TRANSACTIONS OF THE INSTITUTE OF AVIATION
3 (252) 2018, pp. 87–102
DOI: 10.2478/tar-2018-0024

PRELIMINARY DESIGN ANALYSIS OF REGENERATIVE COOLING FOR N_2O/ALCOHOL SMALL SCALE LIQUID ROCKET ENGINE

Patryk Palej, Tomasz Palacz
AGH Space Systems, AGH University of Science and Technology
ul. Mickiewicza 30, 30-059 Kraków, Poland
patrykpalej@gmail.com, palacztt@gmail.com

Abstract

This paper presents a concept of a small scale liquid-propellant rocket engine designed in AGH Space Systems for sounding rocket. During preliminary design of thermal aspects various ways of cooling were evaluated and described. Possible issues and design approaches for ablative, radiation and regenerative cooling are raised. The authors describe available solutions. Regenerative cooling is especially concerned as it is most popular solution in bi-liquid engines, in which alcohol fuel acts as coolant and is preheated before it reaches combustion chamber. To estimate a possible temperature distribution - and thus an applicability of such a system in the engine - a mathematical model of heat transfer was developed. Unique element of said engine is its oxidizer - nitrous oxide, which have been rarely used to date. Comparison between typical LOX bi-liquids is given and major differences that affect cooling arrangement are discussed. The authors compared different combinations of coolants, fuel/oxidizer ratios etc. to optimize the temperature distribution which is a key factor for the engine performance.

Keywords: liquid rocket engine, regenerative cooling, nitrous oxide, sounding rocket.

Subscripts

acu	–	Accumulated	*L*	–	Left
am	–	Arithmetic mean	*out*	–	Outer
cool	–	Coolant	*R*	–	Right
f	–	Fluid	*w*	–	Wall
h	–	Hydraulic	*wg*	–	Wall, gas side
in	–	Inner	*wc*	–	Wall, coolant side
0	–	Initial			

Nomenclature

A	–	Area [m^2]	r	–	Radius [m]
c_p	–	Const. Pressure specific heat $\left[\frac{J}{kgK}\right]$	R	–	Thermal resistance $\left[\frac{m^2K}{W}\right]$
d	–	Wall thickness, diameter [m]	Re	–	Reynolds numer [-]
h	–	Convective heat transfer coefficient $\left[\frac{W}{m^2K}\right]$	q	–	Heat flux $\left[\frac{W}{m^2}\right]$
k	–	Thermal conductivity $\left[\frac{W}{mK}\right]$	T	–	Temperature [K]
L^*	–	Characteristic length [m]	α	–	Heat diffusion coefficient $\left[\frac{m^2}{s}\right]$
$\dot{m}$	–	Mass flux $\left[\frac{kg}{s}\right]$	λ	–	Heat conductivity $\left[\frac{W}{mK}\right]$
Nu	–	Nusselt numer [-]	μ	–	Dynamicviscosity $[Pa \cdot s]$
Pr	–	Prandtl numer [-]	ρ	–	Density $\left[\frac{kg}{m^3}\right]$

1. INTRODUCTION

Since 2016 in AGH Space Systems there is ongoing research of N_2O/alcohol liquid rocket engines. For purpose of testing the working prototype project Zawisza was started. Its main goal is to design, build and test a small scale N_2O/alcohol rocket engine for a sounding rocket. In September of 2017, Zawisza Z1kN has been successfully test fired. Ablative cooling has been incorporated into this prototype, although from the very beginning Zawisza was intended to be cooled regeneratively and serve as a small scale technology demonstrator for larger units, which would admittedly benefit from such cooling configuration. There are several documented successful projects of test scale N_2O bi-liquids [1][2]. However, there is no research concerning regenerative cooling of such engine. Moreover, N_2O bi-liquid differs significantly from LOX engines and these major differences affect cooling system [3]. Usually, modern regenerative cooled thrust chambers incorporate high-aspect ratio cooling passages to maximize cooling efficiency. These are however difficult and expensive to manufacture. Some smaller units were successfully tested with coaxial shell cooling configuration. Despite the fact that this configuration is no longer used it is much easier to incorporate into small bi-liquid engine. In order to efficiently design regeneratively cooled N_2O rocket engine the theoretical investigation of heat transfer phenomena must be performed. For that purpose, calculations of heat transfer for Zawisza engine has been carried out using exclusively developed mathematical model for preliminary design of cooling arrangements.

2. COOLING SYSTEMS IN ROCKET ENGINES

2.1. Regenerative cooling

Regeneratively cooled rocket engine employs one of its propellants (usually fuel) as coolant, which is fed into the cooling channels around combustion chamber, therefore convectively cools chamber wall before injection into the combustor [4]. In order to provide sufficient cooling temperature of interior side of chamber wall (T_{wg}), which is directly exposed to hot combustion gases (T_{aw}), must be lowered to some acceptable level. This is usually temperature at which material of chamber still has enough strength to withstand all accompanying stresses. Other limitations of regenerative cooling encompass maximum allowable coolant temperature (called critical temperature), chamber wall thickness or fuel pressure drop in cooling channels. As have been shown heat flux varies strongly within the combustion chamber and is few times larger in nozzle throat area than other parts of the chamber [5].

