Students' Space Association
WARSAW UNIVERSITY OF TECHNOLOGY

# Combustion chamber thermal insulation analysis

## TWR2

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# Contents

# 1 Description of the problem

## 1.1 introduction to the problem

Main focus of given analysis was determining the initial thickness of combustion chamber thermal insulation. Due to hyperthermal environment, combustion processes and highly turbulent flow inside of the rocket engine's combustion chambers the main difficulty in heat transfer related problems is approximating the heat transfer coefficient via convection between the inner wall of combustion chamber's thermal insulation and hyperthermal environment. The values of Heat transfer coefficient is usually approximated by computational simulation or empirical equations. All of mentioned methods usually requires validation during hotfire test in order to prove that thermal insulation works fine. Additionaly to heat transfer coefficient calculations, properties of exhaust gases of given proppelant combination had to be estimated as input values.

## 1.2 description of the problem

In given case the combustion chamber is considered as a axisymmetric tube with nozzle on one end. The heat transfer in the nozzle is not considered in this analysis however due to used equations later in this analysis the nozzle geometry was needed for calculations. The thermal insulation must provide that the temperature of the combustion chamber wall is not higher than $165°C$. This temperature was choosen as maximum temperature at which carbon-epoxy composite material used for combustion chamber still posses its mechanical properties and integrity. The heat transfer problem was considered as an unsteady state.

# 2 Input data for the Analysis

## 2.1 NASA CEA input data

Choosen oxidizer and fuel for the rocket was $N_2O$ and $ABS$. For finding the values of thermal properties of rocket exhaust gasues, in frozen conditions, following initial data was presented in table and figure (Fig.1) below:

| NASA CEA input | | | |
|---|---|---|---|
| Name | Symbol | Value | Unit |
| Combustion Temperature | $T_0$ | 3400 | $K$ |
| Pressure in combustion chamber | $P_0$ | 4.4 | $MPa$ |
| Oxidizer to Fuel Ratio | O/F | 5 | - |
| ABS Molecular weight | M | 57.07 | $\frac{kg}{kg \cdot mol}$ |
| ABS Enthalpy of formation | $\Delta H$ | 63.63 | $\frac{kJ}{s}$ |
| Chemical formula of ABS | $C_{3.85}H_{4.85}N_{0.43}$ | | |

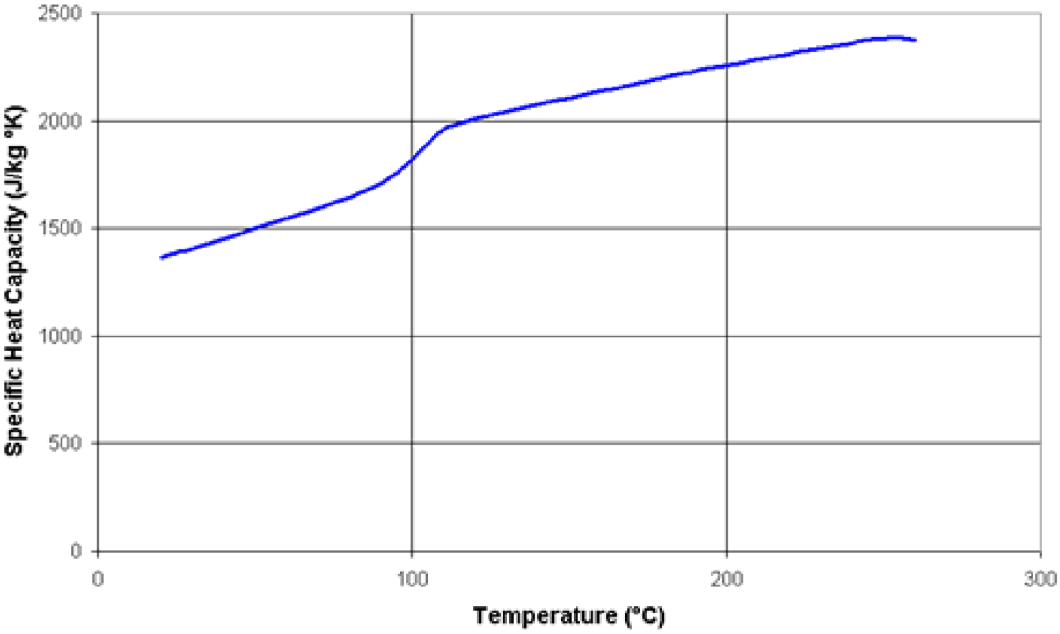



Figure 1: Specific Heat as a function of Temperature for ACRYLONITRILE BUTADIENE STYRENE

