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Assessment of a Rocket Combustion Chamber Model: A Comparative Study of Titanium and Stainless-Steel Material Performance

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Abstract
The thermo-structural simulation of the rocket combustor model was performed with ANSYS software considering two different materials: cast stainless steel and titanium alloy to evaluate the variations in the model parameters. The initial and final operating temperatures of the combustor were 22^{o}c and 1500^{o}c at (5bar) respectively. The finite element analysis reveals that a mesh comprising of 36250 nodes and 105510 elements for the entire volume of the combustor was generated. The minimum and maximum temperature for the stainless steel and titanium alloys ranges between 0^{o}c – 1500^{o}c and 29.94^{o}c – 1500^{o}c, respectively. A percentage dissipation of 40% and 60% for stainless steel and titanium alloys were observed, indicating titanium alloy to have higher heat dissipation quality. While the minimum and maximum stress and strain for stainless steel varies between -14.38MPa – 22.55MPa and 6.54 x 10^{-6}mm – 2.10 x 10^{-4}mm, for titanium alloys the stress and strain values range between 63.38MPa – 74.10MPa and 72.20 x 10^{-5}mm – 77.46 x 10^{-3}mm, respectively. The stress and strain analysis were observed for both materials at maximum thermal stress of the combustor at 1500^{o}c, where the maximum values are 586MPa and 1070MPa for stainless steel and titanium alloys. Finally, the maximum deflection observed were 2.549x10^{-3}mm and 1.5944mm for stainless steel and titanium alloys, respectively. Hence, stainless steel offers better strength, and lower heat dissipation than titanium alloys.

Keywords: *Temperature, Rocket, Combustion, Chamber, Model, Total deformation*

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1. Introduction

Rocket combustion chamber is the part of rocket engine where the combustion of the solid or liquid propellants takes place to produce thrust required to propel the rocket. The high heat energy released from the combustor exceed by far a typical value of that of a nuclear power plant (George and Oscar, 2017). Rocket propulsion is needed for both primary propulsions such as acceleration along the flight path which can be in the form of ascent, orbit insertion, or orbit change manoeuvres and also for secondary propulsion such as attitude control, spin control, momentum wheel and gyro unloading, rendezvous in space, stage separation, and for the settling of liquids in tanks (Amirhossein *et al.*, 2017). Rocket combustion chamber consists of four main components: a charge of propellant which when ignited develop the propulsive force, a hollow tube or chamber within which the propellant is burned, an igniter which initiates the start of combustion of the propellants, and finally a nozzle or outlet through which products of propellant combustion are exhausted. The combustion chamber of a rocket acts as a casing that houses the igniter, propellant, copper wire, and the nozzle. The combustor can be constructed from a range of different materials such as: cardboard material for construction of a small black powder model rocket motors, while aluminium may be used for larger composite fuel motors. Titanium or steel on the other hand may be applied for space shuttle boosters whereas Filament-wound graphite epoxy casings are useful for high performance rocket motors. Thus, the combustion chamber casing construction is designed such that it can withstand the high pressure, temperature and stresses resulting from the combustion of solid propellant (Martin, 2009). Material selection is a vital aspect of any engineering design. The combustion chamber and nozzle walls have to be designed to withstand relatively high temperature, high gas velocity, chemical erosion, and high stress. The wall material must be capable of high heat transfer rates (which means good thermal conductivity), and at the same time have adequate strength to withstand the chamber combustion pressure (Shearer and Gregory, 1996). Hence, this article assessed titanium and stainless-steel alloy material as a rocket combustion chamber designed materials.

2. Materials and methods

2.1 Materials

In the design, simulation, and evaluation of the combustion chamber central to this research, two materials were selected for analysis: (i) Titanium alloy and (ii) Cast stainless steel. In ANSYS software, materials data can be obtained through several methods, for this research it was typically through ANSYS material library: ANSYS provides a built-in material library containing predefined properties for a wide range of materials. Users can access this library to select materials and their corresponding data, such as density, thermal conductivity, and mechanical properties. The properties of these metals are shown in Tables 1 and 2, respectively.

Table 1: Properties of stainless steel

Properties of Outline Row 11: Stainless Steel

	A	B	C
1	Property	Value	Unit
2	Density	7750	kg m^-3
3	Isotropic Secant Coefficient of Thermal Expansion		
4	Coefficient of Thermal Expansion	1.7E-05	C^-1
5	Reference Temperature	22	C
6	Isotropic Elasticity		
7	Derive from	Young's Modulu...	
8	Young's Modulus	1.93E+11	Pa
9	Poisson's Ratio	0.31	
10	Bulk Modulus	1.693E+11	Pa
11	Shear Modulus	7.3664E+10	Pa
12	Tensile Yield Strength	2.07E+08	Pa
13	Compressive Yield Strength	2.07E+08	Pa
14	Tensile Ultimate Strength	5.86E+08	Pa
15	Compressive Ultimate Strength	0	Pa
16	Isotropic Thermal Conductivity	15.1	W m^-1 C^-1
17	Specific Heat	480	J kg^-1 C^-1
18	Isotropic Relative Permeability	1	
19	Isotropic Resistivity	7.7E-07	ohm m

Table 2: Properties of titanium alloy

Properties of Outline Row 13: Titanium Alloy

	A	B	C
1	Property	Value	Unit
2	Density	4620	kg m^-3
3	Isotropic Secant Coefficient of Thermal Expansion		
4	Coefficient of Thermal Expansion	9.4E-06	C^-1
5	Reference Temperature	22	C
6	Isotropic Elasticity		
7	Derive from	Young's Modulu...	
8	Young's Modulus	9.6E+10	Pa
9	Poisson's Ratio	0.36	
10	Bulk Modulus	1.1429E+11	Pa
11	Shear Modulus	3.5294E+10	Pa
12	Tensile Yield Strength	9.3E+08	Pa
13	Compressive Yield Strength	9.3E+08	Pa
14	Tensile Ultimate Strength	1.07E+09	Pa
15	Compressive Ultimate Strength	0	Pa
16	Isotropic Thermal Conductivity	21.9	W m^-1 C^-1
17	Specific Heat	522	J kg^-1 C^-1
18	Isotropic Relative Permeability	1	
19	Isotropic Resistivity	1.7E-06	ohm m

2.2 Methods

The design, simulation and assessment of solid rocket combustor using engineering simulation software takes into account governing equations such as conservation of mass equation (continuity equation), conservation of momentum (Navier-Stokes Equation), and conservation of energy (energy equation). Navier Stokes equations are always solved together with continuity equation but for flows that involve heat transfer or compressibility effect, energy equation is added to the solution. The steady flow energy equation can be used to relate the nozzle exhaust condition of the rocket motor to the conditions in the combustion chamber and the exit pressure. These equations are at the heart of fluid flow modelling. Solving them, for a particular set of boundary conditions.

In design of rocket combustion chamber, some of the major considerations are geometry configuration of the rocket combustion chamber, finite element analysis, heat transfer analysis, weight, material selection and cost. Heat transfer occurs ones any system undergoes a change of process from its equilibrium position. Heat transfer has direction as well as magnitude. The rate of heat conduction in a specified direction is proportional to the temperature gradient, which is the rate of change in temperature with distance in that direction. Heat conduction in a medium, in general, it is three-dimensional and time dependent, and the temperature in a medium varies with position as well as time, that is $T = T(x, y, z, t)$. Heat conduction in a medium is said to be steady when the temperature does not vary with time, and unsteady or transient when it does (Yunus and Afshin (2015).

