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Design and analysis of combustion chamber and nozzle of rocket engine by using Inconel 718

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***Abstract*—Rocket propulsion systems can be classified according to the type of energy source (chemical, nuclear, or solar), the basic function (booster stage, sustainer, attitude control, orbit station keeping, etc.), the type of vehicle (aircraft, missile, assisted take-off, space vehicle, etc.), size, type of propellant, type of construction, or number of rocket propulsion units used in a given vehicle. In this project, an attempt has been made to evaluate the design of a thrust chamber of a typical rocket engine. During the design phase, the detailed theoretical methods were ignored and the only the linear static analysis was carried out. In our project we have designed and analyzed the combustion chamber and the nozzle of a typical rocket engine by using the material of Inconel 718. The material (Inconel 718) has been selected with at most care by considering the availability, strength and cost. After giving the required boundary conditions, the model has been analyzed for the temperature and the stress at various points of the model.**

***Index Terms*—Rocket, propulsion, design, Inconel 718, stress.**

I. Introduction

Rocket propulsion and the mechanics involved in the design has undergone tremendous advancement in the recent years. Hampton et al. submitted a report that the Inconel 718 has high tensile properties and good microstructure and also fabricated with electron beam freeform fabrication (EBF). 718 alloy is Austenitic structure, precipitation hardening generate "γ" made it excellent mechanical performance. G rain boundary generate "δ" made it the best plasticity in the heat treatment.

Mukhtarov et al. found that the reduction in a grain size of Inconel 718 to 80 nm leads to the enhancement of strength up to 1920 MPa at room temperature and the decrease of ductility.

Lin et al. proved that Inconel 718 has compact material for rocket engine applications. Inconel 718 has the higher mechanical and physical properties which are required for the thrust chamber assembly.

The expertise of Ottobrunn's rocket propulsion team applied the concept in, design, analysis and performance prediction of liquid propellant rocket engines and combustion chambers. This experience has been gathered over a period of 40 years and has supported the well proven and successful design of rocket engines, rocket thrust chambers and associated systems covering a range of different propellants including monopropellants, earth storable hypergolic propellants, cryogenic propellants and more recently, alternative 'green' and hydrocarbon propellants.

Activities of the propulsion team

1. Design and optimization of rocket engine combustion chamber profile and nozzle cooling, Nozzle flow and performance analysis. Hydraulic characterization of cooling channels (flow checks). Combustion chamber simulation and associated heat transfer analysis. Production monitoring and assessment of design deviations. Upgrade development and evolution of analytical and prediction models, software and tools. Characterization of thrust chamber operating behavior. Development of multiphase flow codes within national and international technology programs under ESA/ESTEC contracts and R & D projects.
2. Rocket engine test analysis of injection head and ignition, combustion chamber efficiency and cooling. Design of injection component set-up and configuration. Hydraulic flow check characterization of elements. Hydraulic design of dome, distribution and manifolds. Design of ignition systems.
3. Service life analyses. Improvement of service life model (database). Characterization and analysis of crack propagation. Thrust chamber operating characterization (Operating envelope program).

Tools

A number of tools have been developed and are constantly being refined to support the design, analysis, and modeling and performance prediction of rocket engines. These include tools for rocket engine combustion flow analysis, computation of thrust chamber flow phenomena, determination of heat transfer processes in rocket engine thrust chambers, characterization of thrust chamber

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operational behavior and the determination of heat transfer processes in rocket engine thrust chambers. The experienced uses of these tools have repeatedly verified the accuracy of their predictions from the actual test and service life of rocket engines.

ROCFLAM: Rocket Combustion Flow Analysis Module

A Multiphase Naiver-Stokes code for the computation of flow phenomena in thrust chambers using storable propellant combinations such as hydrazine/N2O4 and MMH/N2O4. Developed under ESA/ESTEC Contract for the simulation of film-cooled engines.

CryoROC: Cryogenic Rocket Combustion

A multiphase Navier-Stokes code for the computation of flow phenomena in thrust chambers uses cryogenic oxygen and hydrogen. A derivative of the ROCFLAM code, linked with RFCS, the procedure allows the exact determination of heat transfer in regenerative cooled thrust chambers. The code also enables an assessment of the impact of transpiration cooling and combustion chamber cracks on the performance of rocket engines and the prediction of solid particle acceleration in kinetically compacted cold gas injection nozzles as used for the Vulcain 2 and Vinci rocket engines.

