ISSN: 1231-4005
e-ISSN: 2354-0133
DOI: 10.5604/12314005.1138131

NUMERICAL SIMULATION OF THERMALLY LOADED AIRCRAFT ENGINE TURBINE BLADE COVERED WITH THERMAL BARRIER COATING - TBC

Marek Łazarczyk

Institute of Aviation
Krakowska Av. 110/114, 02-256 Warsaw, Poland
tel.: +48 66049918
e-mail: marek.lazarczyk@yahoo.com

Roman Domański

Warsaw University of Technology, The Faculty of Power and Aeronautical Engineering
Nowowiejska Street 24,00-665 Warsaw, Poland
tel.:+48 22 2347354, fax: +48 22 6257351
e-mail: roman.domanski@itc.pw.edu.pl

Abstract

The objective of this thesis is to present the impact of the turbine blade cooling on blade material temperature as well as to assess advantages and disadvantages of applied cooling method (TBC coating combined with internal cooling). To calculate the conjugated heat transfer analysis generating 3d model and mesh of the blade and its cooling was required. Model mesh was covered with boundary layer in order to properly simulate conditions near the blade walls and obtain accurate results. Calculated blade was put in the canal simulating hot combustion gasses flow. Geometry of model described above was created using Unigraphics NX5 program based on drawings obtained from available literature, and data acquired from the Internet. The discretization was done in commercial pre-processor GAMBIT®. Conjugated heat transfer analysis was conducted in program FLUENT® for two different cases, where the TBC material properties were changed. The goal of this thesis was to obtain temperature fields and distribution in the turbine blade airfoil and to evaluate if applied cooling is sufficient to cool down this thermally loaded part of the engine. Calculated results show that proposed blade heat protection with TBC and internal cooling canal is insufficient during steady state condition, especially on the blade leading and trailing edge. In these two locations, the TBC coating is overheated, and the high temperature level of blade material is unacceptable for materials used in jet engine turbine industry.

Keywords: *aircraft engines, turbine, TBC, coating, blade cooling, CFD*

1. Introduction

This article describes the process of calculating the temperature on turbine blade airfoil surface used in aircraft jet engine and gas turbines applications. Airfoil is covered with Thermal Barrier Coating (TBC) as well as consist single internal cooling channel. The conjugated heat transfer analysis was done to predict areas of highest temperature – “hot spots”. Knowledge of blade material temperature is necessary to introduce proper cooling system to protect airfoil from thermal stress, creep and material damage.

2. Model construction

Geometric model of analyzed blade was made using Unigraphics NX 5 program in accordance to technical specification number 17.41.0114. Blade consists of dovetail, made by two intersecting sketches and airfoil, which is created from five airfoil profiles with different thickness. View of the blade is presented on Fig. 1.

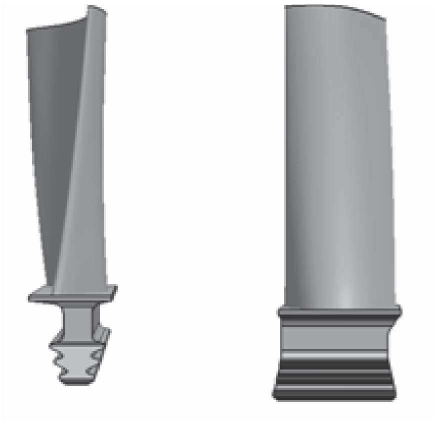

Fig. 1. Complete blade model made in Unigraphics NX 5 program

In order to assess the impact of mentioned cooling method on blade surface temperature, simplified geometry was introduced. It involved the blade, which was made as the extrusion along the straight line with the tilt angle of 20 deg between the bottom and top surface. This allowed keeping the same thickness of the core structure of the blade. Cooling channel was created inside the core and two additional layers of the material ware created by extruding the core horizontal surfaces. Sub layer with 0.1 mm thickness - representing bond coat, and TBC layer with thickness of 0.2 mm.