## 2.2 Insulation thermal properties

Bakelit, paper-phenol resin composite, was used as a thermal insulation of combustion chamber walls. Paper based composite was used as it doesn't leak in comparison to cotton or fiber based composites, this fact is crucial as some of the thermal insulation faces act together with O-rings as seals for the combustion chamber.

| Insulation thermal properties | | | |
|---|---|---|---|
| Name | Symbol | Value | Unit |
| Density | $\rho_i$ | 1300 | $\frac{kg}{m^3}$ |
| Specific heat | $c_i$ | 1507 | $\frac{J}{kgK}$ |
| Thermal conductivity | $\lambda_i$ | 0.337 | $\frac{W}{m^2K}$ |

## 2.3 Combustion chamber and Nozzle geometry

Dimensions of combustion chamber and nozzle needed for calculation are given in table and Figure (Fig. 2) below:

| Nozzle geometry | | | |
|---|---|---|---|
| Name | Symbol | Value | Unit |
| Nozzle throat curvature | $r_c$ | 0.02 | $m$ |
| Nozzle throat diameter | $D_*$ | 0.045 | $m$ |
| Combustion chamber diameter | $D$ | 0.122 | $m$ |

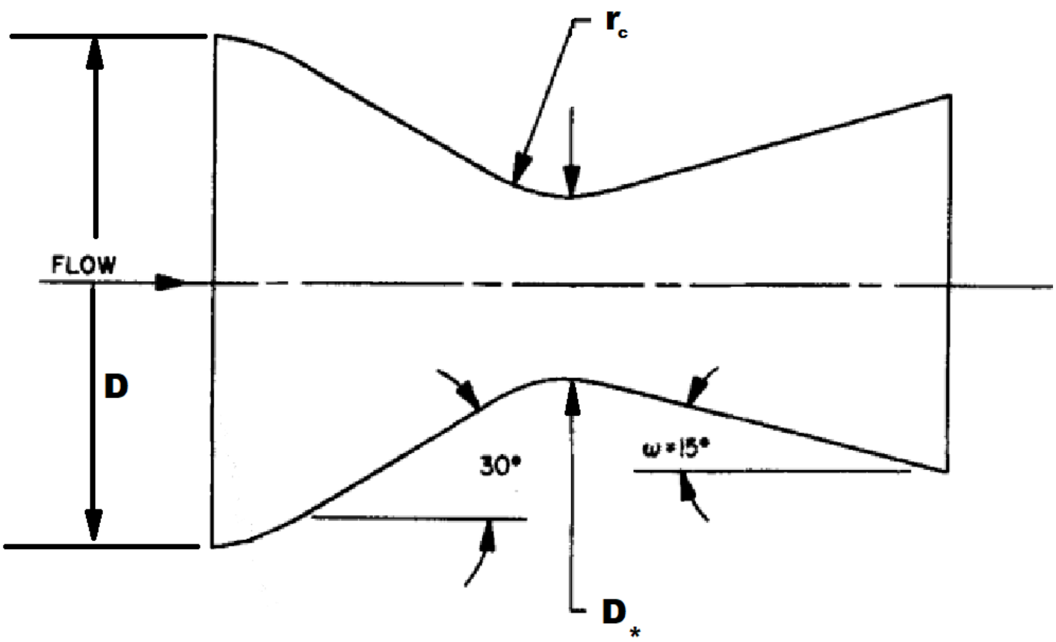



Figure 2

# 3 Tools used during Analysis

For this analysis following programs and papaers were used:

1. Turbulent Boundary-Layer Heat Transfer from Rapidly Accelerating Flow of Rocket Combustion Gases and of Heated Air - D . R . BARTZ 1965
2. SKA, Sekcja rakietowa: Silniki CSL Opis ogólny - Tadeusz Górnicki, Warszawa kwiecień 2016
3. NASA/TP—2002-211556, NASA Glenn Coefficients for Calculating Thermodynamic Properties of Individual Species, Bonnie J. McBride, Michael J. Zehe, and Sanford Gordon Glenn Research Center, Cleveland, Ohio
4. Comparing Hydroxyl Terminated Polybutadiene and Acrylonitrile Butadiene Styrene as Hybrid Rocket Fuels, Stephen A. Whitmore,Zachary W. Peterson, and Shannon D. Eilers† Utah State University, Logan, Utah 84322
5. Heisler graphs for transient conduction
6. NASA CEA
7. ANSYS Transient Thermal