RCFS: Regenerative Coolant Flow Simulation

A program for the determination of heat transfer processes in rocket engine thrust chambers based on advanced Nusselt correlations. Used for hypergolic and cryogenic propellant combinations, the program is a partial module of CryoROC - for computation of heat transfer of the cooling agent, and partial module of Envelope - for the determination of heating-up. Used during the development of the Vulcain 2 and Vinci rocket engines

II. Materials and Methods

For evaluation of the design of combustion chamber and nozzle, the following general steady state heat transfer relations were used.

For heat transfer conduction the following general relation applies:

$$\frac{Q}{A} = -k\frac{dT}{dL} = -k\frac{\Delta T}{L} \qquad (1\text{-}1)$$

Where,

Q is the heat transfer per unit across a surface A,

dT/dL the temperature gradient with respect to thickness L at the surface A,

k the thermal conductivity expressed as the amount of heat transfer per unit time through a unit area of surface for 1° temperature difference as thickness. The negative sign indicates that temperature decreases as thickness increases.

The steady-state heat transfer through the chamber wall of a liquid-cooled rocket chamber can be treated as a series type, steady-state heat transfer problem with a large temperature gradient across the gaseous film on the inside of the chamber wall, a temperature drop across the wall, and, in cases of cooled chambers, a third temperature drop across the film of the moving cooling fluid. It is a combination of convection at the boundaries of the flowing fluids and conduction through the chamber walls. The problem is basically one of heat and mass transport associated with conduction through a wall.

The general steady-state heat transfer equation for regenerative cooled thrust chambers can be expressed as follows

$$q = h(T_0 - T_l) = \frac{Q}{A} \qquad (1\text{-}2)$$

$$= \frac{T_g - T_l}{1/h_g + t_w/k + 1/h_l} \qquad (1\text{-}3)$$

$$= h_g(T_0 - T_{wg}) \qquad (1\text{-}4)$$

$$= \left(k/t_w\right)(T_{wg} - T_{wl}) \qquad (1\text{-}5)$$

$$= h_l(T_{wl} - T_l) \qquad (1\text{-}6)$$

Where, q is heat transfer per unit area per unit time, W/m^2

T_g the absolute chamber gas temperature, K

T_l the absolute coolant liquid temperature, K

T_{wl} the absolute wall temperature on the liquid side of the wall, K

T_{wg} the absolute wall temperature on the gas side of the wall, K

h the overall film coefficient, W/m^2 K

h_g the gas film coefficient, W/m^2 K

h_l the coolant liquid film coefficient, W/m^2 K

t_w the thickness of the chamber wall, m

k the conductivity of the wall material. W/mK

Finite Element Method was used for mesh analysis and modelling.

Analytical methods provide quick and close form of the solutions, but they treat only simple geometries and capture only the idealized structural theory. Using the experimental techniques, representative or full-scale models can be tested. Experimental is costly, however, both in terms of the test facilities, the model instrumentation and the actual test. Relative to analytical methods, numerical methods require very few restrictive assumptions and can treat complex geometries. They are far more cost effective than experimental techniques. The current interest in the engineering community for development and application of computational tools based on numerical methods is thereby justified by the development of the most versatile numerical method, the finite element method (FEM).

Finite element analysis (FEA) is a computer simulation technique used in engineering analysis. It uses the Finite element method (FEM). In general, there are three phases in any computer-aided engineering task:

1) **Preprocessing** in which the analyst develops a finite element mesh to divide the subject geometry into sub-domains for mathematical analysis, and applies material properties and boundary conditions.
2) **Solution** during which the program derives the governing matrix equations from the model and

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solve for the primary quantities, and

3) **Post-processing** in which the analyst checks the validity of the solution, examines the values of primary quantities (such as displacements and stress), and derives and examines additional quantities (such as specialized stresses and error indicators).

Inconel 718 – Nickel-Chromium alloy

Alloy 718 is a precipitation hardenable nickel-based alloy designed to display exceptionally high yield, tensile and creep-rupture properties at temperatures up to 1300°F. The sluggish age-hardening response of alloy 718 permits annealing and welding without spontaneous hardening during heating and cooling.