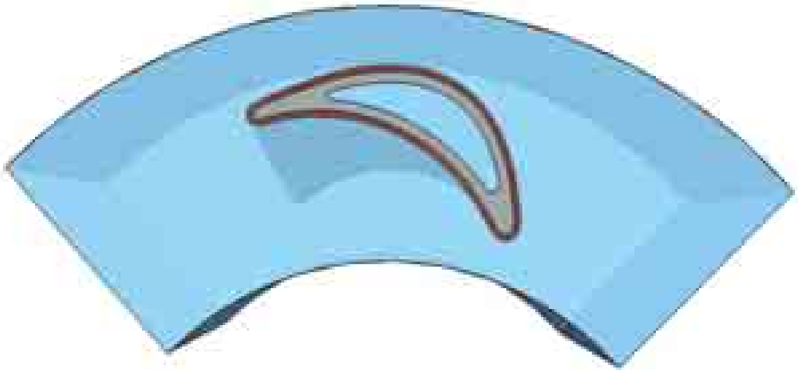

Fig. 2. Blade and surrounding gas canal

To model complex conjugated heat transfer phenomenon blade was placed in combustion canal (which simulated hot combustion gasses flow around the blade). Geometry of this canal was modelled to accurately reassemble the space between two working blades. View of the blade and canal is presented on Fig. 2.

3. Discretization Process

During this study, it was determined that hexagonal mesh is the best option to discretise the model in subject. This type of mesh is continuant to use due to its predictable and repeatable pattern as well as relatively low number of generated elements. Hexagonal mesh usage allows reducing the required computational resources and speeds up the overall computational time. Described model was created with hexagonal mesh and in total had 449000 elements (some of those ware the hex/wedge type). View of the blade's and canal's mesh is presented on Fig. 3.

Surrounding gas channel appeared to be most challenging element to mesh. attempts to cover mentioned element gave unsatisfactory results. In the end element was meshed using tetrahedral elements, which are easy to generate. Using this type of elements increased number of total model elements up to 1.4mln (channel alone had around 1mln of elements).

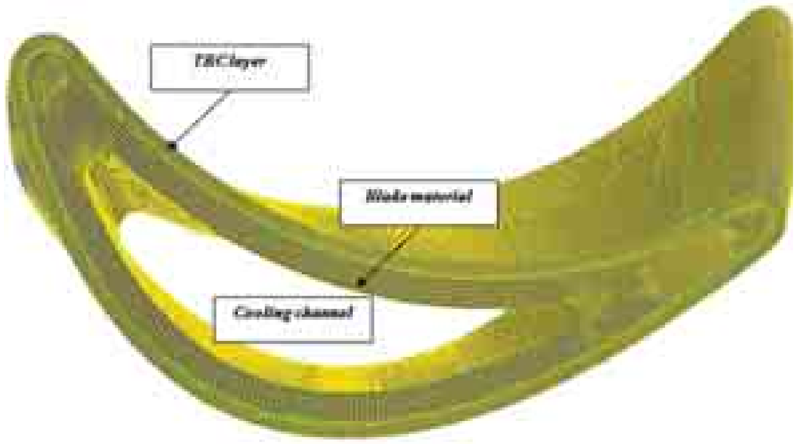

Fig. 3. View of meshed blade model with TBC and bond coat

4. Boundary conditions and material properties

Boundary conditions were created in pre-processor Gambit. Both, inlet to the canal as well as the inlet of the cooling channel ware modelled as velocity inlets. This type boundary condition allowed defining the temperature and velocity of the gases entering the domain. It also allowed defining the parameters used to model the turbulence. Opposite surfaces were defined as the pressure outlets. This boundary condition allowed defining the static pressure on the exhaust.

Following parameters were used to model exhaust gases in the canal.

Pressure 4.5 atm.

$$C_p = 1.3 \text{ [kJ/kg*K]}.$$

Heat transfer coefficient and the density were modelled as variables related to the temperature profile. Parameters are presented in Tab. 1.

