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Transient Thermal Analysis of Solid Propellant Rocket Motor

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ABSTRACT

Rocket engines convert the chemical energy of propellants into the kinetic energy of projectiles. Consequently, the creation of stable, high-temperature combustion of propellant is critical in its operation. Due to the high operating temperature and pressure, it is likely that the structural strength of the rocket's engine component will deteriorate. This is particularly true for small rocket engines, because, unlike large rocket engines, the small ones has no active cooling system. Heating of engine parts beyond its designed temperature will eventually increase the likelihood of mission failure. Hence, conducting a transient thermal analysis of the components during the design of the rocket engine is critical. The thermal analysis can also be used to estimate the thermal energy distribution in parts that come into contact with the heat flow induced by propellant combustion, aiding in the selection of the correct material and geometry for rocket nozzle construction. The present study is intended to analyse the transient heat transfer by observing the temperature distribution across the material and components of the rocket, including the propellant, inhibitor, nozzle, and outer casing. Besides that, the study was aimed at investigating the rate of temperature rise within a few seconds of rocket firing. A transient analysis package in SOLIDWORKS software was employed for this purpose. A constant temperature of 1326.85 ºC was imposed on the inner surface of the propellant, representing the intended peak combustion temperature, while the rest of the engine parts were set at an ambient temperature of 26.85 ºC. The analysis revealed that the inner surface of the rocket engine casing had reached a temperature as high as 181.8 °C within 0.6 s of the rocket firing. It is also found that the surface temperature increases almost linearly with time.

Keywords: Thermal Analysis, Transient Analysis, Heat Transfer, Temperature Distribution

Nomenclature

k_{prop}	Thermal conductivity of the propellant (W/(m·K))
k_{inh}	Thermal conductivity of the inhibitor (W/(m·K))
k_{case}	Thermal conductivity of the casing (W/(m·K))
h_o	Heat transfer coefficient (W/(m2·K))
T_{comb}	Temperature of the combustion (ºC)
$T_{surf,initial}$	Initial temperature of the surface (ºC)
D_A	Diameter of A (mm)
D_B	Diameter of B (mm)
D_C	Diameter of C (mm)
D_D	Diameter of D (mm)
$\sum R$	Total thermal resistance (K/W)
L	Length (mm)
$\dot{Q}$	Rate of heat transfer (J/s)
$\Delta T_{overall}$	Subtraction of T_{comb} with $T_{surf,initial}$ (ºC)
T_1	The temperature of the outer surface of KNSu propellant (ºC)
T_2	The temperature of the outer surface of the inhibitor

Abbreviations

6061	(Unified Numbering System (UNS) designation A96061) is a precipitation-hardened aluminium alloy, containing magnesium and silicon as its major alloying elements.
T6	Solution heat treated and artificially aged
PVC	Polyvinyl Chloride
KNSu	The potassium nitrate (KNO_3)/sucrose ($C_{12}H_{22}O_{11}$) propellant

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1.0 INTRODUCTION

The evolution of more powerful solid propellant rocket motors has resulted in an increase in the heat-transfer rate from the exhaust gas to the motor components, raising the temperature of these components. Component failures such as case rupture, nozzle failure, bore choking, structural failure, and ignition failure are common [1]. To avoid structural damage, heat must be eliminated around the melting temperatures of the components. Higher accuracy estimates of the amount of heat transported from the hot exhaust jet to the internal structure will lead to more efficient motor and cooling system design, increasing the likelihood of a successful mission [2]. Previously, rocket engines were built to be used only once, therefore fatigue from thermal stresses was not an issue, and correct temperature gradient calculations were not required. However, in new reusable spacecraft engines such as the SSME (Space Shuttle Main Engine), OTVE (Orbit Transfer Vehicle Engine), and HLLV (Heavy Lift Launch Vehicle), minimizing fatigue due to thermal loads is a critical component in extending engine life [3]. The thrust chambers of rocket engines work at high temperatures and pressures. This is crucial from a structural standpoint for a reason. The high operating temperature can harm several material qualities; such as yield strength [4]. One of the biggest impediments to the evolution of solid-rocket propulsion is the erosion of rocket-nozzle materials throughout engine firings. It is preferable to have a non-eroding nozzle since it maintains a consistent expansion ratio and optimal thrust efficiency.