Regenerative cooling is in most cases considered as a steady-state process, in which an acceptable temperature distribution occurs in the combustion chamber and nozzle wall. Given that condition holds up, regenerative cooling can work virtually for the infinite time and is only limited by available amount of propellants. Moreover, as some of the heat is transferred to the fuel there is slight increase in specific impulse of such engine owing to regain of certain amount of energy, which would be otherwise lost as a heat to the walls. Although this boost in performance is minor, regenerative cooling is most popular in first stage engines of launch vehicles [6].

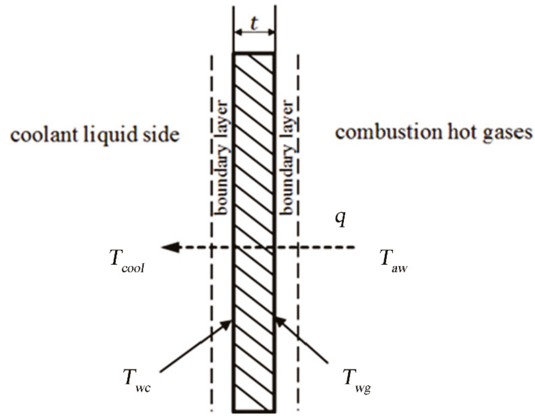


Fig. 1. Regenerative cooling diagram for rocket engine.

2.2. Heat sinks

An alternative to steady state process of regenerative cooling is unsteady process of heat sink. In such case, a one-dimensional model neglecting curvature of chamber walls can be considered as shown on Fig. 2.

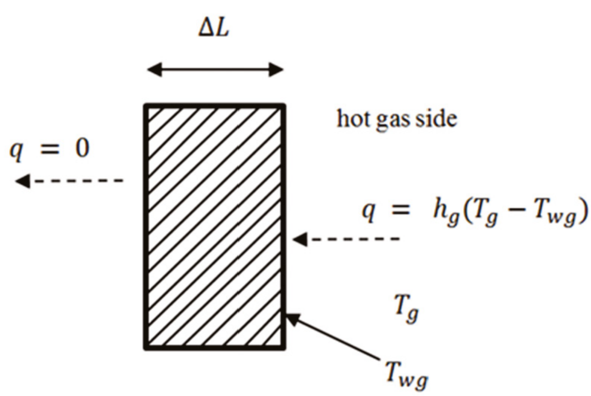


Fig. 2. Simple, one-dimensional model of the heat sink

For simplicity, assume that radiation heat transfer can be neglected as well as heat transfer from the wall to outer side is negligible. In this case, equation describing such conduction is reduced to [7]:

$$\frac{\partial^2 T}{\partial x^2} = \frac{1}{\alpha}\frac{\partial T}{\partial t} \tag{1)}$$

where x is coordinate along wall thickness ΔL from the cold gas side and α is thermal diffusivity of the wall material. Solution to this equation depends on thermal diffusivity of the solid, wall thickness and heat flux to the wall. Assuming boundary conditions for $x = 0$ and $x = \Delta L$ as initial wall temperature T_0 and convective heat transfer with coefficient h_g respectively, the time t an be found after which wall temperature T_{wg} on the hot gas side reaches given allowable temperature.

With use of above model of combustion chamber wall heating, short study for applicability has been performed for Zawisza rocket engine. Given that stainless steel is used as chamber material with *3*mm wall thickness. Gas temperature was assumed to *2660*K and maximum allowable wall temperature set to *1200*K. With such conditions burn time is limited to maximum 2 seconds of full thrust. Increase of wall thickness would help extend this time, but that would impose unfavorable mass penalty. Use of material with high conductivity and heat capacity is recommended in heat sinks. Tokudome et al. reported successful use of ceramic composite materials in test firing of small N_2O/Ethanol propulsion system, which worked as a heat sink. Heat sink was not considered further as standalone cooling method for Zawisza engine.

2.3. Ablation

In process of ablative cooling, the combustion chamber compromises additional layer of material that is placed one the interior side. By that means it is exposed directly to the hot gases of combustion, which melt and vaporize some of the material dissipating heat and reducing heat flux to the outer, structural wall. This method is limited by amount of ablative material that can be sacrificed for heat protection. Historically it has been used in solid rocket motors, but then it was adapted to small liquid rocket engines. More recently advances have been made in evaluation of ablative materials for low cost, lightweight rocket engines with chamber pressure up to *0.9MPa* [8]. Feasibility of application of ablation in long-life liquid rocket engines has been indicated by Fio Rito [9]. His study based on plastics material, which impart another mechanism of thermal protection. When subjected to high heat flux decomposition proceeds with creation of relatively cold gases and solid residue, which forms layer of char on top of ablative material [10]. Char layer strongly reduces heat flux to the original material, such that decomposition occurs to a lesser extent. This effect builds up with time and limits regression of ablative material.