Tab. 1. Exhaust gasses parameters

T [K]	300	600	1500
P [kg/m^3]	3.74	2.5	1.6
λ [W/m*K]	0.02	0.045	0.06

Gas used in the cooling channel was defined as dry air taken from the last stage of the compressor, with following parameters:

$$C_p = 1.3 \text{ [kJ/kg*K]}$$

Heat transfer coefficient and the density were modelled as variables related to the temperature profile. Parameters are presented in Tab. 2.

Tab. 2. Cooling gasses parameters

T [K]	300	600	1500
ρ [kg/m^3]	3.5	2.32	1.4
λ [W/m*K]	0.022	0.04	0.07

TBC layer was defined as ZRO_2+$7Y_2O_3$ with following parameters:

$$\rho \text{ [kg/m}^3\text{]} = 5170.$$

Specific heat and heat transfer coefficient were modelled as variables related to the temperature profile. Parameters are presented in Tab. 3.

Tab. 3. TBC parameters

T [K]	300	600	1500
Cp [kJ/kgK]	593	621	706
λ [W/m*K]	0.261	0.522	1.305

Bond coat layer was defined as NiCrAlY with following parameters:

$$\rho\ [kg/m^3] = 7000.$$

Specific heat and heat transfer coefficient were modelled as variables related to the temperature profile. Parameters are presented in Tab. 3.

Tab. 4. Bond coat parameters

T [K]	300	600	1500
Cp [kJ/kgK]	638	669	763
λ [W/m*K]	4.767	6.204	10.515

Steel has been chosen as a material of the airfoil. Properties of the material were taken from FLUENT database:

$$\rho\ [kg/m^3] = 8030,$$

$$\lambda\ [W/m*K] = 16.27,$$

$$Cp\ [kJ/kgK] = 502.4.$$

5. Results

A number of results of CFD analyses have been obtained. After verification of the results, the most significant and important (according to the author) have been published in this work. One of the biggest problem occurring during calculations turned out to be the reverse flow (that occurred on the pressure outlet). Reverse flow on the aft side of the airfoil was affecting pressure distribution, which was resulting in changed velocity and temperature values. To address this problem a computational domain was extended, so the reverse flow did not reach the airfoil.

5.1. Case 1 – Nominal TBC parameters

Boundary conditions on the inlet of computational domain were set as temperature (1500K) and gas velocity (400 m/s). BC on the outlet of the domain was set as pressure (4.5 atm). Calculated Mach number (on the airfoil) for these conditions was equal to 0.6.

Boundary conditions on the inlet of cooling canal were set as temperature (600K) and velocity (100 m/s). On the outlet of cooling passage BC was set as pressure (6.5 atm).

Temperature plots of the blade in steady state conditions are shown in Fig. 4 and Fig. 5.

Temperature distribution on the airfoil is non-linear. Max. temperatures (hot spots) are observed on the leading edge and trailing edge of the blade. Cooling of the airfoil in these locations is ineffective. Cooling drops leading edge temperature only by 60 K with respect to the temperature of surrounding hot combustion gasses. Trailing edge temperature decreases only by 20 K. Cooling effect is well seen in others areas of airfoil. Temperature of airfoil top surface decreases by 250K while there is a drop of 210K on the bottom surface. This temperature drop is mainly a result of internal passage cooling.

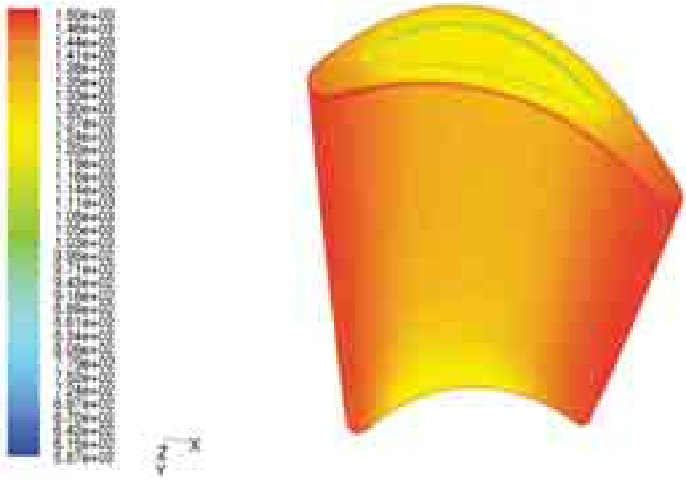

Fig. 4. View of the blade total temperature [K]

