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Numerical Modeling of Turbine Blade Cooling for Aero Engine Applications with the Use of Surrogate Models

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The optimization of the thermodynamic cycles of aero engines has always been in the main targets of engineering efforts for environmental and economic reasons. The aero engine thermodynamic cycle is significantly affected by the selection of the cycle maximum temperature, which at the same time, should be high enough to achieve increased cycle efficiency but also be always kept within turbine blades material temperature limits to ensure turbine blades endurance and integrity. For these reasons, cooling techniques are used where a part of the compressor discharge air is usually used as a cooling medium for the turbine blades. Since this part of air does not participate in the heat addition process inside the combustion chamber its amount should be carefully estimated in order to simultaneously provide cooling air protection for the turbine blades and avoid loss of potential turbine work. Towards this direction, numerical tools such as the one presented in this work are developed. The present work is focused on the development of a surrogate model network for the calculation of turbine blade cooling for aero engine applications. In the analysis the turbine blade is modelled as a heat exchange unit composed of a number of interconnected sub-units. In each sub-unit the effect of turbine blade local geometrical features, such as inner channels length and hydraulic diameter, on heat transfer and pressure losses was incorporated through the use of literature based correlations. The investigations were focused on typical turbine blade conditions for recuperative aero engine applications as presented in Salpingidou et al. (2017). For the cooling mass flow calculations the suggestions of Young and Wilcock (2002a and 2002b) and Wilcock et al. (2005) were taken into consideration. Furthermore, after the development of the surrogate model network, the effect of incorporating heat transfer enhancers on inner flow turbine blade geometry was numerically assessed for two heat transfer augmentation surfaces, such as the ones presented in the work of Alam and Kim (2018), using experimental data available in literature describing their heat transfer and pressure loss characteristics targeting the identification of configurations with promising effect on the aero engine.

1. Introduction

Gas turbines are used in various applications related with either with power propulsion or power generation. More specifically, gas turbines for aero engine applications produce significant pollutant emissions i.e. CO_2, NO_x and increased fuel consumption due to their operational thermodynamic cycle gas turbines. As a result, a large number of research activities are performed targeting the optimization of aero engines performance for both environmental and economic reasons. Currently, the performance of even the most advanced gas turbines for aero engine applications operate near the 40% threshold by also keeping in mind that the performance of the individual aero engine components (e.g. compressors, turbines) operate already with a very high efficiency. As it is evident, the remaining optimization margin is limited especially if this was to be achieved by the further increase of the component's efficiency. In this respect, the further refinement of the

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aero engine thermodynamic cycle can provide additional optimization potential. The aero engine operational thermodynamic cycle is strongly affected by the cycle temperature ratio of maximum to minimum temperatures. Since the cycle minimum temperature is determined by the aero engine ambient conditions, in order to increase the temperature ratio it is necessary to apply the maximum possible temperature on the aero engine cycle. The higher the maximum cycle temperature is applied, the higher the cycle overall efficiency potential is. However, this maximum temperature should be kept always within the limits set by the turbine blades materials in order to ensure the endurance and structural integrity of turbine blades which are encountering extremely high temperatures inside the aero engines. In order to further increase the maximum cycle temperature and simultaneously protect the turbine blades from excessive temperatures turbine blade cooling is applied. Typically turbine blade cooling is achieved by the utilization of a part of the compressor discharge air which is used as a cooling medium for the turbine blades. Unfortunately, since the cooling air by passes the combustion chamber its amount does not contribute to turbine work. Thus, the applied cooling air amount should be carefully calculated so as to provide sufficient turbine blades protection and also avoid unnecessary loss turbine work. Towards this direction, the development of numerical tools can be of great practical value, especially if time efficient calculations are targeted. The present work is focused on the development of such a numerical tool, in the form of a surrogate model, for the modelling of turbine blade cooling for aero engine applications. In this respect, the turbine blade is modelled as a heat exchange unit composed of a number of interconnected sub-units. Each sub-unit incorporates the effect of turbine blade local geometrical features, such as inner channels length and diameter, on heat transfer and pressure losses was incorporated through the use of literature based correlations. For the development of the surrogate model CFD computations on a detailed turbine blade cooling model were performed.