In analyzing the heat transfer of the solid combustor, a steady one-dimensional heat transfer analysis is considered for a cylindrical object. The heat transfer rate based on Fourier's law of heat conduction over a cylinder is expressed as:

$$\dot{Q} = -KA\frac{dT}{dr} \quad (1a)$$

where $\dot{Q}$ is the heat transfer rate, K is the thermal conductivity of the material, A is the heat transfer area, $\frac{dT}{dr}$ is the temperature gradient over the changing radius of the cylinder.

In order to calculate the geometric parameters related to the design of the combustion chamber and the nozzle (volume of the combustion chamber, the flow area at the nozzle exit and the throat), different desired engine specifications such as thrust, specific impulse and characteristic length were combined with different pressure and temperature combinations resulting from propellant combustion. Therefore, the design of rockets from conception to its launching has gained tremendous success and acceptance and as well has become a familiar spectacle. Observers following major space companies like SpaceX, Virgin Galactic, and Blue Origin on social media have likely seen the large quantities of gases released during rocket lift-off from the launch pad. As a result, a perceptive observer may have recognized the principle of propulsion, which connects reaction force to the ejection of mass. This principle of propulsion expressed as an equation is given by the relation:

$$F = q.V_e \quad (1b)$$

where F is the reaction force called thrust, q is the gas mass flow rate and V_e is the exhaust velocity of the gas. It may be assumed that propellant burn in parallel layer, and that the burn rate is only a function of the pressure (Alian 1993). Under these conditions, the flow rate resulting from the combustion at a given time is:

$$q = \rho.S.v \quad (2)$$

Where ρ is the density of the propellant, S is the burning surface and v is the burning rate of the propellant. The flow rate of the gas (q') passing through the nozzle is given as (Alian 1993):

$$q' = C_D.P.A_t \quad (3)$$

The operating point of the rocket motor corresponds to the equality of the gas flow rates which are created from the combustion of the propellant grain and ejected by the nozzle (Alian, 1993) as shown below.

$$\rho.S.v = C_D.P.A_t \quad (4)$$

From the relationship above at any given time in the combustion chamber of a rocket; A_t and S having values specific to the rocket motor containing a known propellant which defines C_D and ρ, the burning rate v is proportional to the pressure P given by the expression below (Alian, 1993).

$$v = \frac{C_D}{\rho} . \frac{A_t}{S} . P \tag{5}$$

The burning rate is thereby defined by the expression.

$$v = aP^n \ (n < 1) \tag{6}$$

Assuming that the gases are ideal, it can be show that the propellant discharge coefficient is affected only by the nature and temperature of the gas flowing through the nozzle as show in Equation (7)

$$C_D = \frac{\Gamma(\gamma)}{\sqrt{\gamma r T}}; \ \Gamma(\gamma) = \gamma . \left(\frac{2}{\gamma+1}\right)^{\gamma+1/_{2(\gamma-1)}} \tag{7}$$

where P is the combustion pressure at a given time, A_t, is nozzle throat area, C_D, is the propellant discharge coefficient, T is the combustion temperature, γ is the ratio of specific heat of combustion gases at constant pressure and volume, and $r = \frac{R}{M}$ is the ratio of the universal gas constant (8.134J/kg. K) to the molar weight in kg (approximately 29×10^{-3}kg for the propellant).

Equations (1) to (7) play a vital role in simplified design of rocket motors and so achieve this method, an equipment with the following detail is used:

i. A computer system as rated below,

Windows Edition

Windows10 Pro.© 2019 Microsoft Corporation. All right reserved.

System Processor: Intel(R) Core (TM) i5-8250U CPU @ 1.60GHz1.80GHz

Installed Memory (RAM): 8.00GB (7.43GB usable)

System Type: 64-bit Operating System, x64-based processor

Pen and Touch: No pen and touch input is available from this display

Computer name, domain and workgroup settings

Computer Name: DESTOP-V6ACV0N

Workgroup: WORKGROUP

Windows activation

Product ID: 00331-10000-00001-AA223

ii. An ANSYS software R15.0 version

iii. Rocket combustion chamber design guides from National aeronautics space agency NASA

3. Results and discussion

3.1 Combustion chamber model

The rocket combustion chamber model was designed using SolidWorks software, the design comprises three components such as heat shield, combustor cylinder and exit burner as shown the Fig. 1 to 4. The models in Fig. 1 to 3 are assembled into one component as shown in Fig. 4. The analysis was run on ANSYS software package (ANSYS structural and ANSYS steady state thermal). The geometry of the combustor was created using SolidWorks and was imported into the geometry workbench of ANSYS for analysis. The dimensions and the boundary conditions of the model were obtained including material selection. The next task that was carried out was meshing the geometry created known as FEA. The created geometry was imported to the meshing workbench. The mesh used is quadrilateral mesh. The mesh was then resized by changing from coarse to fine mesh at the mesh interface to generate a suitable and an acceptable range of element and nodes as required by FEA standard.

