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NUMERICAL SIMULATION OF HEAT LOADED AIRCRAFT ENGINE TURBINE BLADE – INTERNAL COOLING

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Abstract

The objective of this thesis was to show present the impact of the turbine blade cooling on blade material temperatures well as familiarize with processes and methods of numerical simulation by designing airfoil internal cooling system. The model of airfoil was created based on drawing print nr 17.41.0114 of engine Tumański R-11F2S-300.

Geometry of all models described above is created using SIMENS NX4 program based on drawings obtained from available literature, and data acquired from the Internet. The discretization into a structural finite volume grid took place in commercial pre-processor GAMBIT® (GAMBIT and FLUENT – commercial CFD codes from Ansys Inc.).

Conjugated heat transfer analysis was conducted in program FLUENT® for four different cases, where the blade 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 would be able to cool down this thermally loaded part. Calculated results show that proposed blade heat protection method is insufficient during steady state condition, mainly on the blade leading and trailing edge. In these two locations, the blade is overheated, and the high temperature level of applied material is unacceptable for used in jet engine turbine industry.

Keywords: *CFD, internal cooling systems, thermal protection*

1. Introduction - process of geometry creation

The objective in this paper is three-dimensional heat-flow analysis of turbine blade based on drawing print nr 17.41.0114 of engine Tumański R-11F2S-300. According to acquired specification the 3D model was created using software Unigraphic NX4. Due to complex geometry, only the part of turbine blade was analyzed. Only airfoil, the platform and dovetail were omitted.

Fig. 1. Blade geometry created in program Unigraphics

Geometry simplified this way was exported in parasolid format. Meshing process took place in program Gambit that was unable to create a mesh with appropriate quality on airfoil geometry, which is rotated and less thick at the end. Therefore in the next iteration the same airfoil thickness along the airfoil height were used. For that geometry the four cooling channels was created.

In part of channel at the leading edge area in each of fourteen sections were drilled three cooling holes (Fig. 2). This will allow cooling air to decrease leading edge temperature. Creating more holes in more sections would increase the efficiency of cooling but also increase the complexity of the model, which would have a significant impact on analysis time and in worst-case result in creating mesh to complex for available computers to handle.

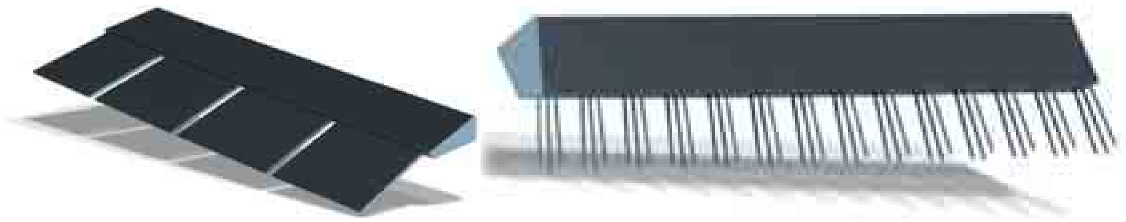

Fig. 2. On the left - cooling channels created in trailing edge. On the right cooling channel in leading edge area

Next two channels were created without any cooling holes, which would help to create cooling film and decrease temperature of concave and convex side of airfoil. In channel placed near trailing edge instead of cooling holes four straight holes were created (Fig. 3). This option was chosen to decrease number of elements and size of mesh.

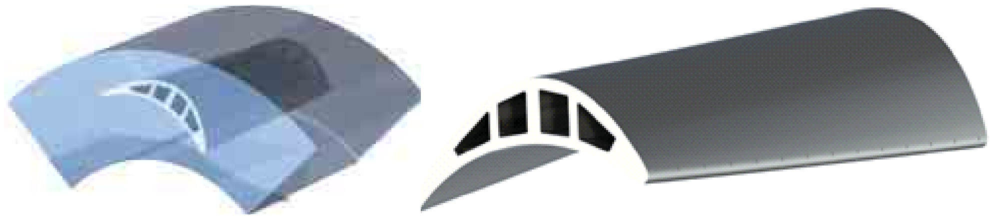

Fig. 3. On the left - placement of airfoil in the hot gas channel, right - airfoil with cooling geometry

In final stage of creating geometry, the virtual gas channel was created. The purpose of this volume was to simulate the hot gases flowing around the prepared airfoil. Ideal volume geometry should change like the airfoil, and in each point, it is perpendicular to the airfoil surface. After consulting, it was also decided for the gas channel to be perpendicular only in one section.