This alloy has excellent weldability when compared to the nickel-base super alloys hardened by aluminum and titanium. This alloy has been used for jet engine and high-speed airframe parts such as wheels, buckets, spacers, and high temperature bolts and fasteners.

Estimation of thrust chamber dimensions

A regenerative cooled liquid-propellant rocket engine is to be designed for chamber pressure of 2.068 Mega Pascal, chamber temperature 3133.5K and thrust 1300 N.

Step-1

$$w_t = \frac{F}{I_{sp}} \qquad = \frac{1300}{238}$$

Total propellant flow rate , $w_t = 5.462 N/\sec$

$$w_f = \frac{w_t}{(r+1)} \qquad = \frac{5.462}{(1.25+1)}$$

Fuel propellant flow rate , $w_f = 2.427 N/\sec$

$$w_t = w_o + w_f$$

$$w_o = w_t - w_f \qquad = 5.462 - 2.427$$

Oxygen rate , $w_o = 3.025 N/\sec$

Step-2

$$T_t = T_c\left[\frac{1}{1+\frac{\gamma-1}{2}}\right] \qquad = 3133.15\left[\frac{1}{1+\frac{1.2-1}{2}}\right]$$

Throat temperature , $T_t = 2848.033K$

Step-3

$$T_e = T_c\left[\frac{P_e}{P_c}\right]^{\frac{\gamma-1}{\gamma}} \qquad = 3133.15\left[\frac{0.1035\times10^6}{2.068\times10^6}\right]^{\frac{1.2-1}{1.2}}$$

Exit temperature, $T_e = 1893.362K$

Step-4

$$P_t = P_c\left[1+\frac{\gamma-1}{2}\right]^{-\frac{\gamma}{\gamma-1}}$$

$$= 2.068\times10^6\left[1+\frac{1.2-1}{2}\right]^{-\frac{1.2}{1.2-1}}$$

Throat pressure , $P_t = 1.167 MPa$

Step-5

$$A_t = \frac{w_t}{P_t\sqrt{RT_t/\gamma g_c}}$$

$$= \frac{5.462}{1.167\times10^6\sqrt{(287\times2848.033)/(1.2\times9.81)}}$$

Throat area $A_t = 1.233\times10^{-3} m^2$

Step-6

$$A_t = \frac{\pi}{4}d_t^2 \quad , \quad d_t = \sqrt{\frac{A_t\times4}{\pi}} \quad , \quad = \sqrt{\frac{1.233\times10^{-3}\times4}{\pi}}$$

Throat diameter $d_t = 0.0396 m$

Step-7

$$M_e^2 = \frac{2}{\gamma-1}\left[\left(\frac{P_c}{P_{atm}}\right)^{\frac{\gamma-1}{\gamma}} - 1\right]$$

$$= \frac{2}{1.2-1}\left[\left(\frac{2.068\times10^6}{0.10135\times10^6}\right)^{\frac{1.2-1}{1.2}} - 1\right]$$

$$M_e^2 = 6.532 \qquad M_e = \sqrt{6.532}$$

Exit Mach number $M_e = 2.55$

Step-8

$$A_e = \frac{A_t}{M_e}\left[\frac{1+\frac{\gamma-1}{2}M_e^2}{\frac{\gamma+1}{2}}\right]^{\frac{\gamma+1}{2(\gamma-1)}} ,$$

$$= \frac{1.223\times10^{-3}}{2.55}\left[\frac{1+\frac{1.2-1}{2}2.55^2}{\frac{1.2+1}{2}}\right]^{\frac{1.2+1}{2(1.2-1)}}$$

Exit area, $A_e = 4.496\times10^{-3} m^2$

Step-9

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$$d_e = \sqrt{\frac{A_e \times 4}{\pi}}\,, \quad = \sqrt{\frac{4.496\times10^{-3}\times 4}{\pi}}$$

Exit diameter , $d_e = 0.0756m$

Step-10

$$\frac{A_e}{A_t} = \frac{4.496\times10^{-3}}{1.233\times10^{-3}}$$

Area ratio , $\frac{A_e}{A_t} = 3.646$

Step-11

$$L_n = \frac{(r_e - r_t)}{\tan\theta}\,, \quad = \frac{(0.0378 - 0.0198)}{\tan 15}$$