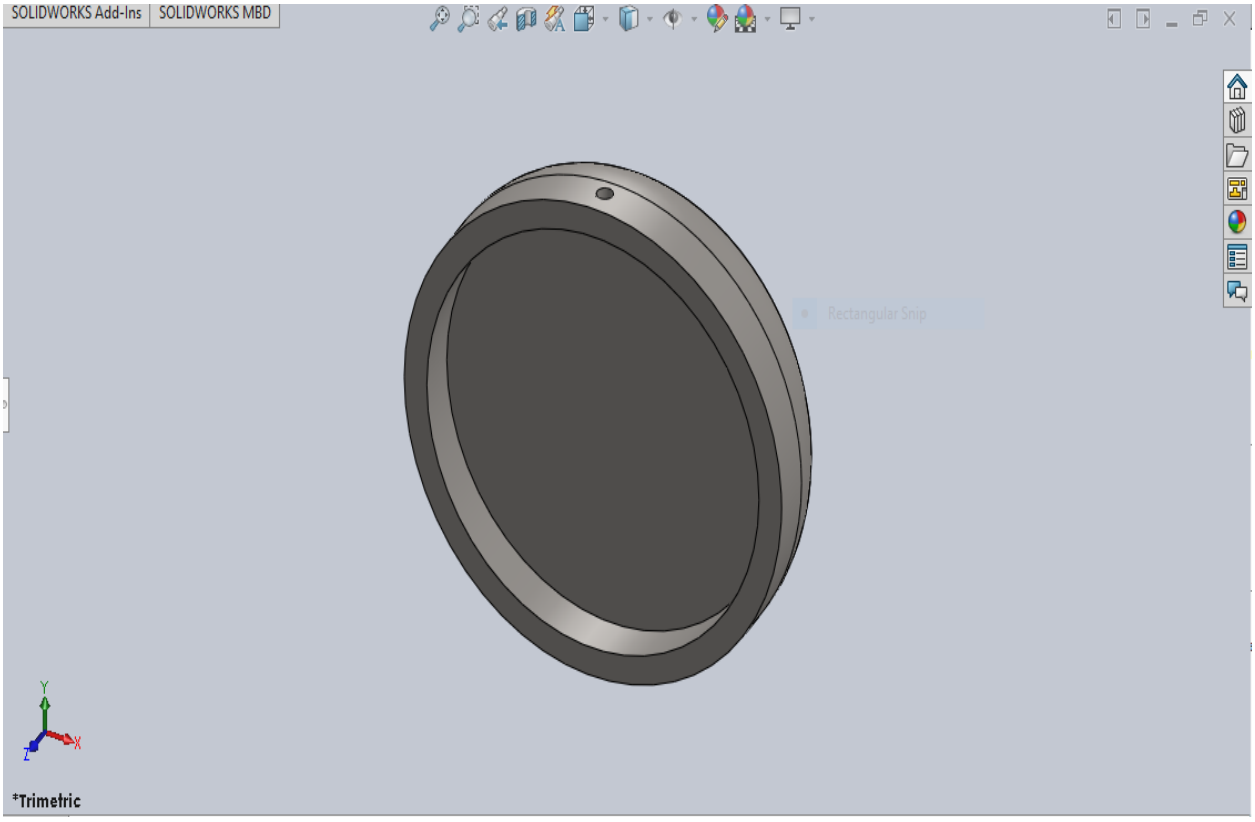

Fig. 1: Heat shield

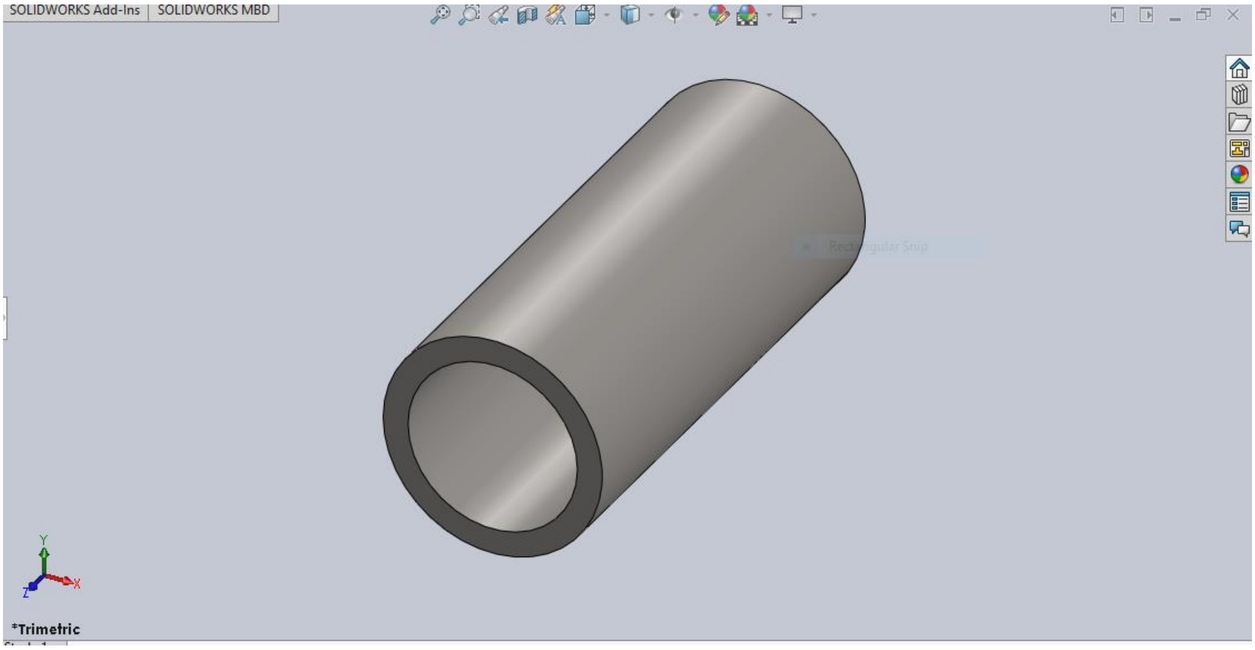

Fig. 2: Combustor cylinder

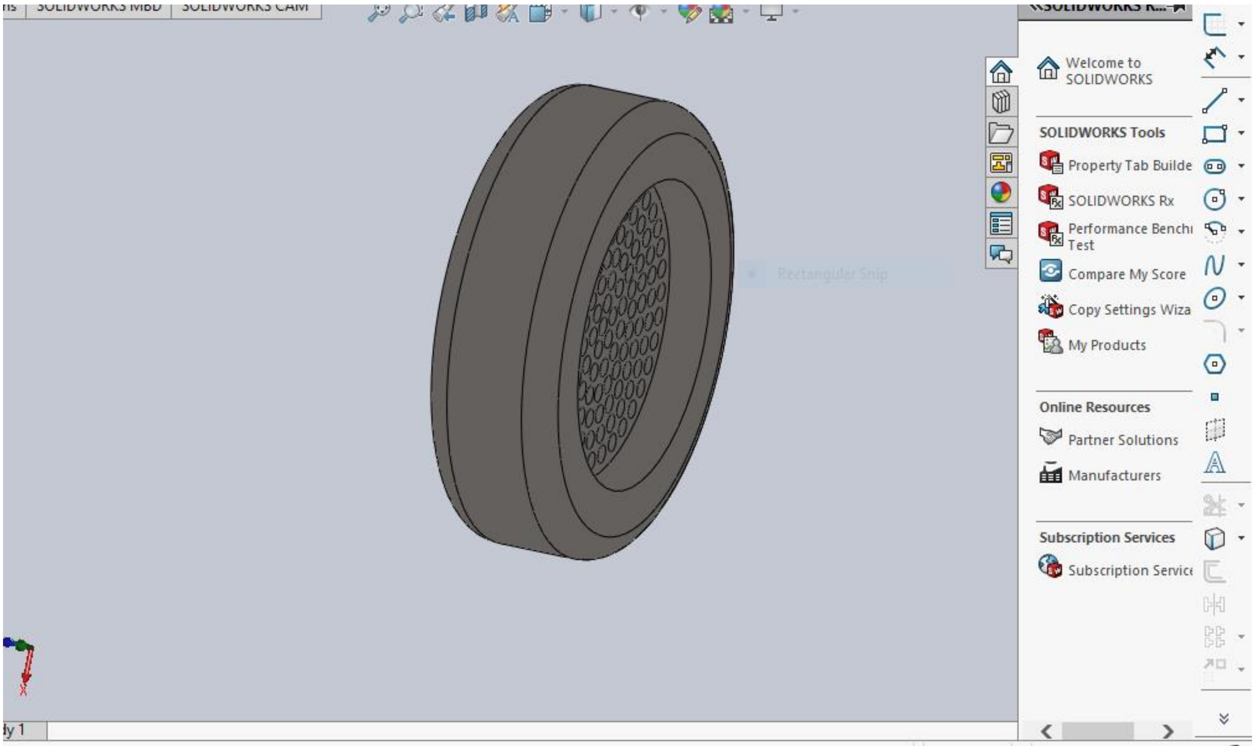

Fig. 3: Exit burner

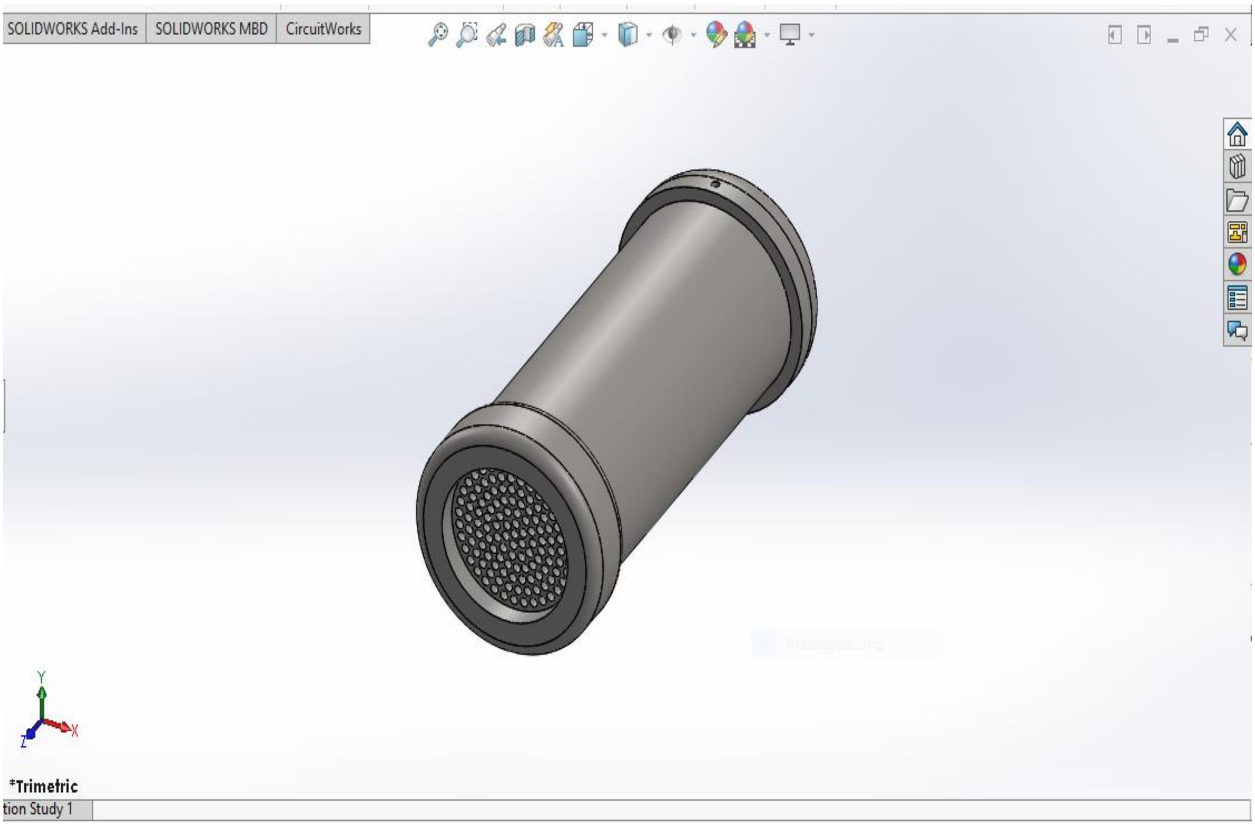

Fig. 4: Assembled combustor

3.2 Finite element analysis (mesh)

The following results were obtained from the simulation in ANSYS steady state thermal at same boundary conditions. Two cases of materials such as stainless-steel alloy and titanium alloy were considered and used to evaluate the model parameter variation. The initial ambient temperature of 22ºc was set and the combustion chamber was fired at temperature of 1500ºc with equivalent pressure of 5bar. The following results in Fig. 5 to 15 were obtained through simulation of the modelled combustor.

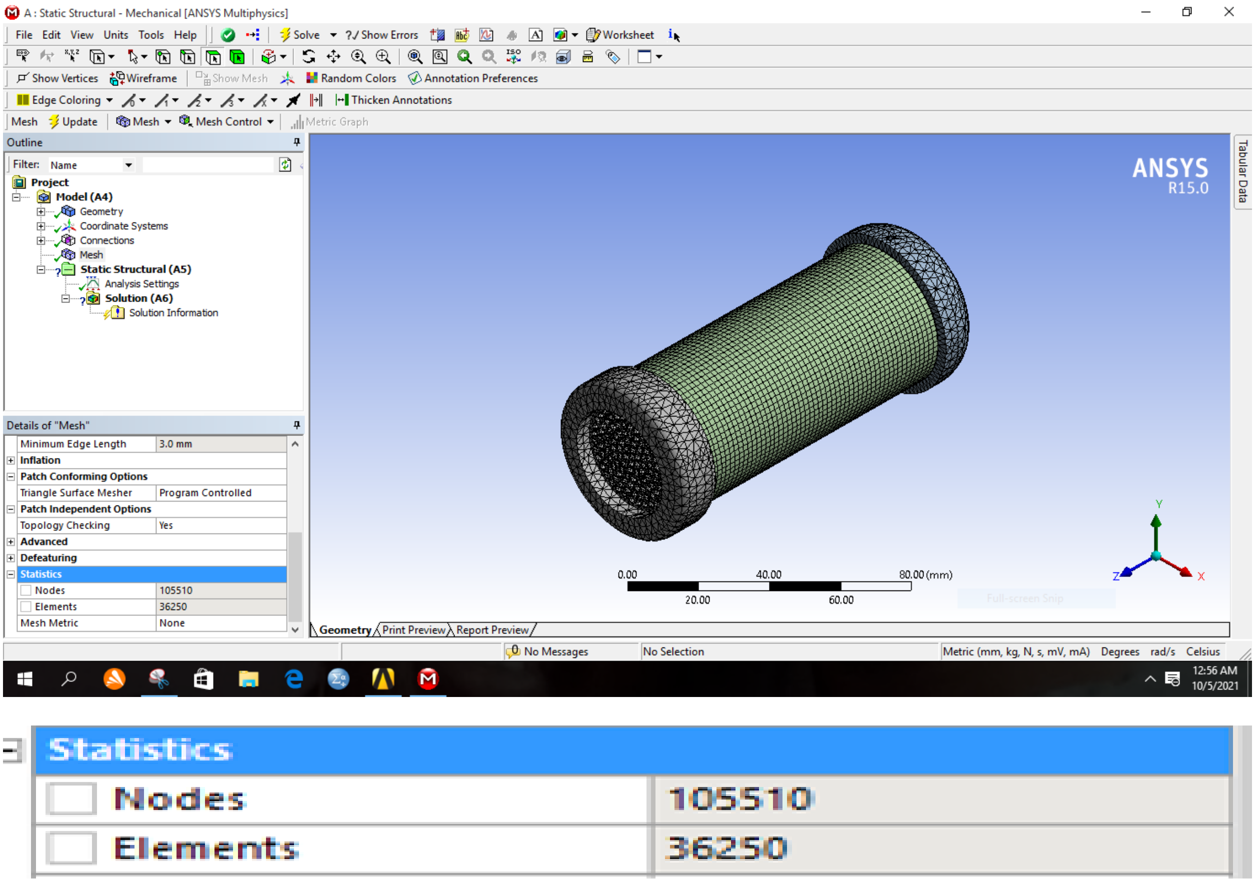

Fig. 6: Finite element analysis (FEA) of the model

3.3 Material simulation

Fig. 7 to 9 show the results obtained from material simulation of the modelled combustor. The Fig. 7 to 8 respectively show the variations of temperature across the combustion chamber of stainless steel and titanium alloy material as considered. The outer wall of the combustor was maintained at ANSYS default temperature of 22ºC. The internal wall of combustion chamber was set to a temperature of 1500ºC. The analysis as shown by the colour contours reveal that the steel temperature varies from the range of 0ºc min to 1500ºc max while that of titanium alloy ranges from 29.937ºC min to 1500ºC max. The percentage of combustion heat dissipation by stainless steel and titanium as further illustrated by Fig. 9 are 40% and 60% respectively. This means that titanium alloy compared to stainless steel has better (up to a fraction of 0.2) heat dissipation and when used in fabricating the combustion chamber of a rocket requires more cladding than stainless steel.