# 4 Course of the analysis

## 4.1 Adding ABS into NASA CEA database

First step in order to start heat transfer calculation was adding ABS into NASA CEA thermodynamic database. It was necessary in order to estimate thermodynamic properties of exhaust gases inside of combustion chamber. The whole process is done according to Appendix C of [3]. The first step was finding coefficients $a_1, a_2, a_3, a_4, a_5, a_6, a_7, b_1$ and $b_2$ that appear in equations 1, 2 and 3:

$$\frac{C_p(T)}{R'} = a_1T^{-2} + a_2T^{-1} + a_3 + a_4T + a_5T^2 + a_6T^3 + a_7T^4 \tag{1}$$

$$\frac{H(T)}{R'T} = -a_1T^{-2} + a_2\frac{ln(T)}{T} + a_3 + a_4\frac{T}{2} + a_5\frac{T^2}{3} + a_6\frac{T^3}{4} + a_7\frac{T^4}{5} + \frac{b_1}{T} \tag{2}$$

$$\frac{S(T)}{R'} = -a_1\frac{T^{-2}}{2} - a_2T^{-1} + a_3ln(T) + a_4T + a_5\frac{T^2}{2} + a_6\frac{T^3}{3} + a_7\frac{T^4}{4} + b_2 \tag{3}$$

Where:

- T - Temperature $[K]$
- $C_p(T)$ - molar heat capacity at constant pressure at temperature T for standard state $[\frac{J}{mol \cdot K}]$
- $H(T)$ - molar enthalpy at temperature T $[\frac{J}{mol}]$
- $S(T)$ - entropy at temperature T for standard state $[\frac{J}{mold \cdot K}]$
- $R'$ - universal gas constant, 8.314510 $[\frac{J}{mol \cdot K}]$

The graph presented on Figure (Fig. 1) was approximated by three linear functions in three different ranges of Temperature. Enthalpy equation was calculated from $H(T) = \int C_p(T)dT$, fact that $H(298.15) = \Delta H$ (given in [3]) and fact that $H(T)$ must be continuous function. The function of Entropy was omitted as changing $b_2$ did not effect output values of needed thermodynamic properties of exhaust gases. The values final form of functions are given below:

$$\frac{C_p(T)}{R'} = A \cdot T + B \qquad \frac{H(T)}{R'} = \frac{1}{2}A \cdot T^2 + B \cdot T + C \tag{4}$$

| Range of temperatures $[K]$ | A (=$a_4$) | B (=$a_3$) | C (=$b_1$) |
|---|---|---|---|
| 288 - 365.809 | 0.03431971651 | -0.7893534798 | 6242.604806 |
| 365.809 - 383.747 | 0.09955056052 | -24.65137189 | 10607.07327 |
| 382.747 - 530 | 0.0212095848 | 5.411738818 | 4838.759703 |

The ABS molar weight was calculated from its chemical fromula $C_{3.85}H_{4.85}N_{0.43}$ but just as $\Delta H$ its vale was taken from [4].
All of the data was added to NASA CEA files by adding it into Thermo.inp and running the program. Data was added to Thermo.inp in following format showed in Figure (Fig 3) with explanation of every line written in table below:

| Record | Contents | Columns |
|---|---|---|
| 1 | Species name of formula<br>Coments and data sources | 1 to 16<br>19 to 80 |
| 2 | Number of T intervals<br>Reference date code<br>Chemical formula - symbols (all capitals) and numbers<br>Zero for gas; nonzero for condensed<br>Molecular weight<br>Heat of formation at 298.15 $K$, $[\frac{J}{mol}]$ | 1 to 2<br>4 to 9<br>11 to 50<br>51 to 52<br>53 to 65<br>66 to 80 |
| 3 | Temperature range<br>Number of coefficients fot $\frac{C_p(T)}{R'}$ (always seven)<br>T exponents in empirical equation for $\frac{C_p(T)}{R'}$<br>[always -2, -1, 0, 1, 2, 3, 4]<br>$H(298.15) - H(0)$ $[\frac{J}{mol}]$, if aviable | 1 to 22<br>23<br>24 to 63<br><br>66 to 80 |
| 4 | First five coefficient for $\frac{C_p(T)}{R'}$ | 1 to 80 |
| 5 | Last two coefficient for $\frac{C_p(T)}{R'}$<br>Integration constants $b_1$ and $b_2$ | 1 to 32<br>49 to 80 |
| — | Repeat 3, 4, and 5 for each interval | — |

```
12345678901234567890123456789012345678901234567890123456789012345678901234567890
         10        20        30        40        50        60        70        80

TiN(cr)            Chase,1998 pp1612-4.
 2 j 6/68 TI  1.00N   1.00    0.00    0.00    0.00 1     61.87374    -337648.800
    200.000    800.0007 -2.0 -1.0  0.0  1.0  2.0  3.0  4.0  0.0            5487.000
-5.479117220D+05 9.328691110D+03-6.386263890D+01 2.429925456D-01-4.304234520D-04
 3.792645100D-07-1.317412256D-10                -8.424256140D+04 3.392988560D+02
    800.000   3220.0007 -2.0 -1.0  0.0  1.0  2.0  3.0  4.0  0.0            5487.000
-3.656247060D+05 1.265730431D+03 3.831711190D+00 1.632900455D-03-1.062786626D-07
 1.310931390D-11-5.770548410D-16                -5.027654400D+04-1.652632899D+01
TiN(L)             Chase,1998 pp1612-4.
 1 j 6/68 TI  1.00N   1.00    0.00    0.00    0.00 2     61.87374    -337648.800
   3220.000   6000.0007 -2.0 -1.0  0.0  1.0  2.0  3.0  4.0  0.0            5487.000
 0.000000000D+00 0.000000000D+00 7.548249987D+00 0.000000000D+00 0.000000000D+00
 0.000000000D+00 0.000000000D+00                -3.626039860D+04-3.958296649D+01

12345678901234567890123456789012345678901234567890123456789012345678901234567890
         10        20        30        40        50        60        70        80
```

Figure 3: Example of input data for thermo.inp

The ABS thermo.inp file for given data is presented in the figure below (Fig. 4) :

```
ABS                  TWR2, added based on Cp(T) and heat of formation delta_H(T)
 3 gl3/13 C   3.85H   4.85N   0.43    0.00    0.00 1   57.1537300      62630.000
    288.000    365.8097 -2.0 -1.0  0.0  1.0  2.0  3.0  4.0  0.0             0.000
 0.000000000D+00 0.000000000D+00-7.893534798D-01 3.431971651D-02 0.000000000D+00
 0.000000000D+00 0.000000000D+00                 6.242604806D+03 0.000000000D+00
    365.809    383.7477 -2.0 -1.0  0.0  1.0  2.0  3.0  4.0  0.0             0.000
 0.000000000D+00 0.000000000D+00-2.465137189D+01 9.955056052D-02 0.000000000D+00
 0.000000000D+00 0.000000000D+00                 1.060707327D+04 0.000000000D+00
    382.747    530.0007 -2.0 -1.0  0.0  1.0  2.0  3.0  4.0  0.0             0.000
 0.000000000D+00 0.000000000D+00 5.411738818D+00 2.120958480D-02 0.000000000D+00
 0.000000000D+00 0.000000000D+00                 4.838759703D+03 0.000000000D+00
```

Figure 4: ABS data for thermo.inp

### 4.2 Getting thermal properties of the exhaust gases from NASA CEA

For NASA CEA "Rocket" case was used, both frozen and equlibrum values of thermal properties were calculated but frozen values was choosen for calculations. (equilibrium values does not apply together into perfect gas equations) Input data file for NASA CEA was given as (Fig. 5):

```
problem    o/f=5,
      rocket  equilibrium  frozen  nfz=1  t,k=3400
  p,bar=44,
react
  fuel=ABS wt=100
  oxid=N2O wt=100
output  transport
    plot pranfz pran condfz cond vis
end
```