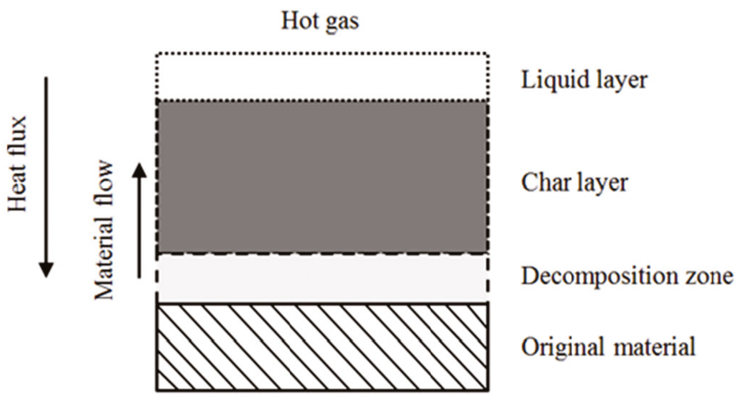


Fig. 3. Schematic representation of plastic material ablation under high heat flux

3. ZAWISZA ROCKET ENGINE DESIGN

Zawisza is pressure-fed N_2O/alcohol liquid rocket engine. As a reference fuel isopropyl alcohol is taken, but ethanol is also considered to be used in final version. Both of these fuels exhibit good cooling capabilities and are readily available, easy to handle, non-toxic and storable in room temperature. Although liquid hydrogen exhibits excellent cooling potentiality it was not taken into the account due to its cryogenic nature.

Table 1. Zawisza rocket engine parameters

Name	Value	Unit	Name	Value	Unit
Chamber Pressure	2	MPa	Thrust (nominal)	4000	N
Mixture Ratio (OF)	3	-	Thrust coefficient	1.41	-
Specific heats ratio	1.23	-	Fuel mass flow	0.48	kg/s
Flame temperature	2500	K	Oxidizer mass flow	1.46	kg/s
Characteristic velocity	1490	m/s	Specific impulse	215	s

As one of the authors presented in his previous work main issue with regenerative cooled N_2O bi-liquid is its relatively high mixture ratio, when compared to traditional LOX engine. This severely limits available fuel mass for use as coolant, reducing its maximum heat capacity, which becomes problematically small. For this reason complex analysis is required in order to determine significance of this fact upon design of regenerative cooling and its feasibility. Moreover, designers decided that for first working prototype of regenerative cooled Zawisza combustion parameters must put less stress upon cooling system, namely combustion will be fuel-rich. Some of parameters are outlined in Table 1 and geometry of the engine is shown on Fig. 4.

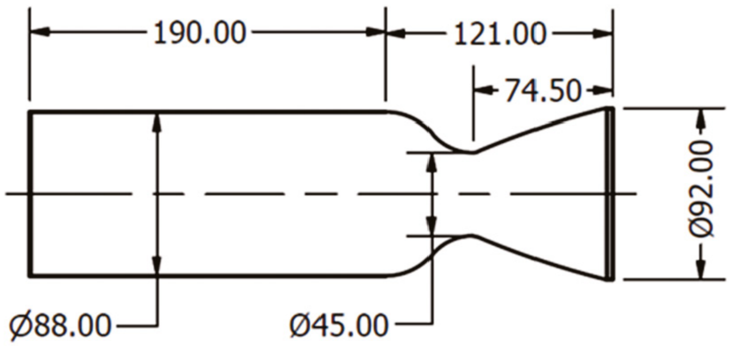


Fig. 4. Geometry of the thrust chamber of Zawisza rocket engine

For cooling configuration it was decided that Zawisza will compromise coaxial shell combustion chamber. In this design there are two shells: inner called the liner, which is made of copper alloy and outer called the jacket, made of steel. These two are separated by helically wrapped copper wire called coil as given in Fig. 5. Essentially this coil has two functions: to provide structural separation and equal distance of two shells along engine axis and restrict coolant flow by passage, which is created between two adjacent wraps or ribs. Number of ribs or "coils" is a design parameter for regenerative cooling of Zawisza.

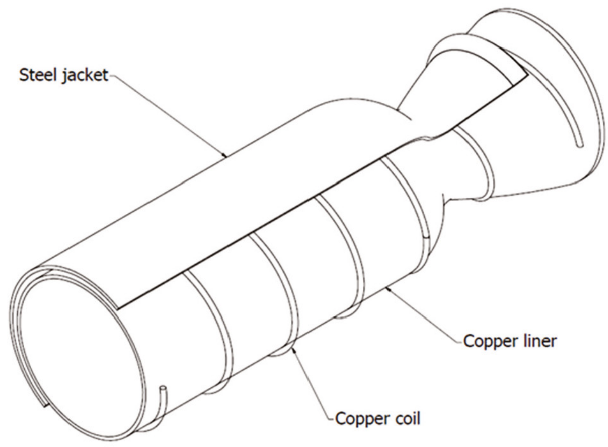


Fig. 5 Structure of Zawisza's regenerative cooled chamber

4. MATHEMATICAL MODEL OF HEAT TRANSFER IN LIQUID ROCKET ENGINE WITH REGENERATIVE COOLING

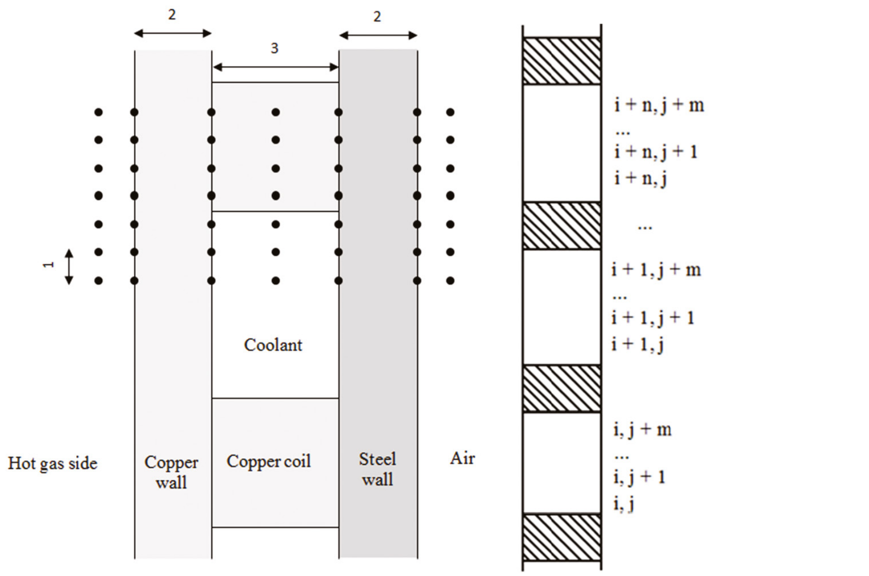
4.1. General information and boundary conditions