The creation of high-temperature combustion gas and efficient and steady propellant combustion is critical in the design and operation of rocket engines [5]. Due to the high temperatures, the nozzle's strength property would deteriorate when the combustion heats up and exerts significant pressure on the nozzle wall. This will eventually increase the failure risk of the rocket body or nozzle [6]. The flow response and microstructural properties of 6061-T6 aluminum alloy are shown to be highly dependent on strain rate and temperature. Raising the strain rate or lowering the temperature causes an increase in flow stress and strain rate sensitivity. Furthermore, when the strain rate and temperature increase, so does the temperature sensitivity [7]. As a result of these concerns, performing a thermal analysis of the components during the early stages of the design of the rocket nozzle or body is critical. The thermal analysis can also be used to estimate the thermal energy distribution in parts that come into contact with the heat flow induced by propellant combustion, assisting in the selection of the correct material and geometry for rocket nozzle construction.

The present study is intended to analyse the transient heat transfer of a newly-designed solid-propellant rocket motor by observing the temperature distribution across the material and components of the rocket, including propellant, inhibitor, nozzle, and outer casing. Furthermore, the study is aimed at investigating the rate of temperature rise within the first few seconds of rocket firing. Additionally, to investigate the time required for the system to reach thermal equilibrium by assuming the worst-case scenario could occur on the rocket component with 1-dimensional heat transfer analysis. By correlating simulations and calculation results, analysis can provide significant assistance in the field of smaller full-scale results interpretation, as well as in the field of material characterization. The software involved in this study is SOLIDWORKS which was used to design the nozzle, outer casing, and propellant by assembling them, then underwent simulation for thermal transient analysis.

The future design of rocket components must be optimal and stable with the goal of reducing fatigue caused by heat stress, thus extending the engine's life. Aside from that, it is critical to do thermal analysis on rocket components before launching them into the atmosphere to avoid any incidents, such as those that occurred when the rocket failed and exploded into pieces [6]. This point is related to this ongoing research in which the investigation of transient analysis on the time of the material to reach steady-state; if the time for the material to reach thermal equilibrium is shorter than the time of the firing during combustion, then failure may occur on rocket components due to thermal loading.

2.0 METHODOLOGY

The present work analysed a rocket motor that was designed for a 10 km mission altitude using sucrose and potassium nitrate as the propellants. The motor casing consists of Aluminum 6061-T6 hollow tube with a 127 mm outer diameter, followed by the inhibitor of polyvinyl chloride (PVC Rigid) with a 117 mm diameter, KNSu propellant cartridge with a 104 mm diameter and the nozzle made of graphite. The overall setup is depicted in Figure 1.

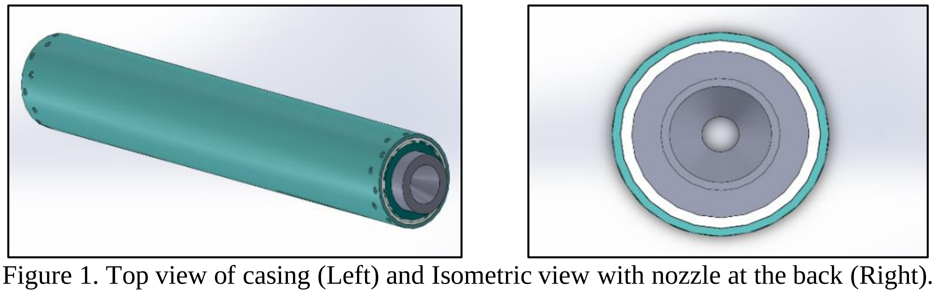

Figure 1. Top view of casing (Left) and Isometric view with nozzle at the back (Right).

The heat transfer evaluation of the rocket motor was done using a transient analysis package in SOLIDWORKS software. An initial temperature of 26.85 ºC was set for all parts and a temperature of 1326.85 ºC was designated at the combustion chamber. The simulation was conducted for 2 seconds with a 0.1 second increment in time. These parameters were chosen given the expected burn time of the propellant of 2 seconds and that the time increment is sufficient to capture the expected high-temperature gradient during the transient heat transfer.