2. The development of the numerical surrogate tool

The first step in the development of the numerical surrogate tool was to select a typical, yet representative, turbine blade geometry. The turbine blade geometry which was selected for this work was based on a similar turbine blade geometry to the one presented in the work of Ho et al. (2014) in which small adaptations were included and in which only internal blade cooling was applied. Typical view of the selected turbine blade geometry (rotor) is presented in Figure 1 and of the computational grid in Figure 2. This geometry was the reference geometry for the numerical tool development. Since the main target was to show the realizability of this approach, the CFD computations were performed on a stationary frame of motion for simplicity.

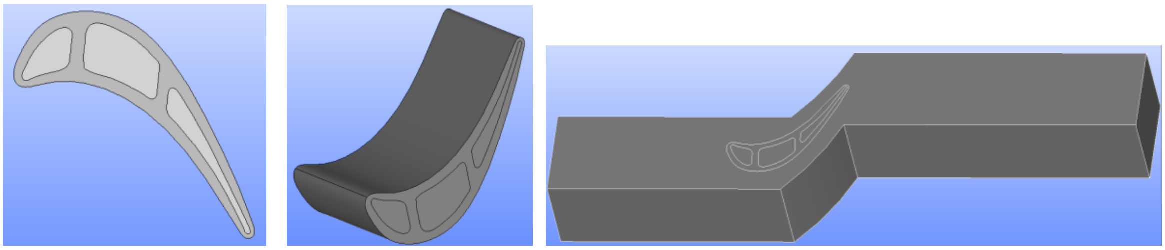


Figure 1: The turbine blade geometry and the overall geometry

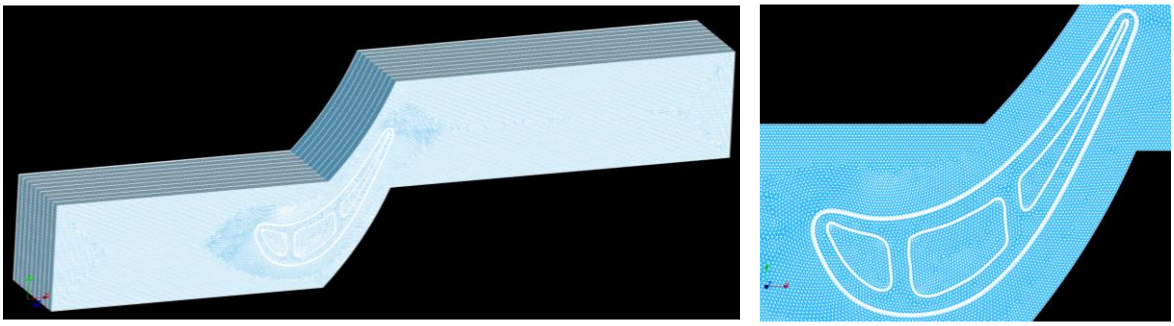


Figure 2: The turbine blade computational grid – perspective and enlarged view

The computational grid was constructed in the freely accessible Salome (2018) platform and the CFD computations were performed in ANSYS/CFX CFD software following the suggestions of ANSYS (2019). The computational grid consists of different domains which are: the inner flow domains, the outer flow domain and the solid walls domain, as shown in Figure 3. The first one covers the region occupied by the cooling air, the second one the region occupied by the hot gas around the turbine blade and the latter the region occupied by the turbine blade walls. These domains are linked together in their interconnecting region so that heat is transferred from the hot gas through the turbine blade walls to the cooling air. The computational grid consisted of 433200 computational nodes. In order to increase the accuracy of the CFD computations special care was provided to refine the computational grid near the turbine blade walls, as shown in Figure 4. For the turbulence modeling the Shear Stress Transport (SST) model of Menter (1994) was used.