To obtain the temperature distribution in the engine and thus determine if such a cooling system would be sufficient, heat transfer calculations have been performed. The model is steady-state, quasi two-dimensional. The calculation domain is a half-section of the engine and cooling jacket which were described above. As thickness of the engine is much smaller than its diameter, Cartesian coordinate system has been applied for simplification.

Inside the thrust chamber there is applied a first-type boundary condition. The assumed temperature is T_{in} = *2500K*. The outer temperature T_{out} = *293K*. Moreover, inlet temperature of the coolant $T_{cool,0}$ = *293K* is treated as a boundary condition in the first passage. In subsequent passages fluid temperature acts as boundary condition as well, after calculating temperature increases caused by the heat flux from inside.

4.2. Geometry and numerical mesh

Geometry of the engine is given in Fig. 4. and Fig. 5. For the needs of numerical simulation the geometry has been discretized as it is presented in Fig. 6a. Along the vertical direction the distances between two points equal 0.001m. Along the horizontal direction there are two points on each wall, one in the cooling channel and two more on both sides as boundary conditions. They represent the temperatures inside the engine and the ambient temperature. Altogether it gives 312x7 = 2184 points mesh.



a) Positioning of mesh

b) Indexing of mesh points in cooling channel through consecutive passages

Fig. 6 Discretization of the engine cooling geometry

4.3. Governing equations

For heat transfer in the walls Fourier's law has been adopted:

$$q = -k\nabla T\left[\frac{W}{m^2}\right] \tag{2)}$$

Convective heat transfer between the wall and cooling channel or thrust chamber has been described by Newton's law:

$$q = h\left(T_w - T_f\right)\left[\frac{W}{m^2}\right] \tag{3}$$

Heat transfer coefficient h_{in} for convection inside the engine has been calculated using Bartz equation [1]:

$$h_{in} = \frac{0.026}{\left(2 \cdot r_{in}\right)^{0.2}}\left(\frac{c_p \mu^{0.2}}{\mathrm{Pr}^{0.6}}\right)\left(\rho v\right)^{0.6}\left(\frac{\rho_{am}}{\rho}\right)\left(\frac{\mu_{am}}{\mu}\right)^{0.2}\left[\frac{W}{m^2 K}\right] \tag{4}$$

Heat transfer coefficient h_{cool} for the cooling channel is estimated using the following formula for the Nusselt number [11]:

$$Nu = \frac{h_{cool} \cdot l}{\lambda} = 0.332 \cdot \mathrm{Re}^{0.5} \cdot \mathrm{Pr}^{0.33}\left[-\right] \tag{5}$$

For calculations of temperature increases in subsequent passages the following reasoning has been carried out. In every segment of the engine's horizontal section there is radially-directed heat flux q_{in} coming out from inside of the engine. In the segments of the mesh where copper wire is located it is equal to the heat flux coming out of the system. But in those segments where coolant is present, there appears a difference of the aforementioned heat fluxes $q_{acu} = q_{in} - q_{out}$ which is connected with the heat accumulated in cooling fluid.

For the first case (a segment without coolant) the following formula can be applied:

$$q_{in} = q_{out} = \frac{T_{in} - T_{out}}{\sum R}\left[\frac{W}{m^2}\right] \tag{6}$$

with thermal resistances summed through the whole wall, whereas for the second case $q_{acu} = q_{in} - q_{out}$ must be considered. As it was stated before, in such cases temperature of the coolant is treated as a boundary condition. Inlet temperature of the cooling fluid is known and the temperatures in a subsequent passage can be calculated from Eq. 7.

$$q_{acu} = \frac{\dot{m} \cdot c_p \cdot \left(T_{i,j} - T_{i-1,j}\right)}{A}\left[\frac{W}{m^2}\right] \tag{7}$$

where *i,j* stands for a specific segment containing cooling fluid and *i-1,j* is the corresponding one in the passage below (Fig. 6b). Here *A* denotes the side area of the helix-shaped channel wall.