Figure 3 represents the placement of airfoil in the gas channel. This position is not accurate, and shows the effect of incorrect airflow distribution on convex and concave side of airfoil. The results will be shown in chapter 3.

2. Uploading geometry, model discretization and setting boundary conditions

Final geometry was exported into Gambit program in which all volumes underwent discretization process. The best type of elements is hexagonal element however do to complex geometrical structure model was meshed using tetrahybrid elements. Analyzed model contained 2.219.349 elements, airfoil alone had 1.952.276 elements (Fig. 4).

In last phase of discretization process boundary conditions must be set. Each face and volume of meshed model needs to have properties added. All the volumes that created airfoil had SOLID type, and main gas channel as well as cooling channels had FLUID Continuum type set. For main channel inlet, the VELOCITY_INLET type was set, and for all outlets (from main gas channel and cooling channel), PRESSURE_OUTLET was set. After setting boundary conditions mesh was exported using FLUENT 5/6 solver. Next important step was setting materials, their parameters and adding their properties to correct volume groups. Hot and cooling gas parameters were provided in Tab. 1 and 2, respectively. Two types of airfoil material were analyzed (Tab. 3).

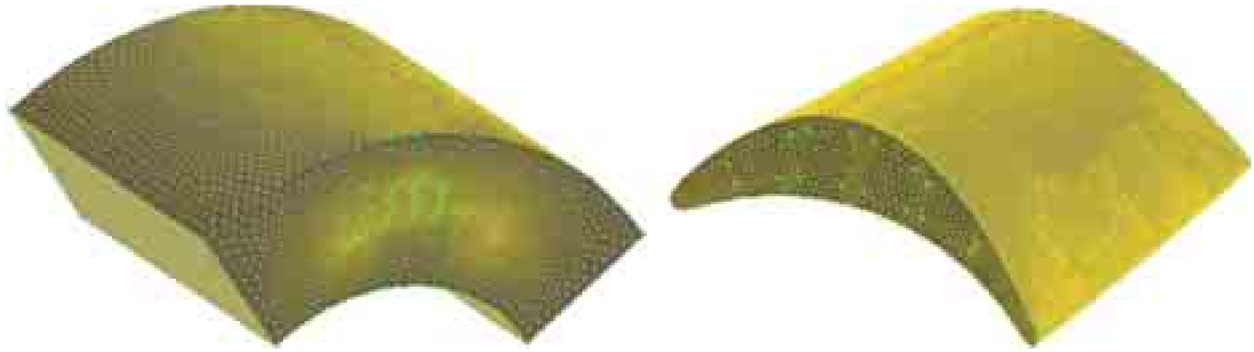

Fig. 4. Analyzed geometry after discretization process

Tab. 1. Hot gas parameters

$C_p = 1300 \left[\frac{\mathrm{J}}{\mathrm{kg} \cdot \mathrm{K}}\right]$	Temperature [K]	300	600	1500
	Density [kg/m^3]	3.74	2.5	1.6
	Thermal conductivity [W/(m·K)]	0.02	0.045	0.06

Tab. 2. Cooling gas parameters

$C_p = 1150 \left[\frac{\mathrm{J}}{\mathrm{kg} \cdot \mathrm{K}}\right]$	Temperature [K]	300	600	1500
	Density [kg/m^3]	3.5	2.32	1.4
	Thermal conductivity [W/(m·K)]	0.022	0.04	0.07

Tab. 3. Blade material parameters

Titan	
Density [kg/m^3]	4850
Specific heat [J/(kg·K)]	544.25
Thermal conductivity [W/(m·K)]	7.44

Steel	
Density [kg/m^3]	8030
Specific heat [J/(kg·K)]	502.48
Thermal conductivity [W/(m·K)]	16.27

Afterwards the boundary conditions were set – pressure velocity turbulence parameters and temperature on intake to and outflow. Due to larger number of varying parameters all materials setting are presented in next chapter. Before initialization the convergence criteria and under-relaxation factors was decreased in order to acquire more accurate results.