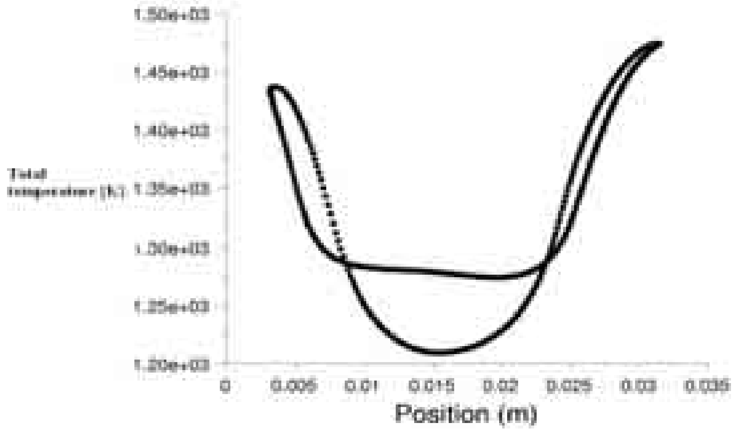

Fig. 5. Blade surface temperature in a cross section

5.2. Case 2 - TBC with lower thermal conductivity

The next step was to examine another TBC coating with the same parameters of flow around the airfoil. New TBC material features lower thermal conductivity (30% lower with respect to ZRO_2+$7Y_2O_3$). Temperature plots of the blade with new TBC coating in steady state conditions are shown in Fig. 6 and 7.

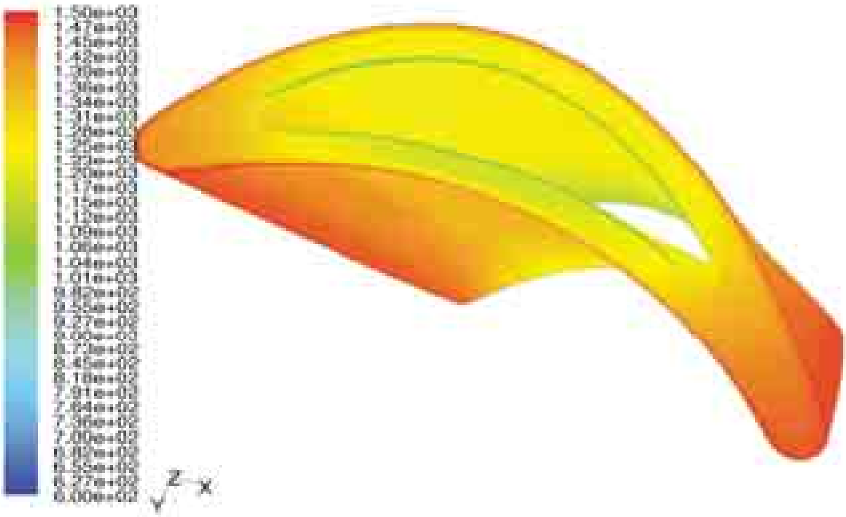

Fig. 6. View of the blade total temperature [K]