T_i and T_{i-1} are also bound by the equation which describes the heat flux coming from inside the engine as a quotient of a known temperature difference and sum of thermal resistances between them (Eq.8). As the temperature of coolant - which amongst others determines value of the heat flux - is changing continuously along the helical channel, arithmetic mean has been used. The formula is analogous to Eq. 6.

$$q_{in} = \frac{T_{in} - 0.5 \cdot \left(T_{i,j} + T_{i-1,j}\right)}{\sum R} \left[\frac{W}{m^2}\right] \tag{8}$$

It is also assumed, that:

$$q_{in} = q_{out} + q_{acu} \left[\frac{W}{m^2}\right] \tag{9}$$

and thus for every calculation point located in cooling channel not being the inlet boundary condition a set of three equations is obtained:

$$\begin{cases} q_{in} = \dfrac{T_{in} - 0.5 \cdot \left(T_{i,j} + T_{i-1,j}\right)}{\sum R} \\ q_{acu} = \dfrac{\dot{m} \cdot c_p \cdot \left(T_{i,j} - T_{i-1,j}\right)}{A} \\ q_{in} = q_{out} + q_{acu} \end{cases} \tag{10}$$

which contains four unknown parameters: q_{in}, q_{acu}, q_{out} and T_i. In this case the initial guess of temperature field has been assumed and first approximation of q_{out} has been calculated. Subsequently, the temperature distribution calculations were carried out basing on heat transfer formulas and Eqs. 10. The procedure has been repeated until convergence occurred.

4.4. Solving the equations

The calculations were performed using a method for numerical solving of heat transfer problems described by Patankar [12]. One dimensional radially-directed heat transfer is assumed for simplification which allows to consider the effect of coolant heating without applying a full 3D model. After discretization, for each point P of the calculation domain the following algebraic equation can be written:

$$a_P \cdot T_P = a_L \cdot T_L + a_R \cdot T_R \,, \tag{11}$$

where T_P, T_L, T_R stand for the temperatures of the point P, left neighbor and right neighbor as follows. The weighting coefficients are defined in a following manner:

$$a_L = \frac{1}{R_L}, \tag{12}$$

$$a_R = \frac{1}{R_R}, \tag{13}$$

$$a_P = a_L + a_R , \tag{14}$$

where R stands for thermal resistance. For convection thermal resistance is defined as an inverse of the convective heat transfer coefficient:

$$R = \frac{1}{h}\left[\frac{m^2K}{W}\right], \tag{15}$$

and for conduction is defined as a quotient of the thickness and thermal conductivity:

$$R = \frac{d}{k}\left[\frac{m^2K}{W}\right], \tag{16}$$

For solving the set of equations Gauss - Seidel iterative method has been adopted .

5. DISCUSSION OF THE RESULTS

Calculations of ratio of heat flux accumulated by fluid to the total heat flux emitted were made to determine optimal number of coils in the cooling jacket. Results of the calculations are presented on Fig. 7. Number of coils which has been chosen for the further calculations is 11.

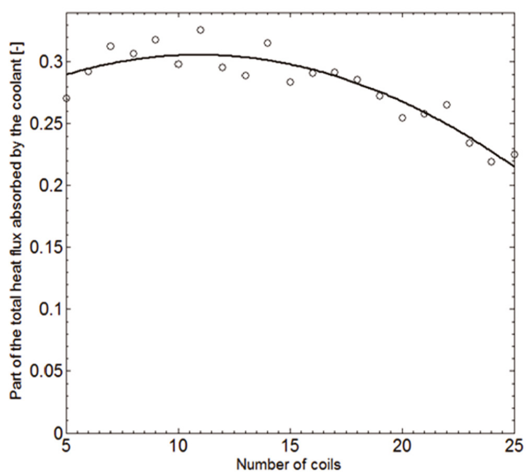
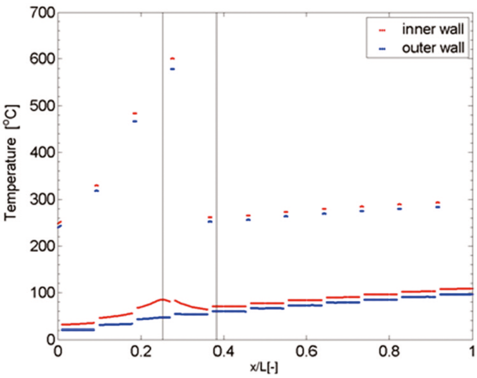


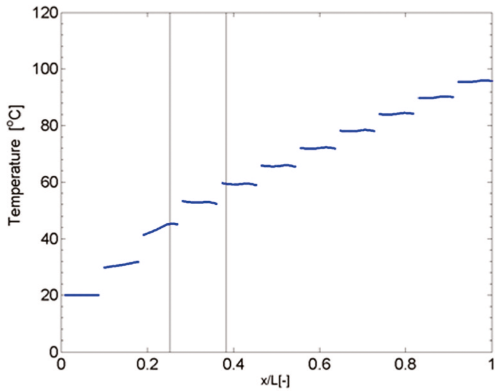
Fig. 7. Dependency of number of coils (ribs) wrapped around copper liner and heat flux absorbed by the coolant

5.1. Temperature distribution

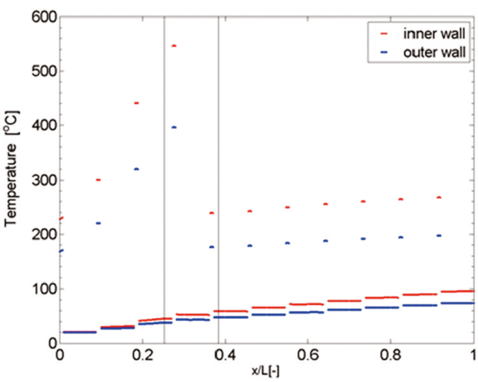
Using the method described in chapter 4 the temperature distribution has been obtained. The results were obtained for the engine walls (Fig. 8a) , for the cooling channels (Fig. 8b) and for the cooling jacket (Fig. 8c). Heat flux along the engine is depicted in (Fig. 8d). As the heat transfer equations were solved in one dimension, conduction in axial direction is not taken into consideration. This may result in getting higher temperatures of engine walls than the temperatures which would actually occur in locations near the coils of cooling jacket. That's because singly located big temperature gradients in axial direction will appear. Two horizontal lines stand for a nozzle throat (left one) and for the nozzle beginning (right one).