3. Fluent results

Analysis results of steel (Case A and B) and titanium (Case C and D) airfoil for two hot gas temperatures 1200K and 1500K are presented below.

Tab. 4. Boundary conditions – case A

Airfoil material	Steel		
Hot gas temperature [K]	1200	Cooling gas temperature [K]	600
Fumes pressure [MPa]	0.45	Cooling gas pressure [MPa]	0.6
Fumes velocity at inlet [m/s]	200	Cooling gas temperature at inlets [m/s]	150

Tab. 5. Boundary conditions – case B

Airfoil material	Steel		
Hot gas temperature [K]	1500	Cooling gas temperature [K]	600
Fumes pressure [MPa]	0.45	Cooling gas pressure [MPa]	0.6
Fumes velocity at inlet [m/s]	200	Cooling gas temperature at inlets [m/s]	150

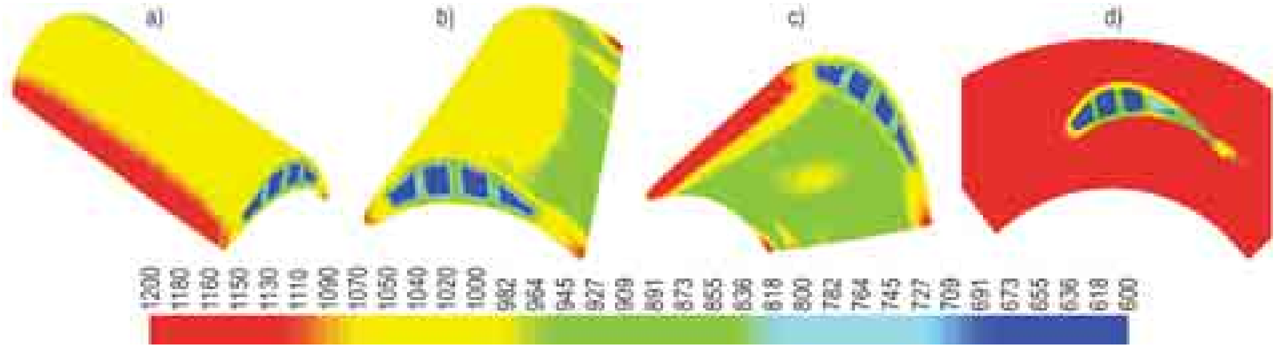

Fig. 5. Total temperature [K] on: leading edge and convex (a), trailing edge (b), concave side (c), cross-section through the middle of airfoil (d) in case A

a) b)

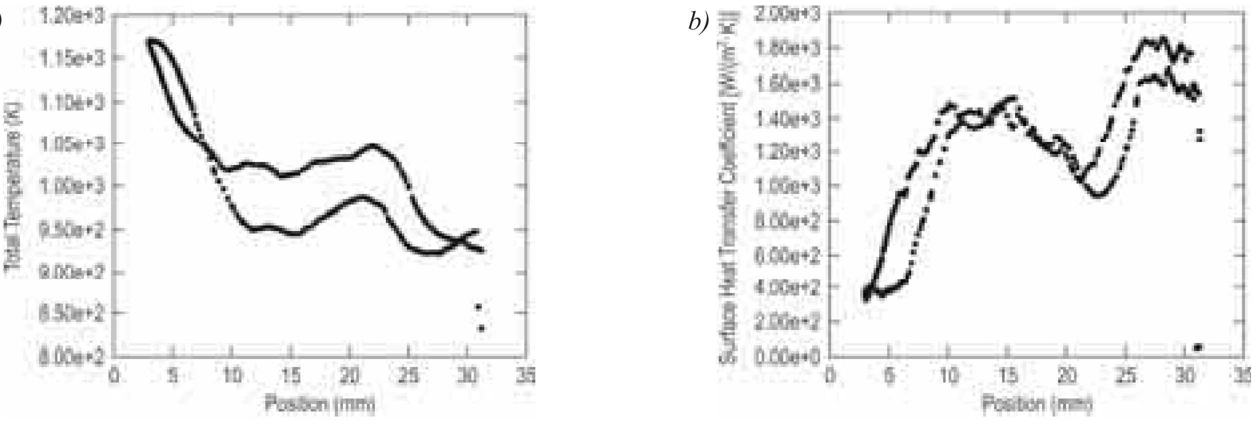

Fig. 6. Total temperature (a) and heat transfer coefficient (b) on airfoil concave and convex surface – case A