Figure 5: Input file for NASA CEA

given input values resulted in following thermal properties of the exhaust gases:

| Thermal properties of exhaust gases | | | |
|---|---|---|---|
| Name | Symbol | Value | Units |
| Mollar mass | $M_g$ | 25.787 | $\frac{kg}{kmol}$ |
| Specific heat at constant pressure | $c_p$ | 1584 | $\frac{J}{kgK}$ |
| Specific heat ratio | $k$ | 1.2556 | - |
| Conductivity | $\lambda$ | 0.24128 | $\frac{W}{mK}$ |
| Viscosity | $\mu$ | $0.99384 \cdot 10^{-4}$ | $\frac{kg}{ms}$ |
| Prandtl number | $Pr$ | 0.6524 | - |

## 4.3 Calculating heat transfer coefficient

Before calculating heat transfer coefficient following additional Properties had to be calculated:

$$R = c_p \frac{k-1}{k} \qquad \dot{m} = \left(\frac{\pi D_*^2}{4}\right) P_0 k \frac{\sqrt{\left(\frac{2}{k+1}\right)^{\left(\frac{k+1}{k-1}\right)}}}{\sqrt{kRT_0}} \qquad \rho U = \frac{\dot{m}}{\frac{\pi D^2}{4}} \tag{5}$$

Where:

- R - Specific gas constant for exhaust gases = 322.451736222 $[\frac{J}{kgK}]$
- $\dot{m}$ - mass flow rate = 4.40508684258 $[\frac{kg}{s}]$
- $\rho U$ - multiplication of exhaust gases density and velocity = 376.829532785 $[\frac{kg}{m^2 s}]$

For calculating heat transfer coefficient the method showed in [1] was used. The following equation was used:

$$\alpha = 0.8209850291 \cdot \frac{0.026}{D^{0.2}} \left(\frac{D_*}{r_c}\right)^{0.1} \left(\frac{\mu^{0.2} c_p}{Pr^{0.6}}\right) (\rho U)^{0.8} \left(\frac{T_w}{T_0}\right)^{-1} \left(\frac{T_w}{2T_0} + `1\right)^{\frac{0.65}{5}} \tag{6}$$

Where $T_w$ is temperature of the insulation wall and " 0.8209850291" part is correction needed for SI units. Usage of $T_w$ creats need for solving this problem iteratively. The iterative process, involving equations that will be presented later in this document, shows that $T_w = 3042.6K$ gives us satisfying results. The calculated value of heat transfer coefficient to be

$$\alpha = 1540.96613033 \left[\frac{W}{m^2 K}\right]$$

## 4.4 Calculating temperatures on insulation

Given problem is unstedy state/transient heat transfer problem. In order to calculate it we have to use Heisler graphs. Due to very limited possibilities of solving transient heat transfer problems on paper we will have to consider only small part of our combustion chamber wall and treat it as infinite plane. The Heisler graphs were taken from book titled "W. Gogół - Wymiana ciepła, tablice i wykresy" and are shown below on Figures 6 and 7. The $\vartheta = \frac{T-T_0}{273.15-T_0}$ where T is the temperature at given point in time.