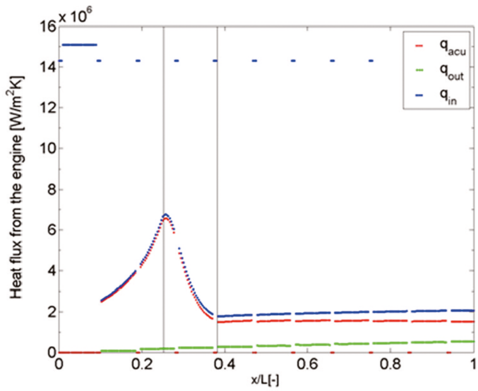
a) Temperature of the copper liner along engine axis



b) Temperature of the coolant along engine axis



c) Temperature of the steel jacket along engine axis

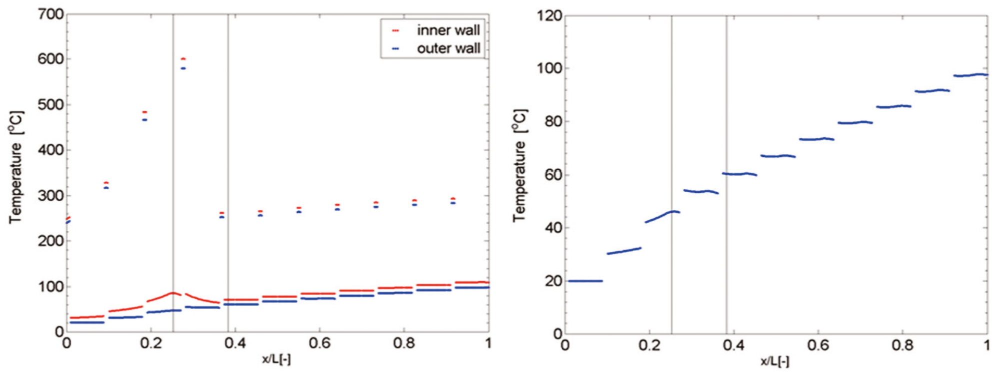


d) Heat flux along engine axis

Fig. 8. Resultant temperature distribution and heat fluxes for 11 coil (ribs)

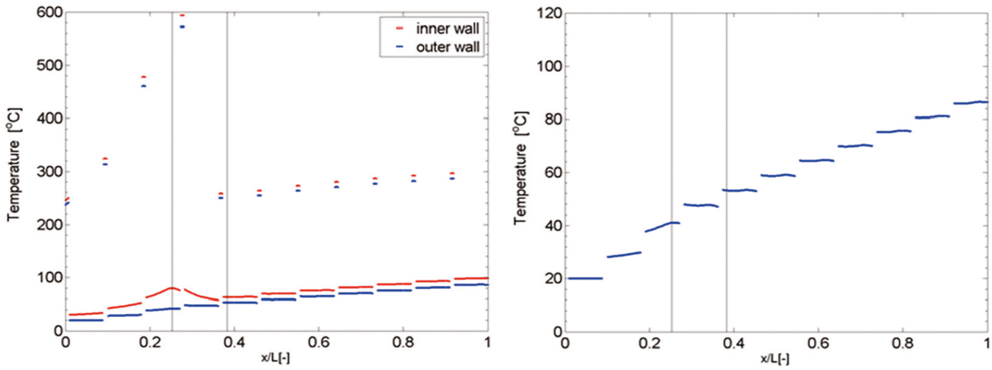
5.2. Comparing results for different propellants' assumptions

An analysis of the combustion process may indicate that other alcohols would act better as fuel. On the other hand their thermophysical properties could be worse. Different OF ratios and mass fractions of water in the fuel may be considered as well. Therefore, analogous calculations have been performed for different propellants' parameters. Figures 10 to 13 show temperatures of engine walls and cooling channel for some single parameters changed.



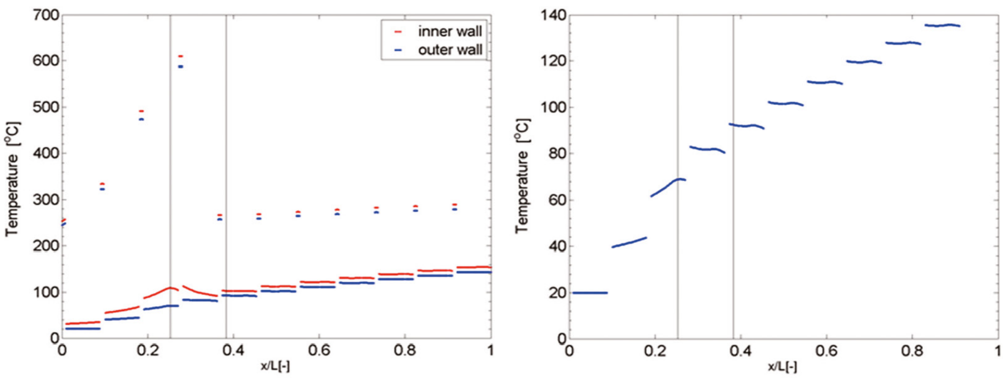
a) Temperature of the copper liner along engine axis b) Temperature of the coolant along engine axis

