumed that the flow through the channel is circumferentially symmetric. As a result, 360^{o}/Number of channel becomes the area of attention. This simplification implies that only half rib and half channel have to be analyzed. The approach holds good if the flow in different channel are the same. If not, it represents the performance of an average channel and assumes that the departure from the performance of an average channel is not significant.

With this in hand, there are three possible ways of having thermal analysis. i.e., one dimensional, two dimensional and three dimensional. In the one dimensional approach, the two possibilities could be, to consider the fin effect or not to consider the fin effect. In both the cases, it is assumed that the heat from hot gas is passed to the coolant channel directly. The major draw back of this method is that conduction factor through the ribs and the outer shell is neglected and it is further assumed that heat passes directly into the cooling channel through a single point entry. It also does not take into account the radiation effects from the hot gas as well as the ambient in an iterative manner. In addition to this, the axial conductivity between the marching sections is also not considered.

In the two dimensional approach, the independent cross sectional domain (x-y direction) is divided into many grids and the grid meeting points are considered as nodes. Each node is connected to the adjacent node by a connecting line. An energy balance is applied to each node, which results in an algebraic equation for the temperature at the node. Separate equations are derived for each node in the control volume. The drawback here is that the axial conduction (in z direction) is neglected.

For the three dimensional approach, the control volume becomes a solid space with thrust chamber cross sectional area as the x-y co-ordinate and the marching distance as the z co-ordinate. The heat transfer model takes into consideration the following five physical aspects (i) heat transfer from combustion gas to the inner wall of the thrust chamber (including convection and radiation effects) (ii) heat transfer through the inner wall, ribs and outer wall through grids (incorporating varying thermal conductivity effects) (iii) heat transfer from the walls and ribs to the coolant through the grids(incorporating varying heat transfer coefficient effects) (iv) heat transfer from the ambient to the outer wall (or vice versa, includes convection and radiation effects) (v) heat transfer between the sections (in the axial direction). Electrical analog of three-dimensional computational model is depicted in Figure1.