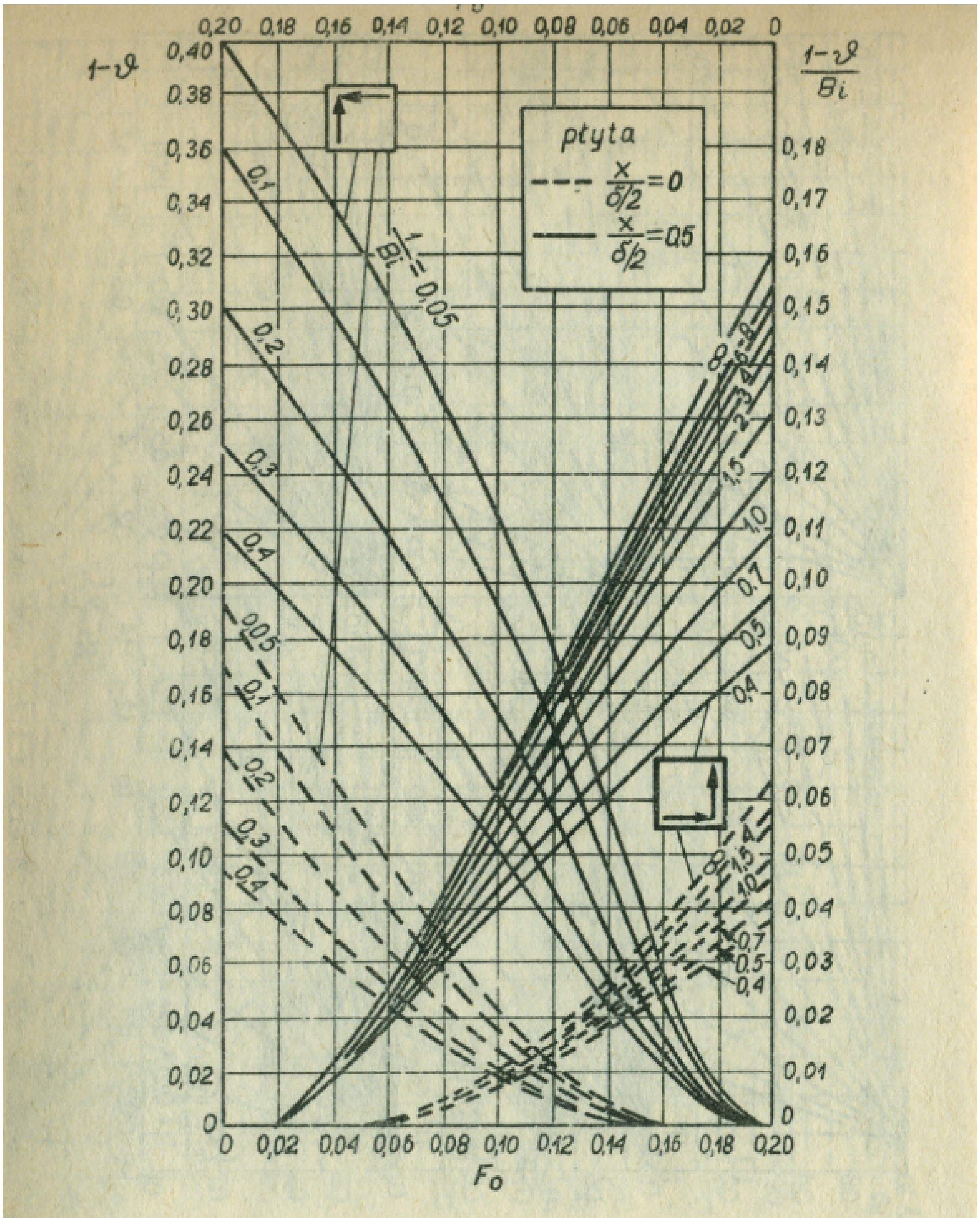
In order to estimate the temperature on the inner and outer surface of thermalinsulation Fourier and Biot numbers were calculated:

$$Fo = \frac{\lambda_i}{\rho_i c_i} \cdot \frac{t}{\delta^2} \qquad\qquad Bi = \frac{\alpha \delta}{\lambda_i} \tag{7}$$

Where the total work-time of the engine is $t = 9s$ and $\delta = 0.004m$ is the thickness of thermal insulation. With the given data values calculated properties are given in the table below:

| Name | Symbol | Value | Units |
|---|---|---|---|
| Fourier number | $Fo$ | 0.086 | - |
| Biot Number inverse | $Bi^{-1}$ | 0.05467 | - |
| Insulation inner surface Temperature | $T_w$ | 360.7018 | $K$ |
| Insulation outer surface Temperature | $T_{out}$ | 3042.6 | $K$ |

Further changes in $\alpha$ and $T_w$ will give us changes of Bi that are too small to read from Heisler graphs.




Rys.5.14.Wykres Heislera do obliczania temperatury wewnątrz płyty nieskończonej dla małych wartości liczb Fouriera