Fig. 10 to 11 respectively show the variations of stress across the combustion chamber of stainless steel and titanium alloy material. The analysis as shown by the colour contours reviewed that the steel stress varies from the range of -14.793MPa to 27.551MPa max while that of titanium alloy ranges from 63.379MPa min to 74.102MPa max. It can be observed that for both materials the maximum thermal stresses experienced in the combustion chamber at 1500ºC are less than the ultimate strength values for stainless steel and titanium alloy which are 586MPa and 1070MPa, respectively. Hence, both materials have high ultimate strength suitable for fabrication of rocket combustion chamber. However, Titanium appears better when the various stress differences are compared (max stress – min stress) which is 36.9293MPa for steel and 10.723MPa for titanium alloy with respect to the combustion chamber wall thickness. An increase in stress resistance value results to the need to increase the combustion chamber wall thickness or chamber cladding thickness.

Fig. 12 to 13 respectively show the variations of strain across the combustion chamber of stainless steel and titanium alloy material. The analysis as shown by the colour contours reveals that the steel strain varies from the range of 6.5444E-6mm to 2.1033E-4mm max while that of titanium alloy ranges from 72.199E-5mm min to 77.463E-3mm max. It can be observed that titanium absorbs higher heat value compared to the stainless steel and hence might require more heat liner when used in chamber fabrication. The high overall temperature levels cause materials to have high strain deformation which can quickly lead to overheating and possibly damage if the cooling system fails.

Fig. 14 to 15 respectively show the variations of deflection across the combustion chamber of stainless steel and titanium alloy materials. The analysis as shown by the colour contours reveal that the steel total deflection varies from the range of 0mm to 2.549E-3mm max while that of titanium alloy ranges from 0mm min to 1.594mm max. It can be observed that titanium absorbs high heat which made it to deflect more when compared to the stainless steel and hence might require more heat carbonization and treatment when used in chamber fabrication.