The analysis was conducted by assuming homogeneous combustion, negligible contact resistance, and the inner surface temperature has reached a maximum combustion temperature. Prior to the analysis, a validation study was conducted using a simple long aluminum cylinder exposed to a convection environment.

Most of the properties of the materials were obtained from the SOLIDWORKS software, except for KNSu propellant, where the properties were adapted from [8]. The material properties are summarized in Table 2.

Table 1: Material Properties

Material	Mass Density (kg/m^3)	Thermal Conductivity (W(m.K))	Specific Heat (J(kg.K))
6061-T6 Aluminium	2700	166.9	896
PVC Rigid	1300	0.147	1355
Graphite	2240	168	712
KNSu	1888	0.5	~

3.0 RESULTS AND DISCUSSIONS

3.1 Solver Validation

The validation study was conducted to validate the method of transient thermal analysis using SOLIDWORKS software [9]. The study was based on a simple long aluminum cylinder, 50 mm in diameter, initially at 200 ºC, suddenly exposed to a convection environment at 70 ºC with a convection coefficient of 525 W/m^2 K. The aluminum properties are summarized in Table 2. The temperature at a radius of 12.5 mm, one minute after the cylinder was exposed to the environment must be determined and compared with the exact solution. The aluminum cylinder was designed as depicted in Figure 2.

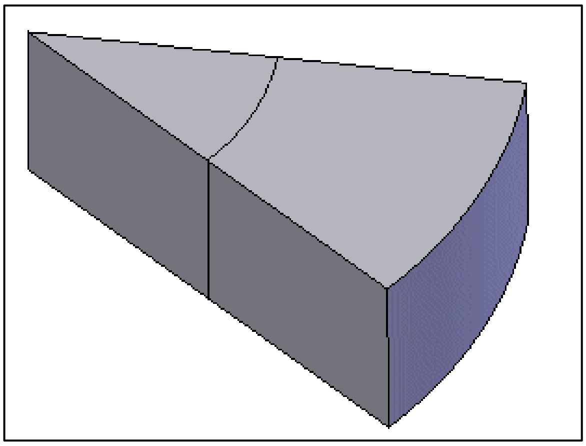

Figure 2. Long aluminium cylinder designed in SOLIDWORKS.

Table 2: Material properties of the Aluminium cylinder.

Material Properties	
Thermal Conductivity (W/m.K)	215
Density (Kg/m^3)	2700
Specific Heat (J/Kg.K)	936.8

Figure 3 shows the distribution of the temperature at which heat is dissipated through convection as it is exposed to the ambient temperature at 70 ºC; as the outermost part of the cylinder reached 118.4 ºC. Meanwhile, at a radius of 12.5 mm, the average temperature was 119.47 ºC. The result obtained was then compared with the exact solution, as presented in Table 3. It is proved that the result from simulation can be accepted as long as the error from the theory and simulation is less than 1%.