Figure 6: Temperature inside the infinite plate

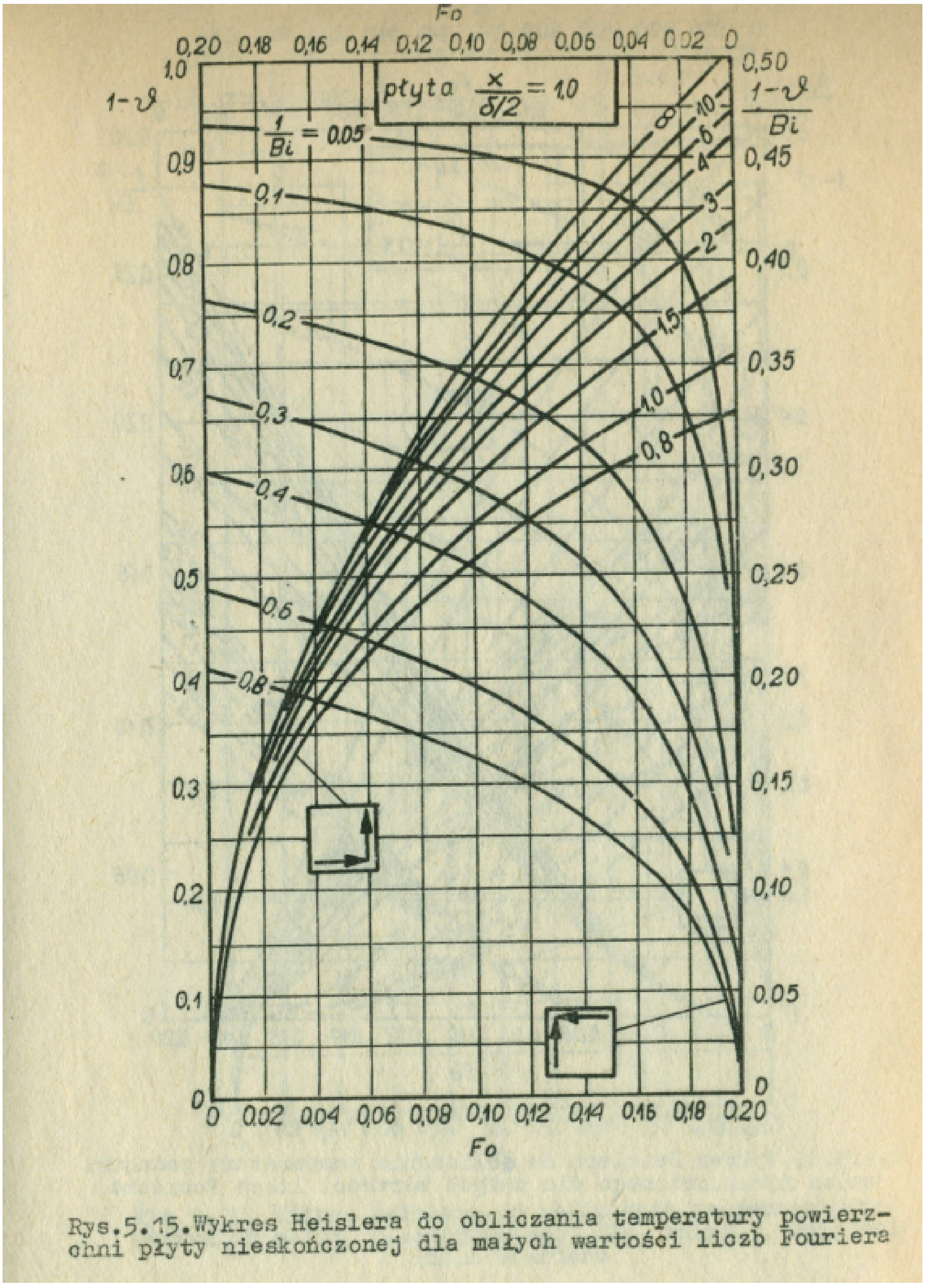



Figure 7: Temperature on the surface of infinite plane

## 4.5 Simulation using Ansys transient thermal

In order to validate used empirical solutions computational simulation of transient thermal case was conducted using ANSYS.

The used model is shown on the figure 8. The inner surface was treated with convection heat transfer boundary conditions with calculated $\alpha$ as heat transfer coefficient and $T_0$ as ambient temperature of environment value. initial Temperature was equal to $20°C$ and every other surface was treated as perfectly insulated. The solution (Fig 9) showed that results of semi-empirical equation gave similar values with around 25 K difference.