Length of the nozzle, $L_n = 0.06717m$

Step-12

For this propellant combination we will assume a combustion chamber L* of 2.032m.

$$V_c = L^* A_t\,, \quad = 0.035\times1.233\times10^{-3}$$

Volume of the chamber, $V_c = 4.315\times10^{-4} m^3$

Step-13

Chamber length

$$V_c = (1.1)(A_c L_c)$$

However, we must find determine the chamber area. We do this by assuming that the chamber diameter is two times the nozzle throat diameter, therefore

$$d_c = 2d_t\,, \quad = 2\times0.0396$$

Chamber diameter, $d_c = 0.0792m$

Step-14

$$A_c = \frac{\pi}{4} d_c^2\,, \quad = \frac{\pi}{4} 0.0792^2$$

Chamber area, $A_c = 4.926\times10^{-3} m^2$

Step-15

$$L_c = \frac{V_c}{1.1\times A_c}\,, \quad = \frac{4.315\times10^{-4}}{1.1\times4.926\times10^{-3}}$$

Chamber length, $L_c = 0.0796m$

Step-16

$$C_t = \sqrt{\frac{2\gamma}{\gamma+1} R T_c}\,, \quad = \sqrt{\frac{2\times1.2}{1.2+1}\times287\times3133.15}$$

Throat velocity, $C_t = 990.021 m/\sec$

Step-17

$$C_e = \sqrt{\frac{2\gamma}{\gamma-1} R T_c \left[1 - \left(\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}}\right]}$$

$$, = \sqrt{\frac{2\times1.2}{1.2-1}\times287\times3133.15\left[1-\left(\frac{0.101325\times10^6}{2.068\times10^6}\right)^{\frac{1.2-1}{1.2}}\right]}$$

Exit velocity, $C_e = 2064.473 m/\sec$

Design parameters

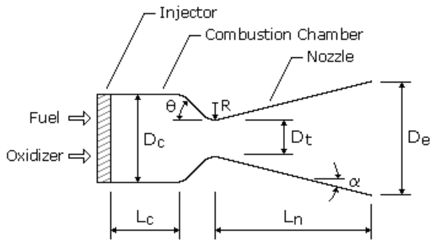

Figure 1 Design Parameters of Nozzle

Chamber temperature	= 3133K
Chamber pressure	= 2.068 Mpa
Chamber diameter	= 0.07892m
Chamber length	= 0.796m
Throat temperature	= 2848.033K
Throat pressure	= 1.167Mpa
Throat diameter	= 0.0396m
Exit temperature	= 1893.362K
Exit pressure	= 0.101325 Mpa
Exit diameter	= 0.0756 m

III. RESULTS AND DISCUSSION

The nozzle and the combustion chamber and they were designed in ANSYS software. The dimensions were obtained from a sample calculation and they were used to the dimensional constraints.

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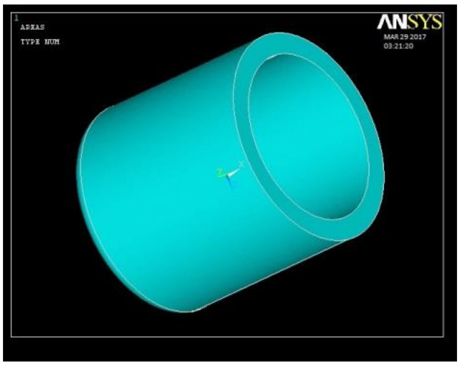