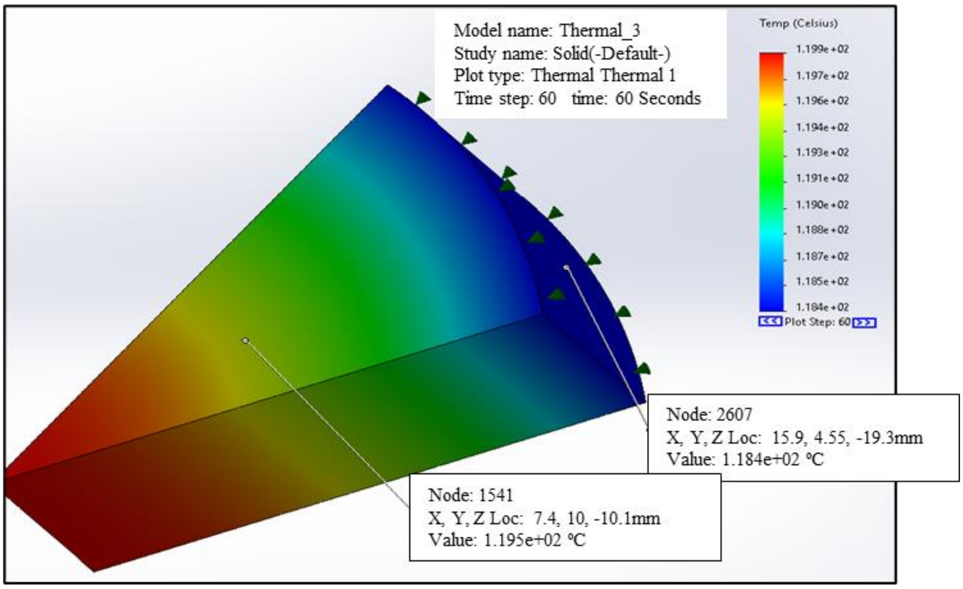

Figure 3. Result of the simulation.

Table 3: The listed temperature is the average temperature at a radius of 12.5 mm at 60 seconds.

	Temperature (ºC) at radius 12.5 mm	**Error (%)**
Exact solution	118.40	-
SOLIDWORKS Simulation	119.46 (average)	1%

3.2 One-dimensional Heat Transfer Analysis

In the present investigation, the physical phenomenon was first modelled by the transient one-dimensional heat equation in cylindrical coordinates and the material properties of the chamber were considered constant. The combustion chamber utilized in this work has a cylindrical geometry. Thus, the physical model can be described as a cylinder composite hollow rocket body casing medium of internal and external radius, r_A, r_B, r_C, and r_D, respectively, as in Figure 4. It is assumed that this problem is a one-dimensional transient heat transfer (radially) and has reached a steady-state, 25% of the propellant mass has been burned, constant *k* with temperatures and all materials are homogenous. The variables involved are;

$\boldsymbol{k_{prop}}$ = 0.5 W/(m·K)	$\boldsymbol{T_{surf,initial}}$ = 26.85 ºC
$\boldsymbol{k_{inh}}$ = 0.147 W/(m·K)	$\boldsymbol{D_A}$ = 70 mm
$\boldsymbol{k_{case}}$ = 166.9 W/(m·K)	$\boldsymbol{D_B}$ = 104 mm
$\boldsymbol{h_o}$ = 8 W/(m^2·K)	$\boldsymbol{D_C}$ = 117 mm
$\boldsymbol{T_{comb}}$ = 1326.85 ºC	$\boldsymbol{D_D}$ = 127 mm

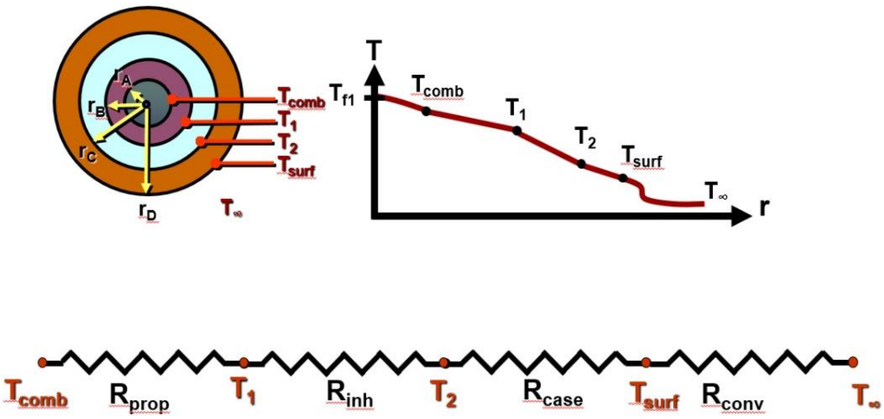

Figure 4. Series composite hollow rocket body casing and thermal network.

Figure 4 shows a typical composite hollow cylinder with both inside and outside surfaces experiencing convection heat transfer. The figure includes the thermal network that represents the system [10].