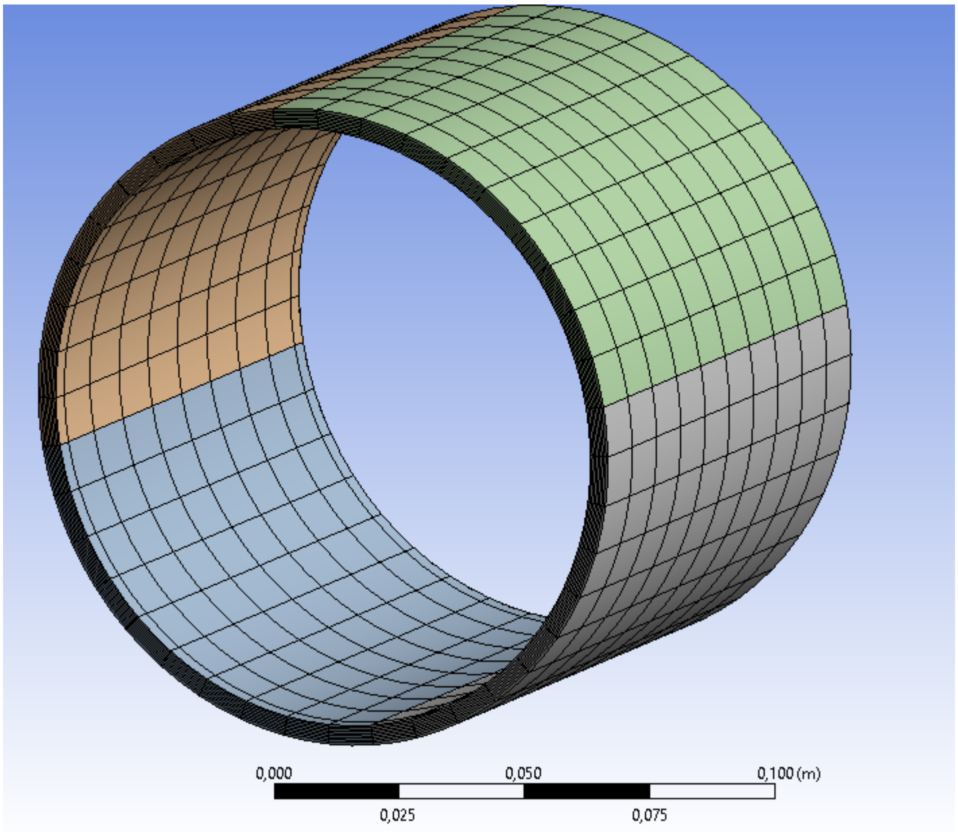



Figure 8: model used for simulation

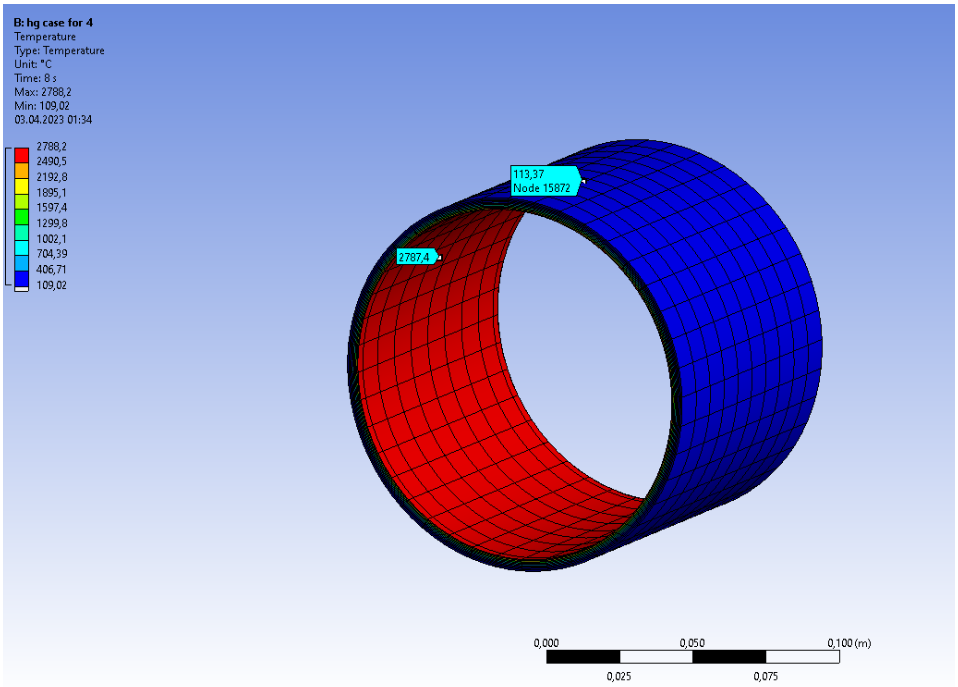



Figure 9: solution of the simulation

# 5 Analysis output

From conducted analysis the thickness of the combustion chamber thermal insulation shall be be eaqual to $\delta = 4mm$.