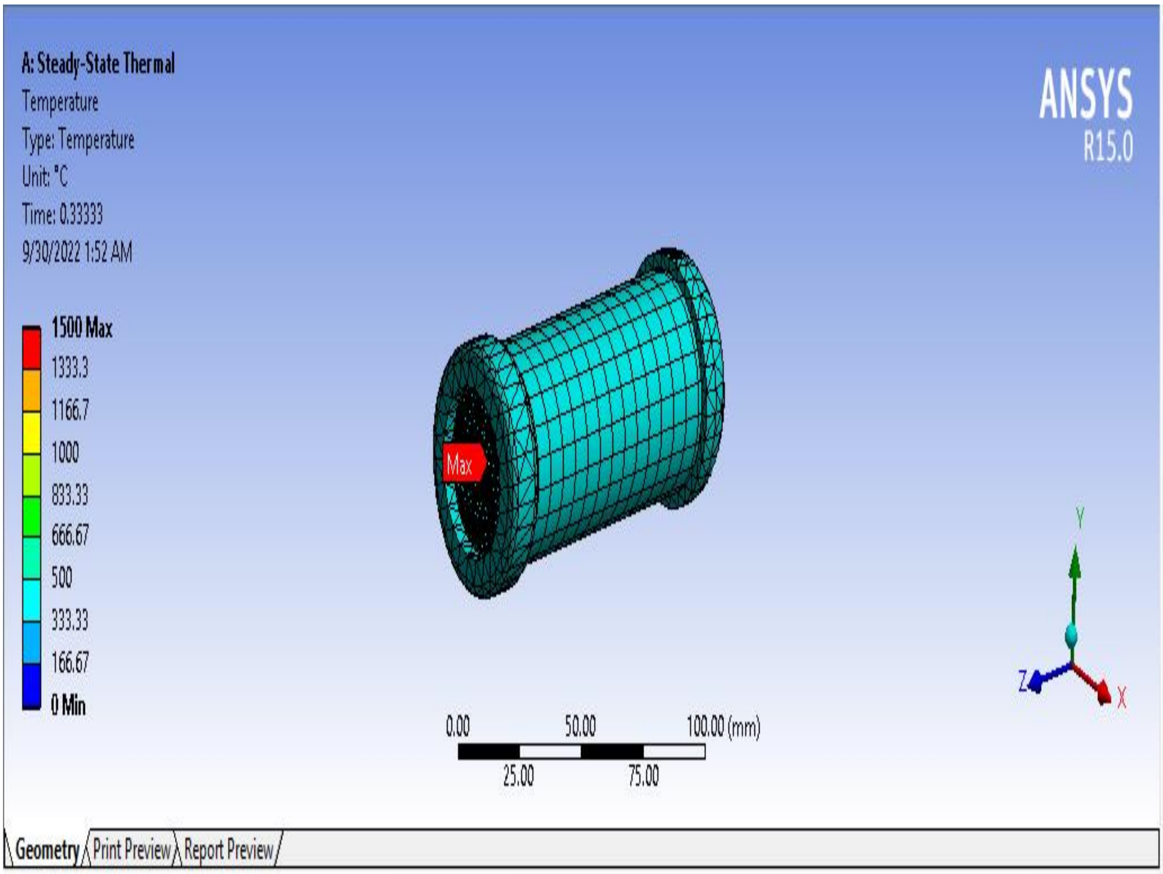

Fig. 7: Temperature variation for stainless steel alloy

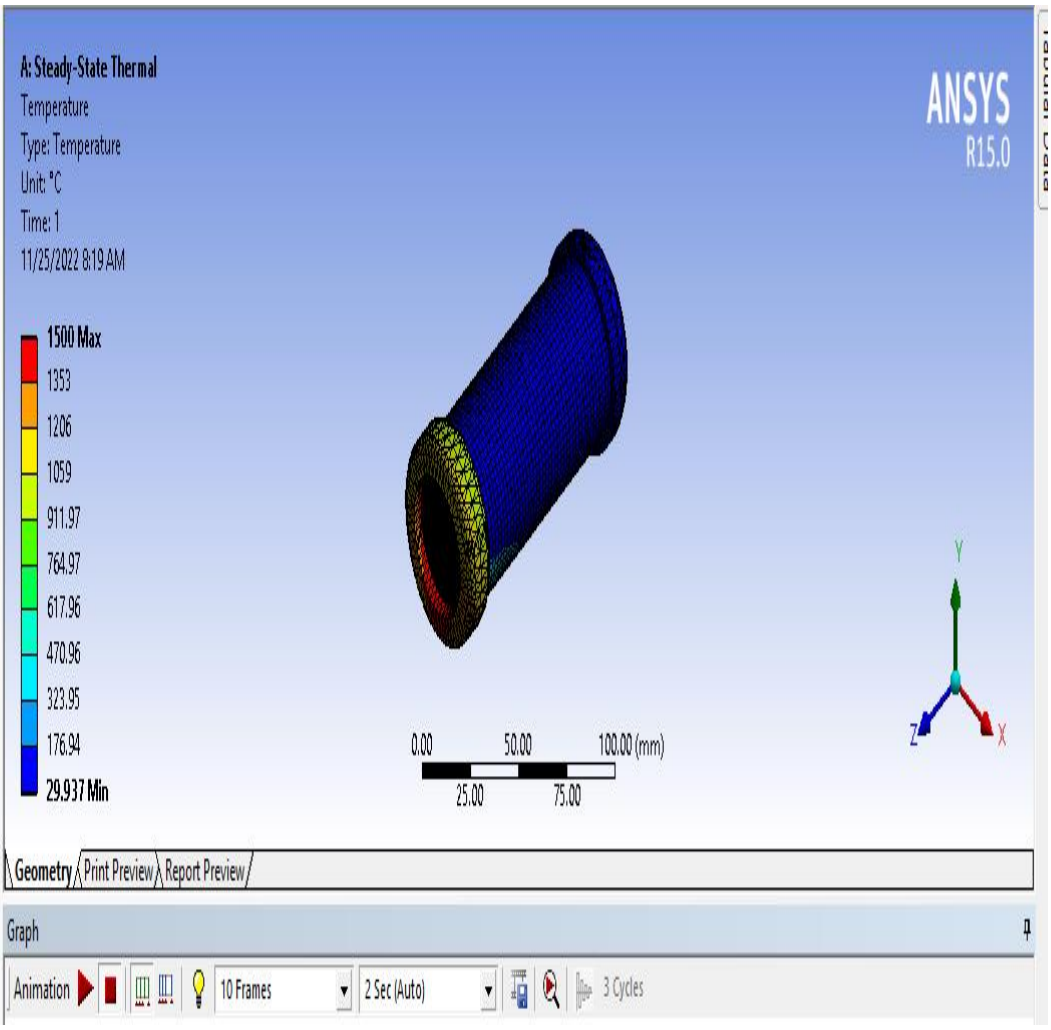

Fig. 8: Temperature variation for titanium alloy

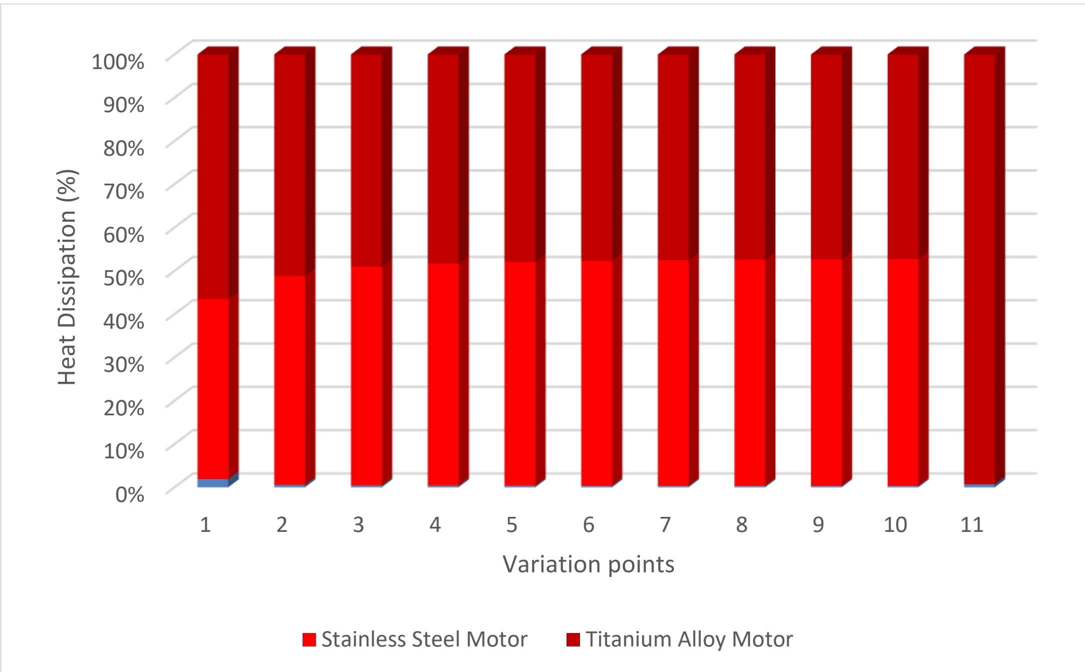

Fig. 9: Percentage temperature variation comparison

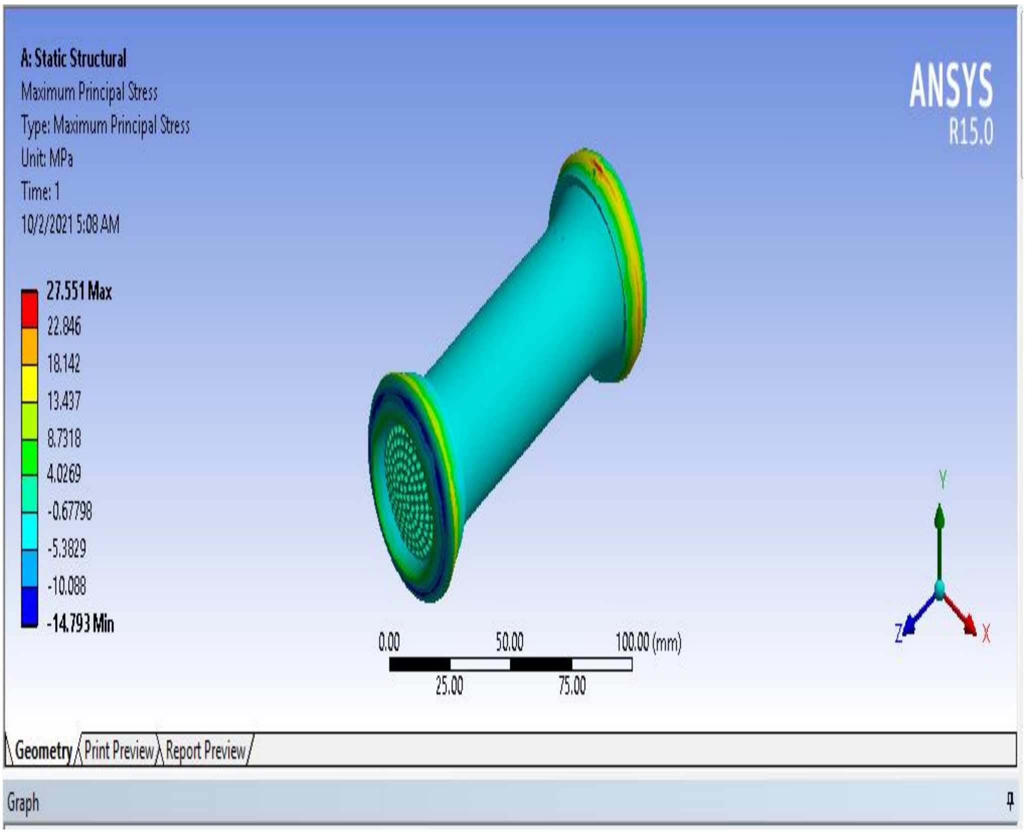

Fig. 10: Stress variation for stainless steel

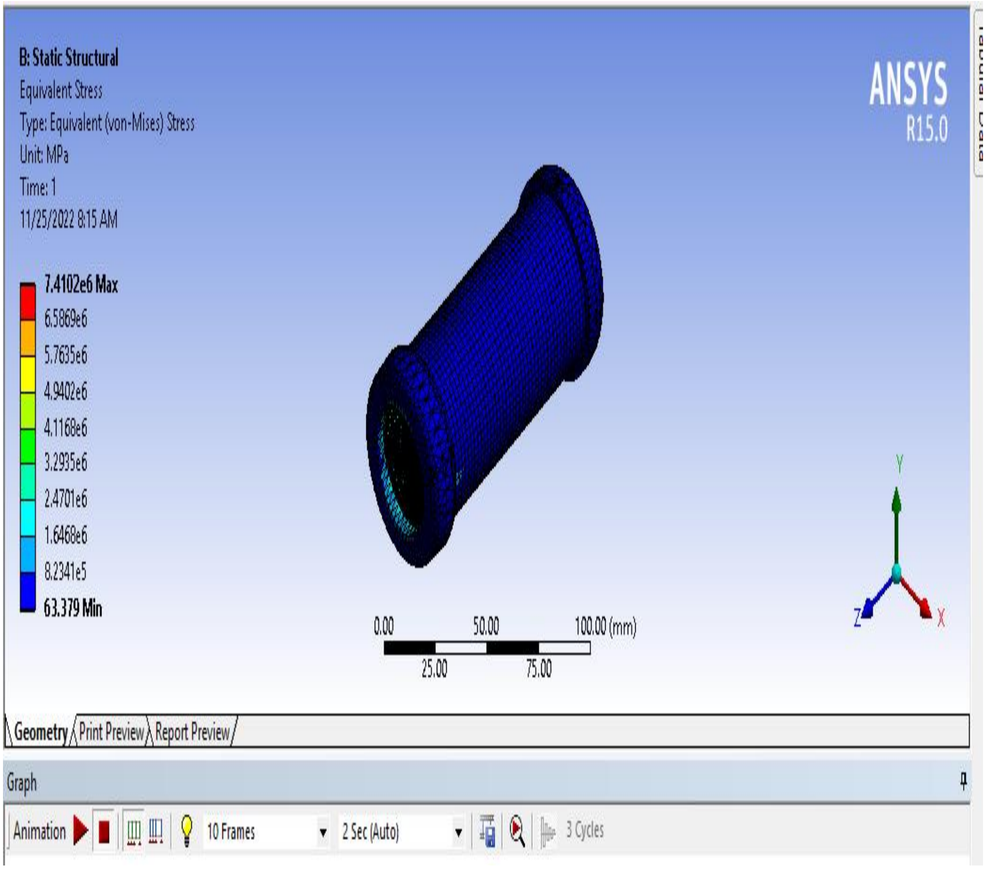

Fig. 11: Stress variation for titanium alloy

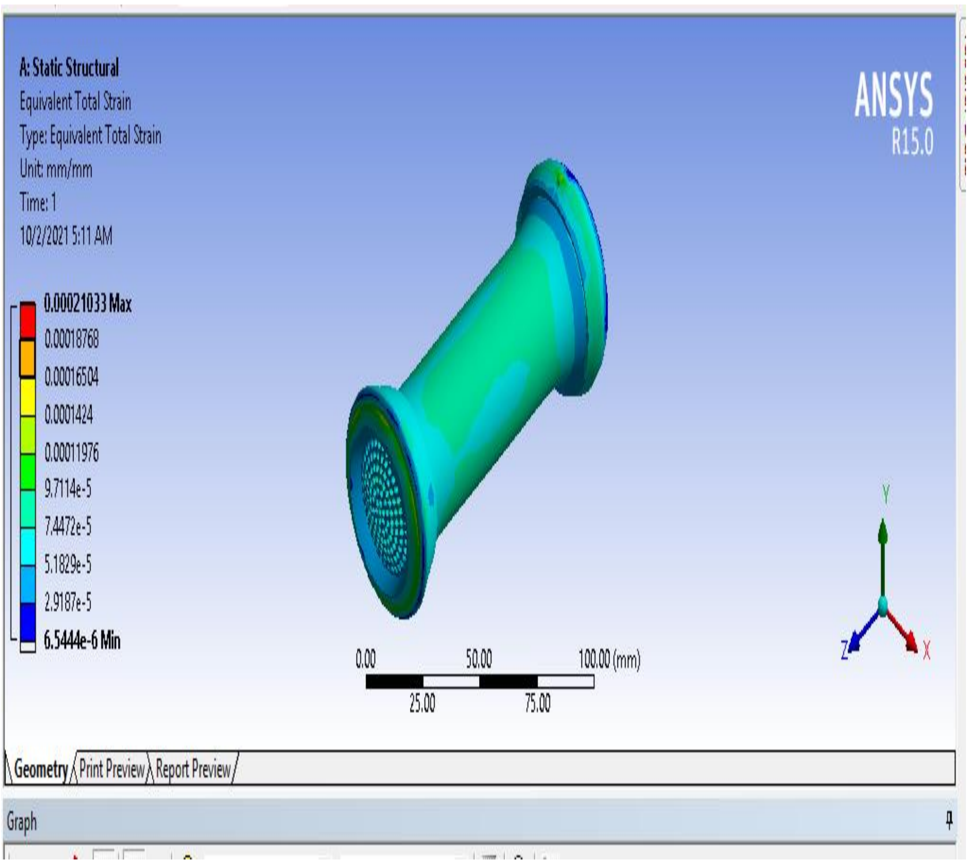

Fig. 12: Strain variation for stainless steel

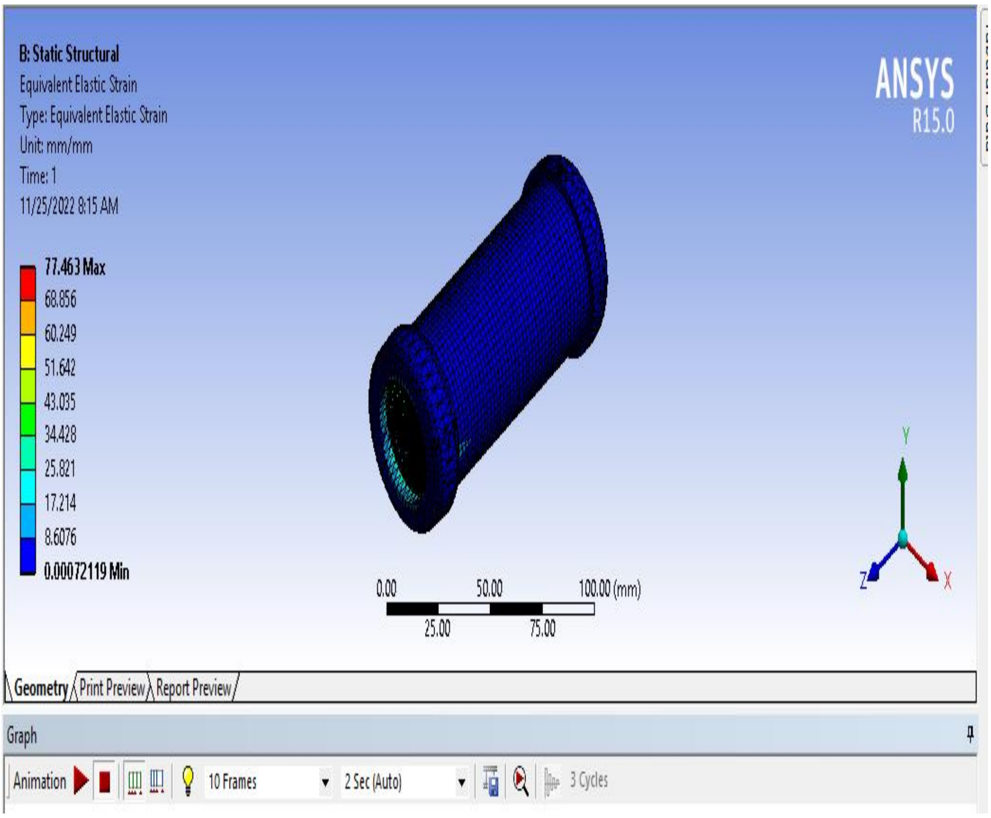

Fig. 13: Strain variation for titanium alloy

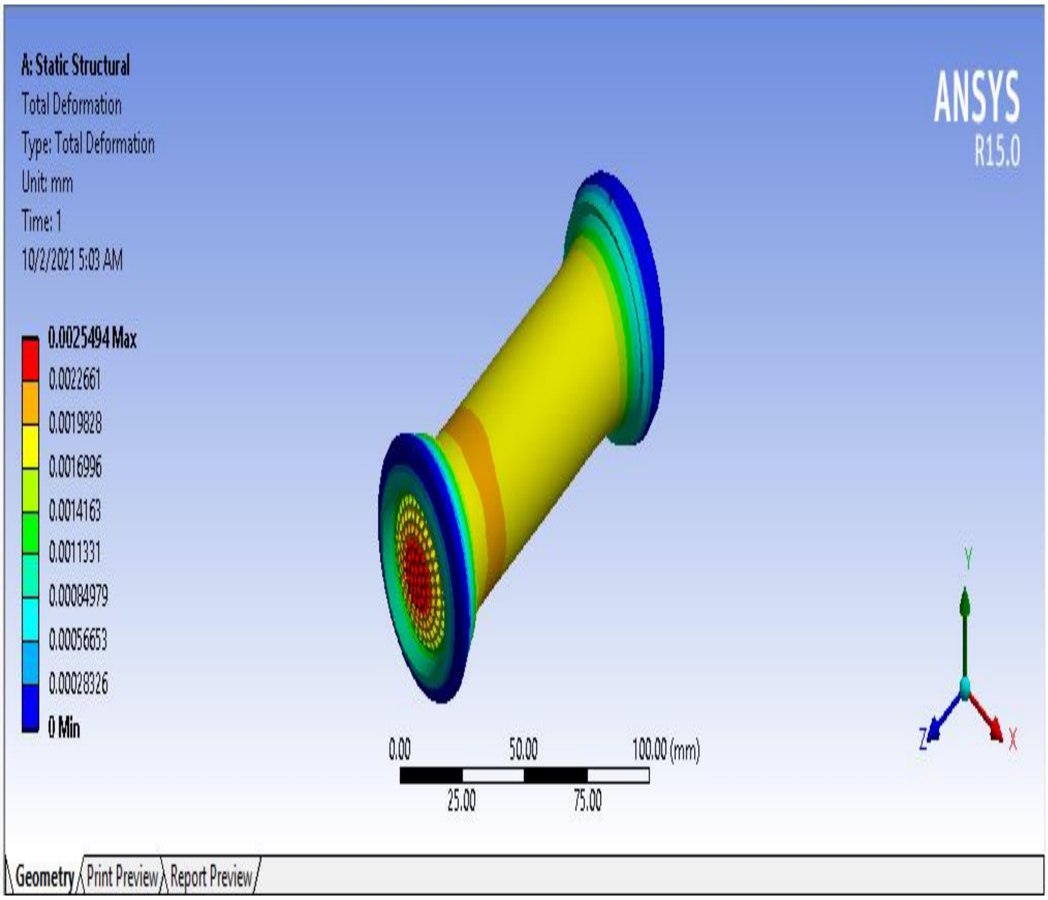

Fig. 14: Total deformation variation for stainless steel

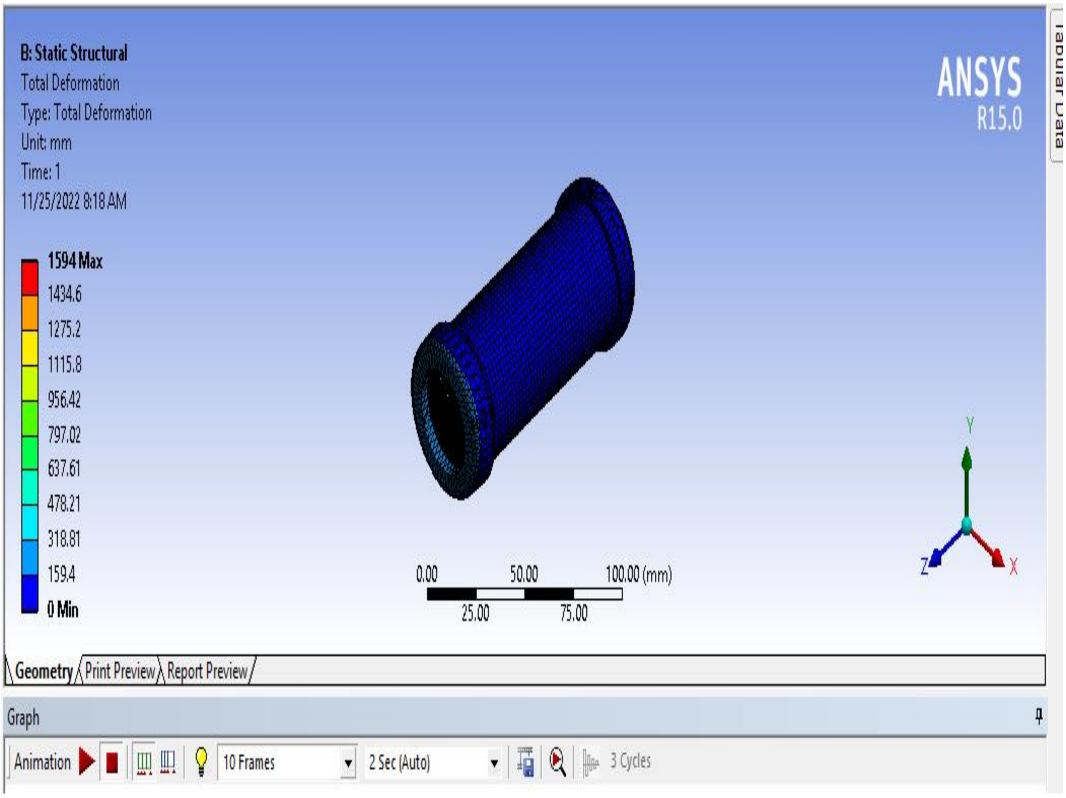

Fig. 15: Total deformation variation for titanium alloy

4. Conclusion