The expression for total thermal resistance is given by;

$$\sum R = \frac{ln\left(\frac{D_B}{D_A}\right)}{2\pi k_{prop}L} + \frac{ln\left(\frac{D_C}{D_B}\right)}{2\pi k_{inh}L} + \frac{ln\left(\frac{D_D}{D_C}\right)}{2\pi k_{case}L} + \frac{1}{2\pi r_D L h_o} \tag{1}$$

Once *R* has been determined, the rate of heat transfer $\dot{Q}$, can be obtained by subtracting T_{comb} with $T_{surf,initial}$ divided by total thermal resistance, *R*:

$$\dot{Q} = \frac{\Delta T_{overall}}{\Sigma R} \tag{2}$$

Then, the temperatures of the outer surface of KNSu propellant, T_1, the outer surface of the inhibitor, T_2, and the outer surface of the Aluminum casing, T_{surf}, can be determined from:

$$T_1 = T_{comb} - \dot{Q} \cdot R_{prop} \tag{3}$$

$$T_2 = T_1 - \dot{Q} \cdot R_{inh} \tag{4}$$

$$T_{surf} = T_2 - \dot{Q} \cdot R_{case} \tag{5}$$

Even though these equations were based on steady-state heat transfer analysis, they can be compared with the simulations conducted in SOLIDWORKS software to verify the authenticity of the values especially on the Aluminium casing of the rocket component. The value of T_1 was then checked on its standard melting point of 6061-T6 Aluminium which is at 650 ºC.

3.3. Transient Analysis

This section presents the initial findings of a transient thermal analysis on the rocket casing. Table 4 shows the temperature data at six different locations on the interior surface of the Aluminium 6061-T6 casing, precisely at burning time of t = 0.6 seconds. It was found that the highest temperature is in the 4th location at 181.8 ºC. The probe locations are indicated in Figure 5.

Table 4: Data were tabulated after probing at each location on the interior surface of the Aluminium 6061-T6 casing.

Locations	Temperature (°C) at 0.6 sec	X-axis (mm)	Y-axis (mm)	Z-axis (mm)
1	27.67	421.48	1075.66	740.24
2	27.68	421.80	1054.04	740.24
3	141.00	421.95	159.07	740.24
4	181.80	421.80	150.04	740.24
5	180.70	421.22	134.75	740.24
6	27.95	421.62	75.95	740.24

As indicated in Figure 5, the highest temperature was recorded on the interior surface of the aluminium casing near the nozzle assembly. The heat seems to dissipate much faster near this location, which is likely due to the presence of O-ring in the nozzle assembly. The O-ring acts as a thermal resistor, which increase the resistance of heat transfer in the axial direction and diverts the heat in the radial direction. Consequently, the local surface temperature of the aluminium casing in that particular area increased.