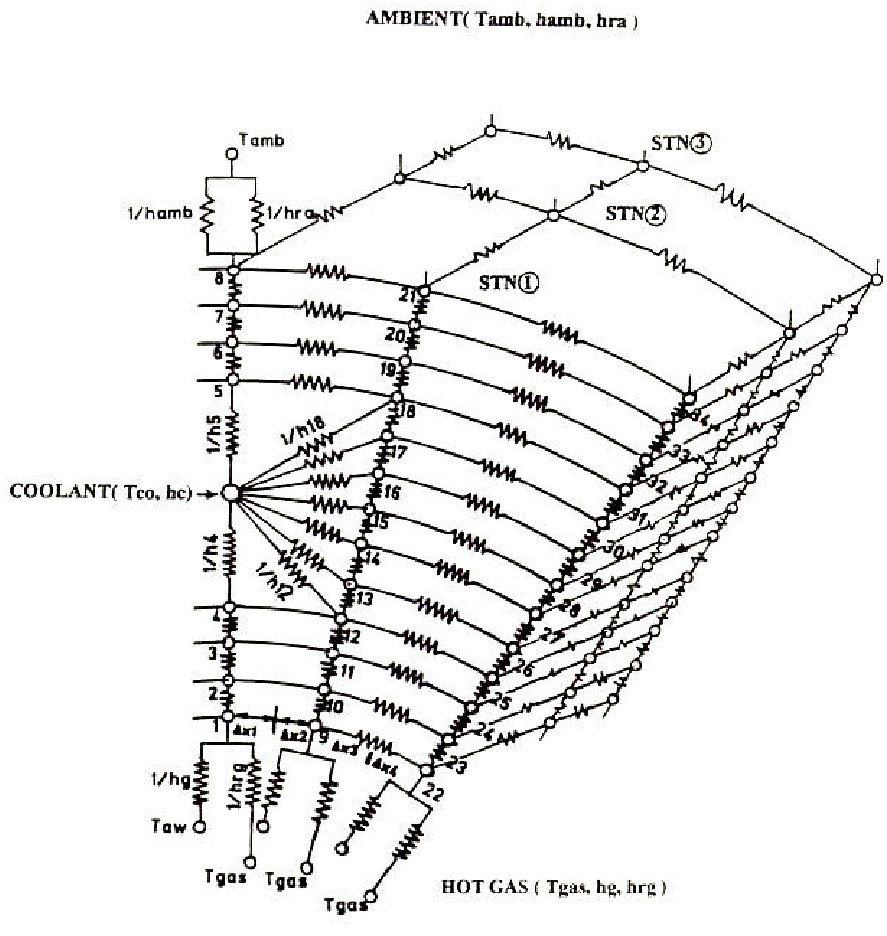

Figure 1 Electrical analog of three dimensional computational model

CALCULATION PROCEDURE

In order to determine the performance of a rocket engine for a given propellant combination and chamber pressure, the composition of the combustion products is assumed to be in chemical equilibrium. The theory presented by Gordan et al [1] is utilised to determine the combustion gas properties.

Throughout the axial length of the engine from the injector end to divergent end, all the combustion gas properties are obtained for a given chamber pressure and mixture ratio. A forward marching method is adopted for finding out the temperature distribution, coolant temperature, heat transfer coefficients, heat flux, and the bulk coolant temperature rise.

Within the thrust chamber, both convection and radiation contribute to the total heat transferred. Before the gas can transfer heat to the wall of the thrust chamber, the heat energy passes through a layer of stagnant gas along the boundary layer. Hydrogen enters the channels in liquid/vapor condition and is in contact with the inner part of the walls and ribs. Hydrogen absorbs the heat energy and thereby raising the bulk coolant temperature.

Heat transfer coefficient hg is predicted with the help of simplified Bartz equation [2]. The coolant heat transfer co-efficient can be estimated by Sieder Tates equation [3].

<u>One dimensional analysis (1D</u>)

In one dimensional analysis, the heat flux has been equated as shown in the equation below.

$$q1 = hg1\,(Taw1 - Twg1) = (Twg1-Twc1)/(L/kA) = hc1(Twc1- Tco1)$$

Where q - heat flux, hg - heat transfer coefficient of hot gas, Taw - adiabatic wall temperature, Twg - temperature on the hot gas side wall, Twc - temperature on the coolant side wall, L - heat passage length, k - conductivity of the material, A - area.

To find out the wall temperature (Twg), an initial value for Twg is assumed and hg, Taw, Twc, hc are found out. Value of Twg is increased step by step till the equation for q1 satisfies the above condition. The subscript 1 denotes for one-dimensional analysis.

The increase in coolant temperature at each axial location is calculated by Tco1_next = Tco1 + q1 / (mc.Cpc) where mc and Cpc are mass of coolant (hydrogen) and specific heat of coolant respectively.

<u>Two dimensional analysis (2D</u>)

Heat transfer analysis of regeneratively cooled thrust chamber has been carried out using two dimensional finite difference method. The computational domain consists of half the channel width and half the rib width because of axi-symmetry and its repetitive nature. Each subdivided area has a length ?xn in the ‘x’ direction and length ?yn in ‘y’ direction. Separate equations are derived for each node in the control volume.

On the hot gas side wall, convective and radiative boundary conditions are imposed. Convective boundary conditions are also imposed on the coolant side. Similarly on the outer skin of the chamber, convective and radiative boundary conditions are imposed. For steady state conditions, an energy balance is applied, as an example, at node 2 (refer Figure 1)

$$\overset{i?4}{\underset{i?1}{?}}\, qi \;?\; 0$$

i.e.; $q1\text{-}2 + q3\text{-}2 + q10'\text{-}2 + q10\text{-}2 = 0$

Applying Fourier’s law to each of these terms to express the equation in terms nodal temperatures,

$$q_{1-2} = k_{1-2}.L\frac{(?xn+?x^{(-)}n)\,(T1-T2)}{2.?yn} \qquad q_{3-2} = k_{3-2}.L\frac{(?xn+?x^{(-)}n)\,(T3-T2)}{2.?yn}$$

$$q_{10'-2} = k_{10'-2}.L\frac{(?yn+?yn)\,(T10'-T2)}{2.?xn} \qquad q_{10-2} = k_{10-2}.L\frac{(?yn+?yn)\,(T10-T2)}{2.?xn}$$

<u>Three dimensional analysis (3D</u>)