The conclusions derived from this study indicate that the desired parameters, including model geometry, operating pressure, combustor temperature, and ambient conditions, were established, followed by the design of the rocket combustion chamber model using SolidWorks. The simulation of the combustion model, operating at a temperature of 1500°C, was conducted using ANSYS software to assess the performance of stainless steel and titanium alloy combustion chambers. The results demonstrate that the percentage of combustion heat dissipation is 40% for stainless steel and 60% for titanium, indicating that titanium alloy has superior heat dissipation properties, necessitating more cladding to mitigate material erosion and heat absorption. This heat absorption can reduce the efficiency of the chamber in expanding combusted gases, with expansion analysis revealing that the total deflection for stainless steel ranges from 0 mm to 2.549E-3 mm, while titanium alloy ranges from 0 mm to 1.594 mm. Due to its high heat absorption, titanium may require additional carbonization and treatment in chamber fabrication, which poses challenges as titanium is difficult and costly to manufacture. The complex, energy-intensive extraction process contributes to its high price, and the limited variety of titanium alloys complicates material selection for specific applications. Consequently, manufacturers often opt for stainless steel due to its lower cost, availability, ease of fabrication, and the versatility offered by a wide range of alloying elements suitable for high-temperature applications, such as the combustion chamber of solid propellant rocket engines.

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