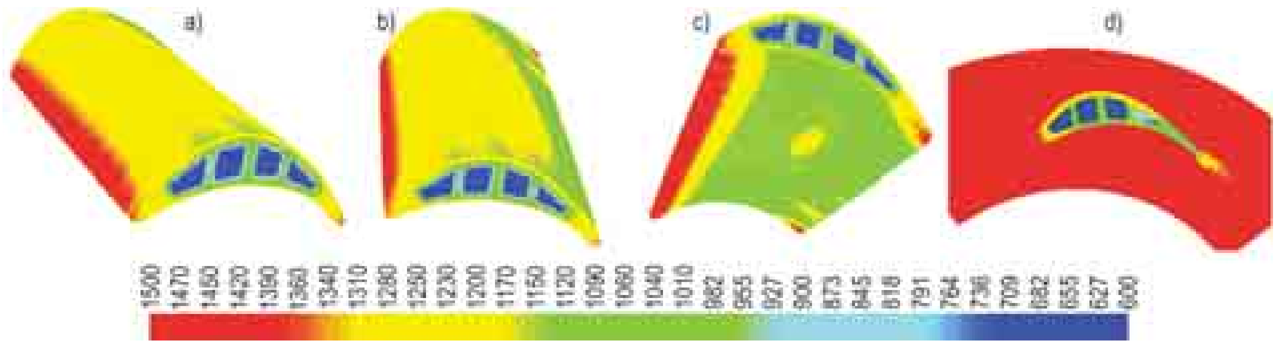

Fig. 7. Total temperature [K] on: leading edge and convex (a), trailing edge (b), concave side (c), cross-section through the middle of airfoil (d) in case B

a) b)

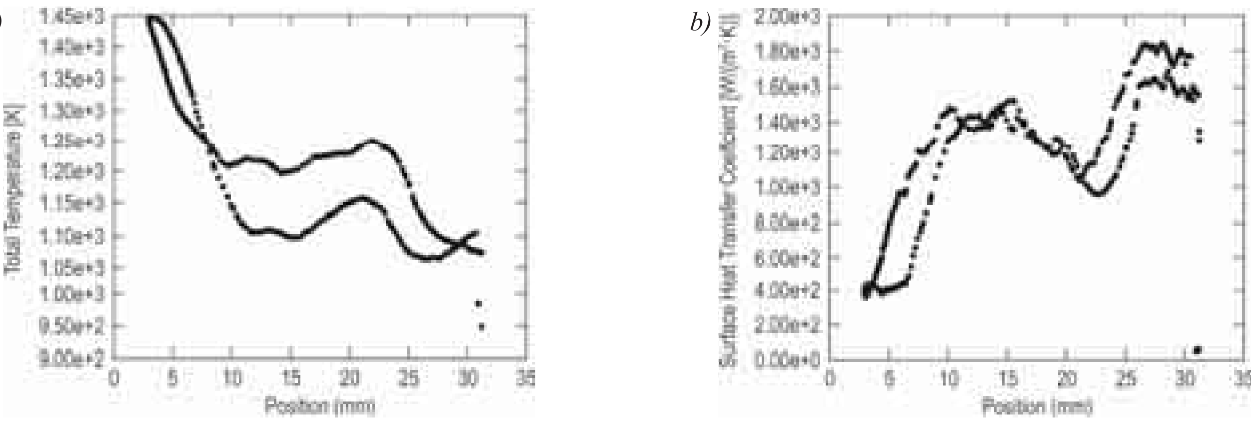

Fig. 8. Total temperature (a) and heat transfer coefficient (b) on airfoil concave and convex surface – case B

Tab. 6. Boundary conditions – case C

Airfoil material	Titan		
Hot gas temperature [K]	1200	Cooling gas temperature [K]	600
Fumes pressure [MPa]	0.45	Cooling gas pressure [MPa]	0.6
Fumes velocity at inlet [m/s]	200	Cooling gas temperature at inlets [m/s]	150