For steady state conditions, an energy balance is applied, as an example, at node 2 (refer Figure 1);

$$\overset{i?6}{\underset{i?1}{?}}\ qi \ ?\ 0$$

i.e.; q2'-2 + q2''-2+ q1-2 + q3-2 + q10-2 + q10'-2 = 0

$$q_{2'-2} = k_{2'-2}.L_z\frac{(?\,xn + ?\,x^{(-)}n)(?\,yn+?\,yn)(T2' - T2)}{2*2*(?\,z^{(-)}n)+?\,zn)/2} \qquad q_{2''-2} = k_{2''-2}.L_z\frac{(?\,xn + ?\,x^{(-)}n)(?\,yn+?\,yn)(T2'' - T2)}{2*2*(?\,z^{(-)}n)+?\,zn)/2}$$

$$q_{1-2} = k_{1-2}.L_z\frac{(?\,xn + ?\,x^{(-)}n)(?\,z^{(-)}n)+?\,zn)(T1-T2)}{2*2*(?\,z^{(-)}n)+?\,zn)/2} \qquad q_{3-2} = k_{3-2}.L_z\frac{(?\,xn + ?\,x^{(-)}n)(?\,z^{(-)}n)+?\,zn)(T3- T2)}{2*2*(?\,x^{(-)}n/2)}$$

$$q_{10'-2} = k_{10'-2}.L_z\frac{(?\,yn+?\,yn)\ (?\,z^{(-)}n)+?\,zn)(T10' - T2)}{2*2*?\,yn/2} \qquad q_{10-2} = k_{10-2}.L_z\frac{(?\,yn+?\,yn)\ (?\,z^{(-)}n)+?\,zn)(T10- T2)}{2*2*(?\,xn/2)}$$

The result of the finite difference method is 'N' algebraic equations for N nodes and these equations replace the single partial differential equations and applicable boundary conditions. For each node, the temperature is calculated with changing properties of the material used, in an iterative manner. A computer program has been written down to include computational models of 1D, 2D and 3D analysis.

RESULTS

The results obtained from the three dimensional analysis has been compared with the results of the two dimensional as well as the one dimensional analysis. Temperature profiles for all marching distances (1mm in critical area) over the axial length are found out using the above methods. Inner wall temperatures predicted with 1D, 2D, 3D methods were as follows; at the beginning of the thrust chamber, the predicted inner wall temperatures were 616 K, 519 K, 514 K respectively. Highest temperature observed near the throat was 692 K, 589 K, and 583 K respectively. At the hydrogen inlet (divergent end) these were 606 K, 501 K, and 496 K respectively. The temperatures at 34 points on the cross section (refer Figure 1), T1 to T34 has been found out. As an example, at throat section, the temperatures at points 1 to 34 were 583K, 553 K, 525K, 501K, 335K, 358K, 360 K, 360 K, 582K, 552K, 521K, 489K, 451K, 421K, 397K, 378K, 364K, 353 K, 359 K, 360 K, 360 K, 583 K, 553 K, 523 K, 493 K, 464 K, 438 K, 417 K, 400 K, 388 K, 383 K, 362 K, 360 K, 360 K.

The temperatures predicted by 3D analysis were much lower than that predicted by 1D analysis and slightly lower than the 2D analysis. This difference existed because 3D model takes into account all heat transfer aspects. The experimental measurement of bulk coolant temperature was made to evaluate the suitability of 3D analysis. The bulk coolant temperature measured at exit of the channel was 247.0 K against the predicted value of 258 K (by 3D analysis). A small variation (about 4%) in the prediction could be attributed to the film cooling effect, which was not incorporated in the analysis. Thus the suitability of 3D program has been validated. The program thus developed is well suited for design of regeneratively cooled engines as well as a tool to check the safe operation of the engine during hot test conditions.

ACKNOWLEDGEMENT

Authors greatly acknowledge Dr. G. Madhavan Nair, the chairman of ISRO and former director of Liquid Propulsion Systems Centre for having given constant encouragement for this work The help of Dr. P. Balachandran, Mathew George, S. Sreekumar, A. P. Baiju is also appreciated.

REFERENCES

1. Gordan S., Mcbride B.J., Computer Program for Calculation of Complex Chemical Equilibrium Compositions, Rocket Performance, Incident and Reflected Shocks and Chapman - Jouguet Detonations, NASA SP-273, Washington (1973)
2. Bartz, D.R., A Simple Equation for Rapid Estimation of Heat Transfer Coefficients in Nozzles, Jet Propulsion (1957) 27
3. Philip G. Hill, Carl R. Peterson, Mechanics and Thermodynamics of Propulsion, AIAA Progress in Astronautics and Aeronautics - 147, Addison-Wesley, New York (1992)