Fig. 9. Resultant temperature distribution for ethanol as fuel



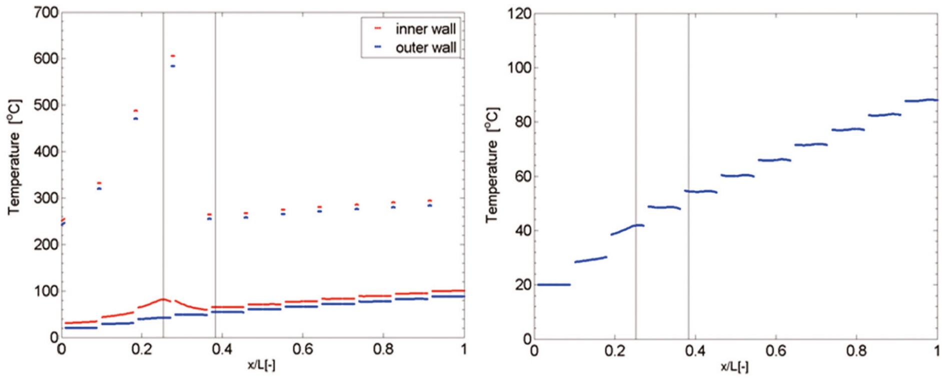
a) Temperature of the copper liner along engine axis b) Temperature of the coolant along engine axis

Fig. 10. Resultant temperature distribution for OF ratio = 2.5 (constant oxidizer flux)



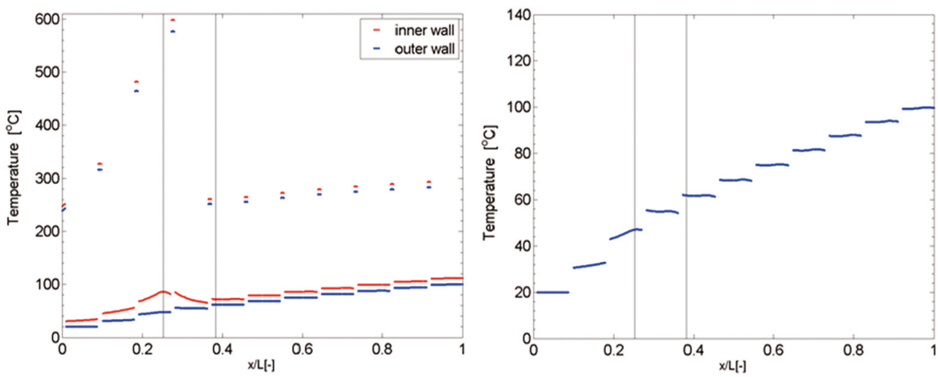
a) Temperature of the copper liner along engine axis b) Temperature of the coolant along engine axis

Fig. 11. Resultant temperature distribution for OF ratio = 4 (constant oxidizer flux)



a)Temperature of the copper liner along engine axis b)Temperature of the coolant along engine axis

Fig. 12. Resultant temperature distribution for water mass fraction in fuel x = 0.2



a)Temperature of the copper liner along engine axis b) Temperature of the coolant along engine axis

Fig. 13. Resultant temperature distribution for water mass fraction in fuel x = 0.05

5.3. Pressure losses

As the temperature of the coolant in upper parts of the cooling jacket may reach about 100 °C, special consideration for the pressure conditions must be given to. Pressure losses along the jacket can be determined by the Darcy-Weisbach equation (Eq. 17). Friction coefficient has been assumed to be equal *0.01* basing on Moody Diagram [13].

$$\Delta p = \lambda \frac{L}{2d_h} \rho v^2 [Pa], \tag{17)}$$

Changing a number of coils *n* in the jacket some of the parameters present in Eq. 17 vary, namely length *L*, hydraulic diameter d_h of the channel and flow velocity *v*. An analysis of the pressure losses as a function of *n* was performed. The results are shown on the Fig. 14. For 11 coils the losses equal 1bar.

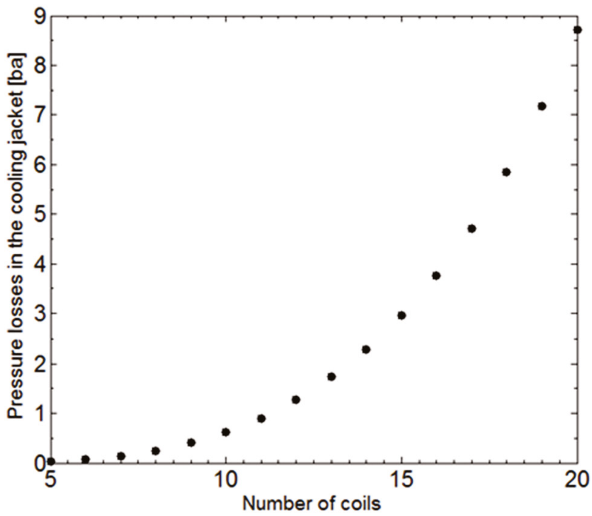


Fig. 14. Dependency of number of coils (ribs) wrapped around copper liner and pressure drop in coolant passage

6. CONCLUSIONS

The results of the present study can be employed to approximate a temperature distribution in a small scale liquid rocket engine with regenerative cooling. It can be determined if a specific variant of cooling conditions would provide suitable conditions for the engine, namely would not overheat the chamber wall or the coolant. According to results of the calculations and referring to the adopted mathematical model, the obtained temperatures are reasonable. For the default parameters of the cooling system temperature of the cooling fluid is not going to exceed *100°C*. For a reference case it's *96°C*. Similarly for the liner, excluding single local peaks around the area without any contact with the coolant. Provided mathematical model and analysis proved current cooling design feasible, despite the fact of using N_2O and high OF ratio.