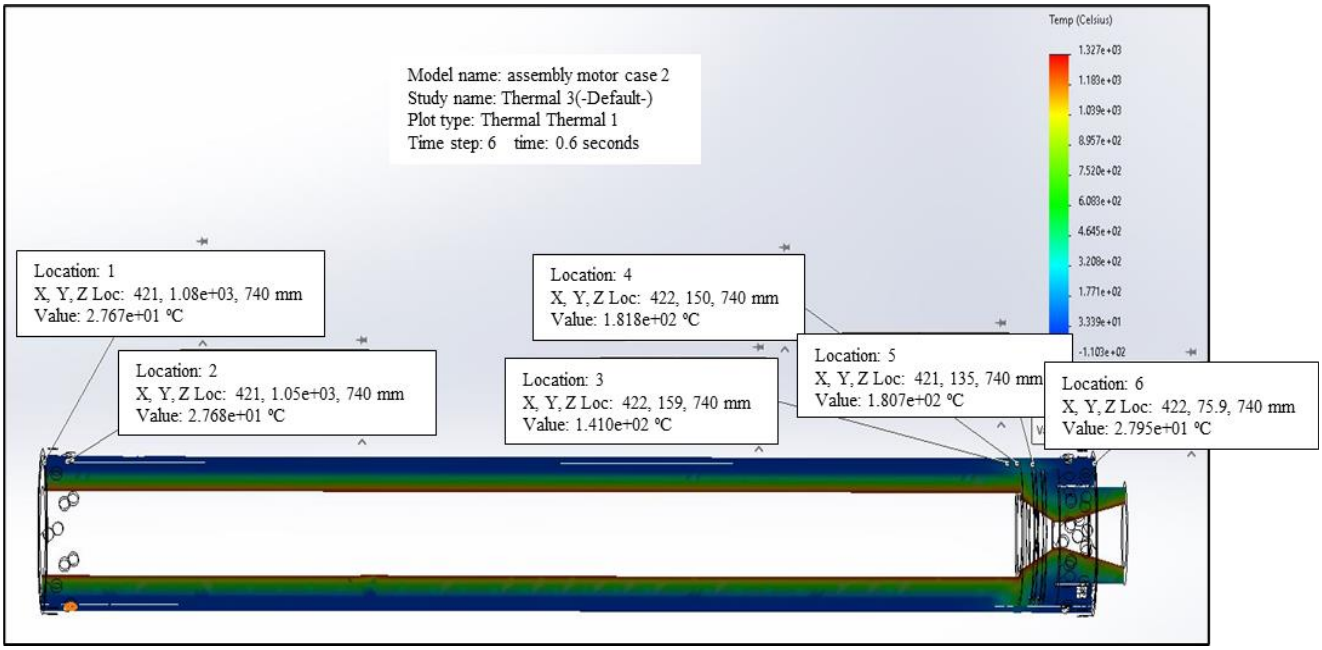

Figure 5. The probe values at 6 locations on the interior aluminum casing.

Table 5 summarizes the data which was used to plot the graph as shown in Figure 6. The data was tabulated within 2 seconds at each of the 6 probe locations (L).

Table 5: Time history of surface temperature at various indicated locations.

Time (sec)	L1	L2	L3	L4	L5	L6
0.1	27.03	27.03	85.52	114.21	128.00	27.00
0.2	26.48	26.48	94.34	125.24	129.66	27.40
0.3	26.48	26.48	107.03	139.59	136.83	27.59
0.4	27.03	27.03	119.17	155.03	148.41	27.70
0.5	27.59	27.59	131.31	169.93	161.10	27.85
0.6	**27.67**	**27.68**	**141.00**	**181.80**	**180.70**	**27.95**
0.7	27.70	27.70	153.93	198.07	188.69	32.55
0.8	27.95	27.95	164.97	210.21	201.93	33.10
0.9	28.14	28.14	174.90	224.00	215.17	34.21
1.0	28.14	28.14	184.38	236.14	227.86	35.86
1.1	28.14	28.14	193.66	247.17	241.66	36.41
1.2	28.14	28.14	201.93	258.76	252.69	39.72
1.3	28.69	28.69	210.21	269.24	264.83	41.93
1.4	28.14	28.14	217.93	279.17	276.41	44.69
1.5	28.69	28.69	224.55	288.00	288.55	48.00
1.6	28.14	28.14	232.28	296.83	298.48	51.31
1.7	28.69	28.69	238.35	304.55	308.97	55.17
1.8	29.24	29.24	244.97	313.38	318.24	59.59
1.9	29.24	29.24	250.48	320.00	328.28	64.00
2.0	28.69	28.69	256.00	327.17	338.21	68.97

Figure 6 shows the time-history of temperatures for six probe locations within 2 seconds of combustion duration. From the plot, it can be observed that within 0.6 seconds, the inner surface of the aluminium casing (location 4) has reached the temperature of 181.8 °C. Although this temperature is much lower than the melting point of aluminium, which is 650 °C, the tensile strength and ductility decreased by 14% and 20%, respectively [11]. Given the high operating pressure, the decrease in these properties can significantly increase the possibility of material failure, which is most likely ductile fracture near the nozzle assembly.