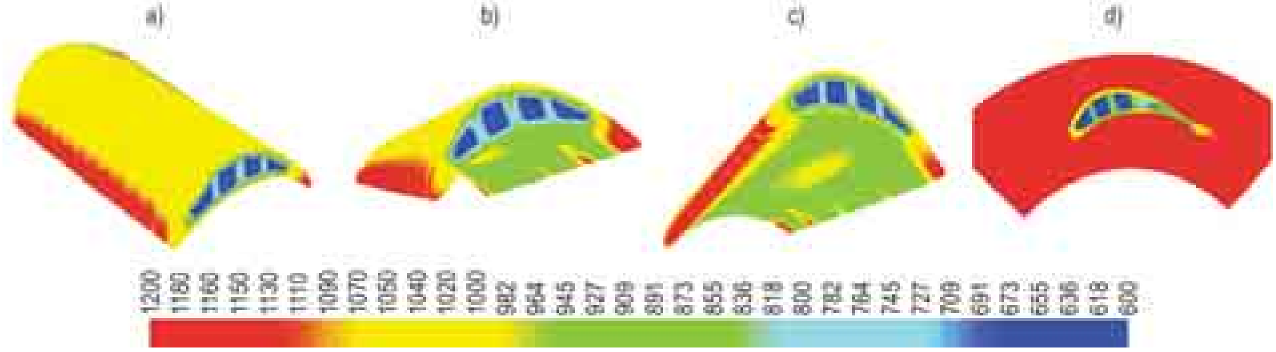

Fig. 9. Total temperature [K] on: leading edge and convex (a), trailing edge (b), concave side (c), cross-section through the middle of airfoil (d) in case C

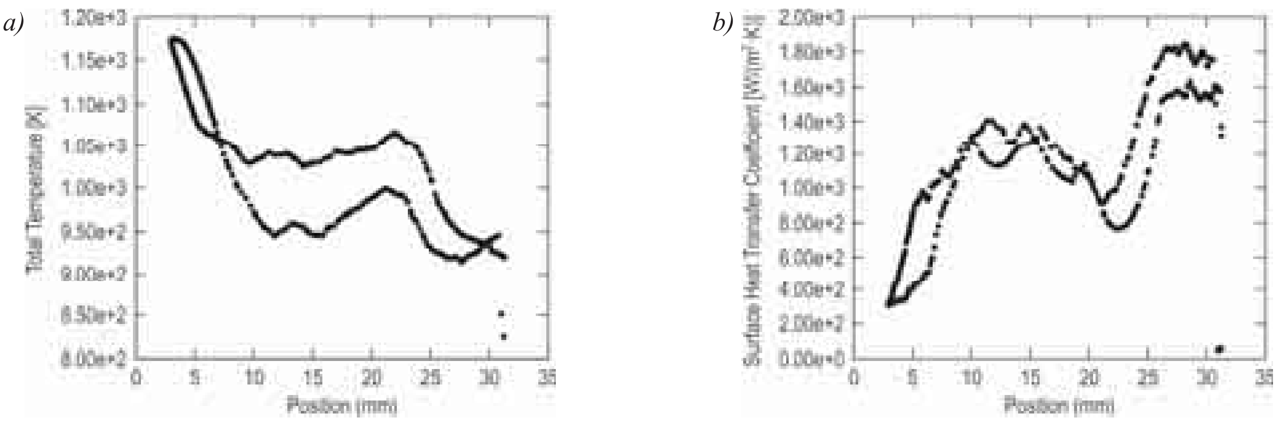

Fig. 10. Total temperature (a) and heat transfer coefficient (b) on airfoil concave and convex surface – case C

Tab. 7. Boundary conditions – case D

Airfoil material	Titan		
Hot gas temperature [K]	1500	Cooling gas temperature [K]	600
Fumes pressure [MPa]	0.45	Cooling gas pressure [MPa]	0.6
Fumes velocity at inlet [m/s]	200	Cooling gas temperature at inlets [m/s]	150