Calculations of needed inlet pressure of the coolant are to be conducted as a next step of the project. Final pressure has to be not smaller than pressure of boiling in a calculated temperature. More complex simulation accounting for greater amount of phenomena is needed for better approximation of temperature distribution, prediction of potential problems like hot spots and stresses. Finally, an experimental study shall be conducted, which is going to confront the calculations and determine the optimal parameters of combustion such as OF ratio or mass fraction of water in the fuel mixture.

Acknowledgements

The authors would like to express appreciation to the people at AGH Space Systems, especially propulsion team members who worked on Zawisza project. Special thanks also go to the Board and supporters of this association. This work was funded by President of AGH University of Science and Technology prof. Tadeusz Słomka. The authors would also like to acknowledge Vice-President of AGH prof. Anna Siwik for support, without which this publication would not be possible.

REFERENCES

[1] Tokudome S., Yagishita T., Habu H., Shimada T., Daimou Y., 2007, „Experimental Study of an N2O/Ethanol Propulsion System," *43rd AIAA/ASME/SAE/ASEE Joint Propulsion Conference & Exhibit*, 8 - 11 July 2007, Cincinnati, OH, Paper No. 5464

[2] Youngblood, S. H., 2015, "Design and Testing of a Liquid Nitrous Oxide and Ethanol Fueled Rocket Engine," MSc thesis, https://infohost.nmt.edu/~mjh/Pubs/Youngblood2015.pdf

[3] Palacz, T., 2017, "Nitrous Oxide Application for Low-Cost, Low-Thrust Liquid Rocket Engines", *7th European Conference for Aeronautics and Space Sciences (EUCASS),* 3 – 7 July, Milan, Paper No. 474

[4] Sutton, G. P., Biblarz, O., 2001. "Rocket Propulsion Elements." John Wiley & Sons, New York

[5] Bartz, D. R., 1957. "A Simple Equation for Rapid Estimation of Rocket Nozzle Convective Heat Transfer Coefficients" Jet Propulsion

[6] Huzel, D. K., Huang, D. H., 1967. "Design of Liquid Propellant Rocket Engines." NASA, Washington, D.C.

[7] Hill, P. G., Peterson, C. R., 1965. "Mechanics and Thermodynamics of Propulsion", Addison-Wesley Publishing Company, Reading, Massachusetts

[8] Richter, G. P., Smith, T. D., 1995. "Ablative Material Testing for Low-Pressure, Low-Cost Rocket Engines", NASA Technical Memorandum 107041

[9] Fio Rito, R. J., 1995. "Ablatively Cooled Pulse Rocket Engine Design", J. Spacecraft Vol. 2, No. 5

[10] Beecher N., Rosensweig R. E., 1961, "Ablation Mechanisms in Plastics With Inorganic Reinforcement", ARS Journal, Vol. 31, No. 4 (1961), pp. 532-539.

[11] Incropera, F.P., DeWitt, D.P., 2007, "Fundamentals of Heat and Mass Transfer", *6th ,* Wiley

[12] Patankar, S.V., 1980, "Numerical Heat Transfer and Fluid Flow", Hemisphere Publishing Corporation.

[13] Moody, L.F., 1944, "Friction factors for pipe flow", Transactions of the ASME.

WSTĘPNY PROJEKT I ANALIZA CHŁODZENIA REGENERACYJNEGO DLA MAŁEGO CIEKŁEGO SILNIKA RAKIETOWEGO ZASILANEGO N_2O/ALKOHOLEM

Streszczenie

W publikacji przedstawiono koncepcję małego, ciekłego silnika rakietowego zaprojektowanego w AGH Space Systems dla rakiet sondujących. Podczas wstępnej analizy termiczne aspekty różnych sposobów chłodzenia zostały wzięte pod uwagę, oszacowane i opisane. Rozważone zostały możliwe problemy i podejścia projektowe dla chłodzenia ablacyjnego, radiacyjnego oraz regeneracyjnego, a autorzy opisują dostępne rozwiązania. Chłodzenie regeneracyjne jest rozważane w szczególności ze względu na swoją popularność wśród silników zasilanych ciekłym materiałem pędnym, w których paliwo pełniąc role chłodziwa zostaje ogrzane zanim dotrze do komory spalania. W celu oceny rozkładu temperatury, tym samym oceny możliwości zastosowania chłodzenia, został stworzony model matematyczny wymiany ciepła. Unikatowym elementem wspomnianego silnika jest jego utleniacz – podtlenek azotu, który dotychczas był rzadko wykorzystywany. Wybór takiego utleniacza i jego implikacje porównano do typowego silnika zasilanego ciekłym tlenem i wskazano główne różnice, które wpływają na układ chłodzenia. Autorzy porównali również ze sobą różne warianty chłodziwa, w szczególności różne stosunki paliwa i utleniacza, w celu optymalizacji rozkładu temperatury.

Słowa kluczowe: ciekły silnik rakietowy, chłodzenie regeneracyjne, podtlenek azotu, rakieta sondująca.