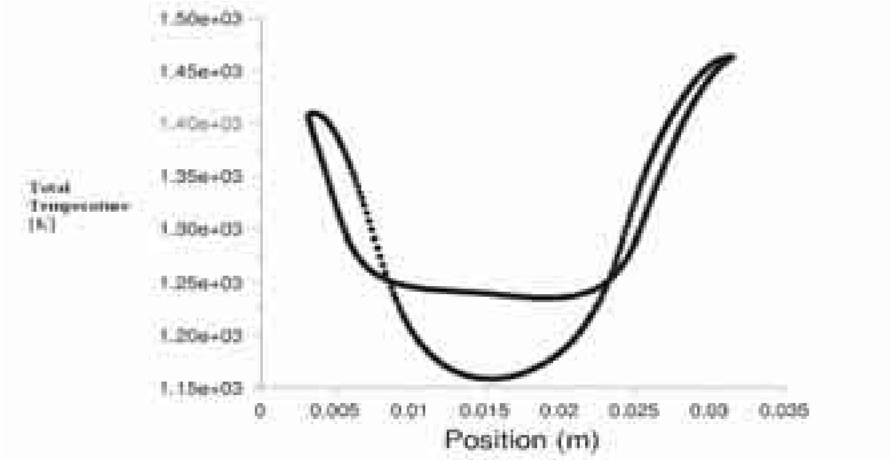

Fig. 7. Blade surface temperature in a cross section

Temperature distribution on the airfoil did not change due to TBC properties change and is non-linear. Max. temperatures (hot spots) are observed on the leading edge and trailing edge of the blade. Cooling drops leading edge temperature only by 80 K With respect to the temperature of surrounding hot combustion gasses. Trailing edge temperature decreases only by 40 K. Fig. 8 shows blade surface temperature comparison between two calculated cases (nominal TBC and TBC with lower thermal conductivity). The biggest temperature drop can be observed in the middle section of the airfoil surface (350K). Bottom surface of the airfoil shows temperature drop of 70K, with respect to the previous TBC coating. Temperature decrease on the leading edge and trailing edge is minor and it is equal to 20K in both cases. This fact confirms low impact of thermal barrier coatings on blade temperature level in steady state conditions.

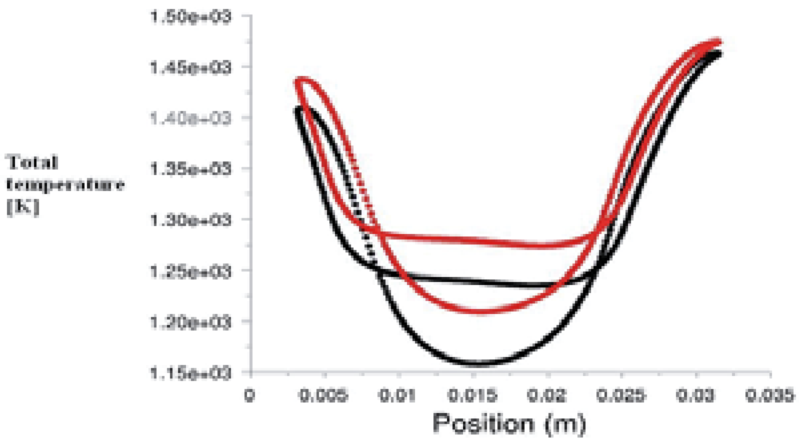

Fig. 8. Blade surface temperature in a cross section – temperature comparison: red - standard TBC, black - TBC with decreased thermal conductivity.

6. Conclusions

This paper presents results of simulation of heat transfer process in the airfoil being used in jet engine turbine. To run this kind of simulations, experience, proper physical model as well as number of simplifications are needed. All these may cause in final result inaccuracy. For that reason, the code validation with experiment is crucial in this kind of research.

Presented study shows that heat protection of TBC and inner cooling canal are insufficient for the turbine blade application, especially on the blade leading and trailing edges. During steady state condition TBC in these areas is overheated. Temperature drop in hot spots ranges from 20K to 70K (comparing to gas temperature). In the middle section of the airfoil we can observe higher temperature decrease (even 350K), which is caused by inner cooling passage. High temperature differences occurring on the outer outline of the airfoil will be a driver for high thermal stresses occurrence. That high temperature level on the leading and trailing edges is unacceptable for temperature bands of materials used in jet engine turbine industry.

To improve cooling performance and decrease the metal temperature of the blade, it would be suggested to introduce additional cooling of leading and trailing edges.

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