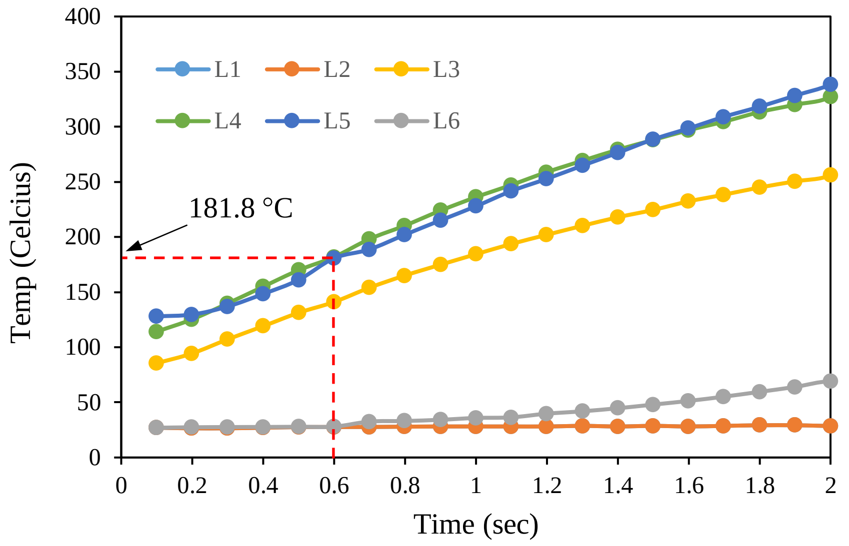

Figure 6. Surface temperature plotted against firing time.

Figure 7 shows the instantaneous temperature distribution while the combustion chamber is heated up for 0.6 seconds. The highest temperature reached was 1327 ºC which occurred at the inner surface of the propellant. It can be observed that the heat distributed at the nozzle has reached the outer surface of the aluminum casing.

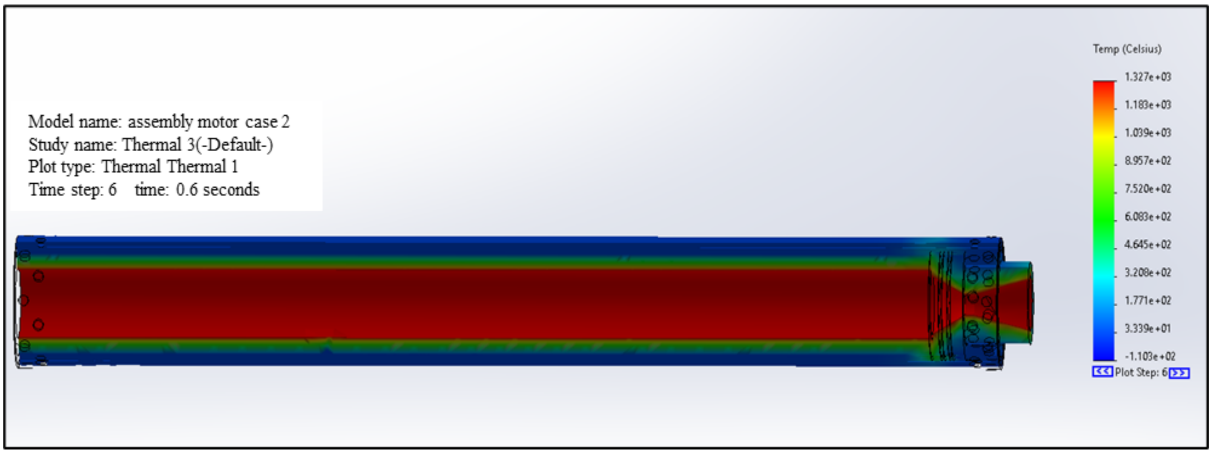

Figure 7. Cross-sectional view of the temperature distribution at firing time of 0.6s.

4.0 CONCLUSION

The present study conducted transient thermal analysis on the rocket motor intender for 10 km mission altitude range application. The heat transfer was simulated for 2 seconds with a 0.1 second increment in time. It was found that the inner surface of the aluminium casing had reached the highest temperature of 181.8 °C within 0.6 seconds. The high rate of heat transfer is attributed to the high operating temperature gradient and the lack of the thermal insulation layer. The analysis also found that the temperature gain near the bulkhead and nozzle assembly is insignificant. Overall, the thermal analysis is crucial, particularly for the design of reusable rocket engine where the engine is intended to be used multiple times across different missions.

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