Figure 2. Modeling View of Combustion Chamber

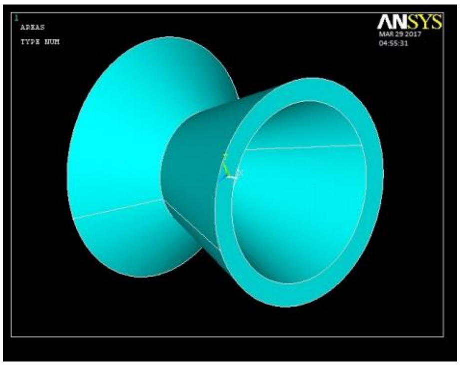

Figure 3 Modeling View of Nozzle

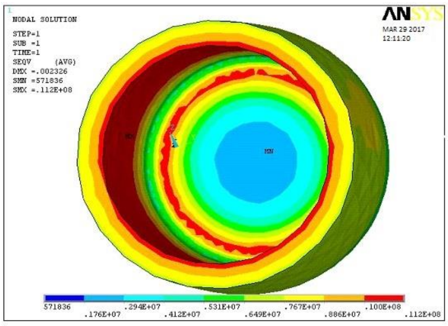

Figure 4 Von Misses Stress Contour in the Combustion Chamber (Front View)

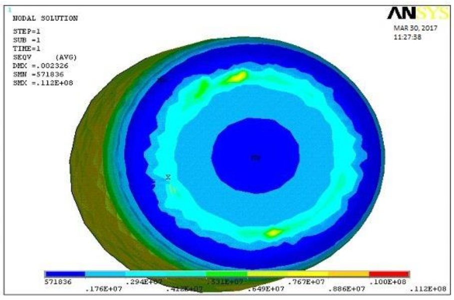

Figure 5 Von Misses Stress Contour In The Combustion Chamber (Back View)

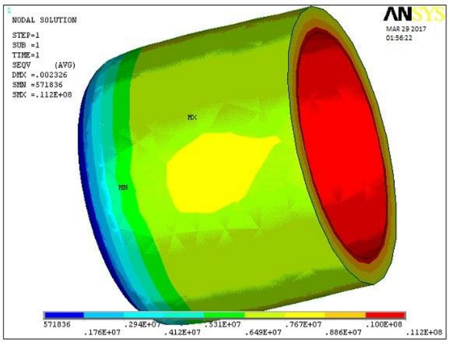

Figure 6 Von Misses Stress Contour In The Combustion Chamber (Side View)

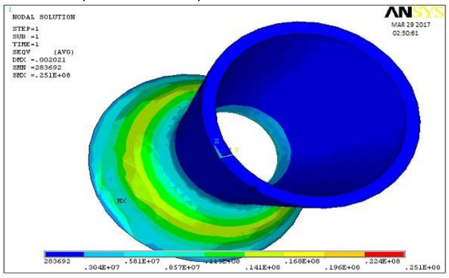

Figure 7 Von Misses Stress Contour In The Nozzle (Back View)

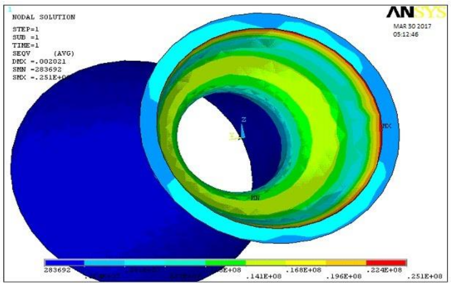

Figure 8 Von Misses Stress Contour In The Nozzle (Front View)

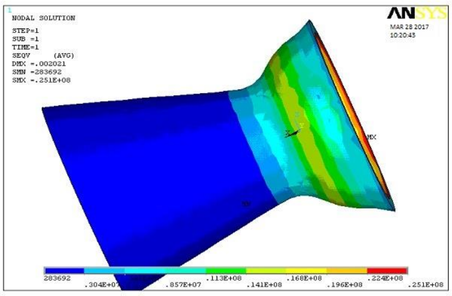

Figure 9 Von Misses Stress Contour in the Nozzle (Side View)

From all the above stress contours of the nozzle and combustion chamber the maximum stress is 0.025Gpa and 0.0112Gpa respectively. Then the material yield stress is1.034Gpa. So the design is found satisfactory

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IV. Conclusion

In this paper, the combustion chamber and the nozzle of a typical rocket engine were designed and analyzed by using the material of Inconel 718. The material (Inconel 718) has been selected with at most care by considering the availability, strength and cost.

The combustion chamber and the nozzle were designed separately in ANSYS Software by the design parameters.

The model was analyzed for the temperature and the stress at various points of the model in ANSYS.

Throughout the model the stress is well below the yield limit of the material selected and the results were found satisfactory

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