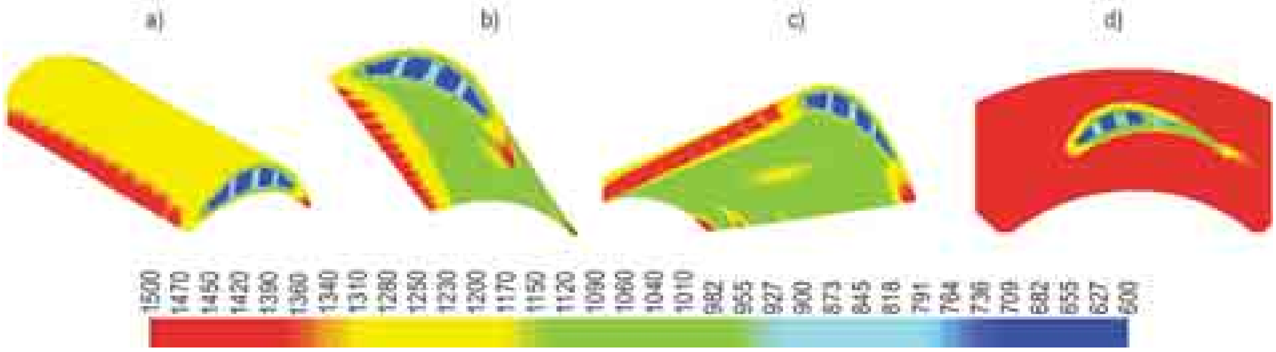

Fig. 11. Total temperature [K] on: leading edge and convex (a), trailing edge (b), concave side (c), cross-section through the middle of airfoil (d) in case D

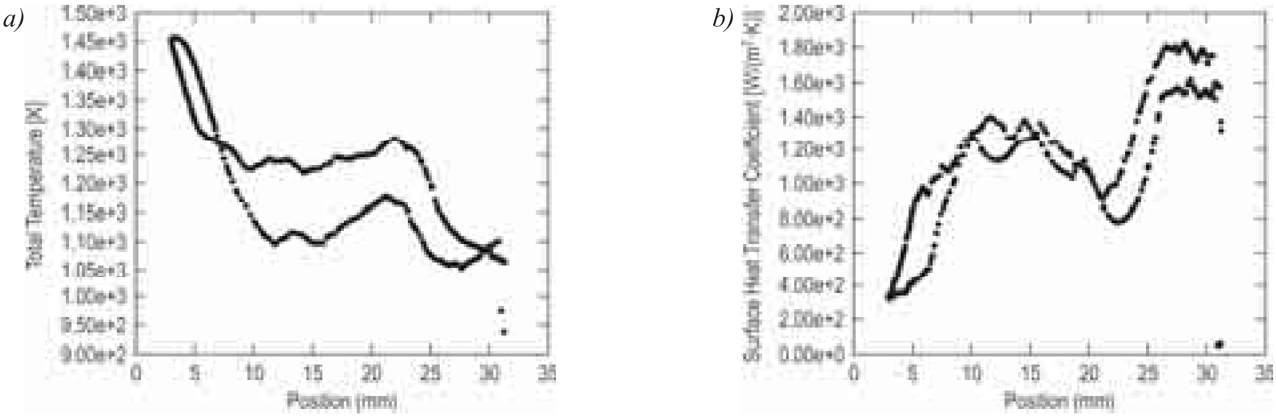

Fig. 12. Total temperature (a) and heat transfer coefficient (b) on airfoil concave and convex surface – case C

4. Conclusion

Technological progress in development of aircraft engines allows achieving higher temperatures before first stage of turbine, which leads to higher efficiency. In this paper, preliminary design of internal cooling method was analyzed. Simplifications and several assumptions that decreased the complexity of the design. This allows for better understanding processes that are taking place. Designing the internal cooling systems is very problematic mainly due to complex internal geometry and material properties, which are not available for general public.

Accuracy and quality of obtained result is impacted by limitations of available computing power. In order to avoid expensive experimental studies simplified models are generated. This method minimizes costs and allows acquiring results in short period of time.

However simplifications generate imperfections, but without some level of simplification analysis would be impossible. One of most crucial elements, which have significant impact on results, is mentioned in chapter 2 shape of the main channel and position of aero itself. The current position of aero forces hot gases to push large amount of cooling air flowing from leading edge downstream. As a result, convex side of aero has insufficient heat protection. The leading edge cooling film is also overheated due to not enough quantity of cooling holes drilled in leading edge. In this situation, it would be recommended to increase the number of holes, or their diameter in order to obtain higher mass flow rate. Increasing the diameter could also affect the endurance of aero, so the proposed geometry with internal cooling system should be analyzed from point of thermal strain, stress.

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