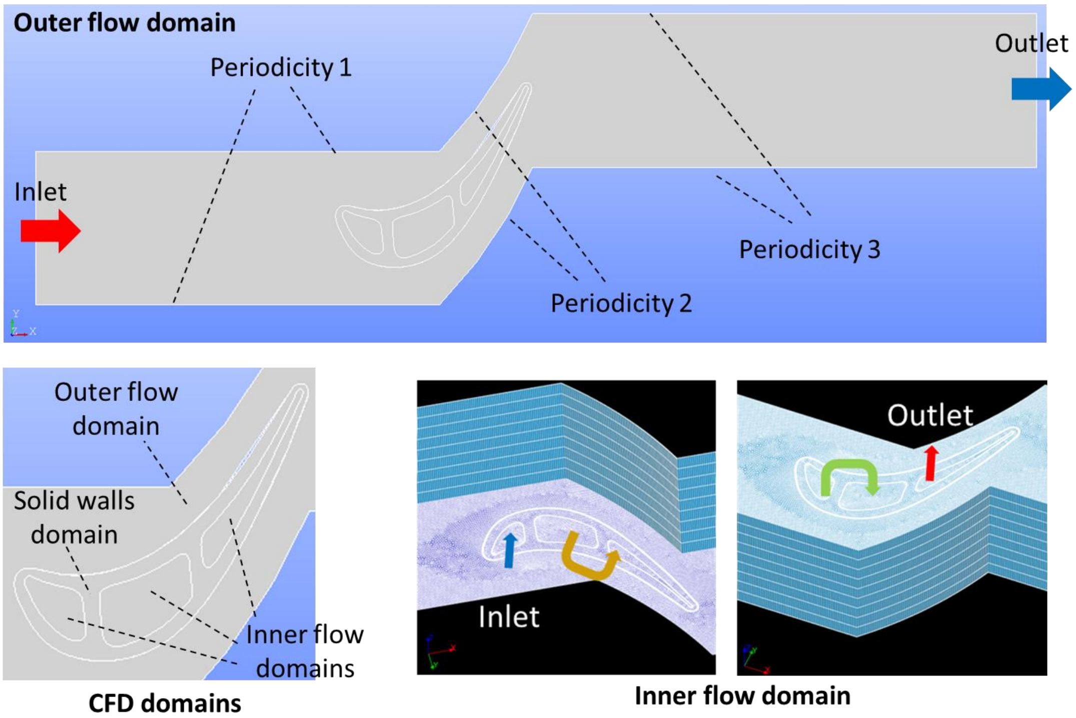


Figure 3: The CFD model domains and boundary conditions

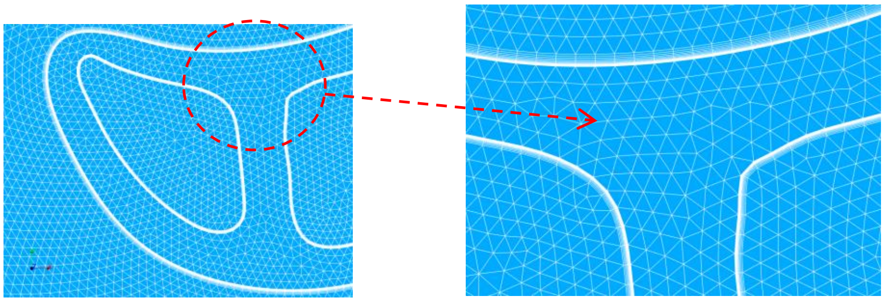


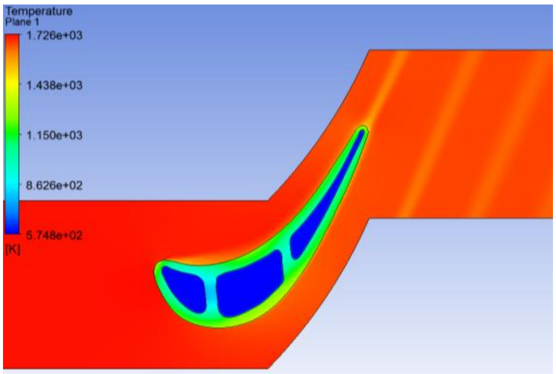
Figure 4: Views of the computational grid near turbine solid walls.

For the selection of the applied thermodynamic conditions conclusions from the work of Salpingidou et al. (2017), which were focused on typical turbine blade conditions for recuperative aero engine applications, were used. The applied conditions are presented in Table 1. Regarding the turbulence intensity for the hot-gas side the latter was selected to be equal to 10 % while for the cold-air side the turbulence intensity was selected to be equal to 5 %. For the calculation of the required cooling mass flow the methodology of Young and Wilcock (2002a and 2002b) and Wilcock et al. (2005) was taken into consideration. For the cooling mass flow calculation the necessary parameters describing the technology level of the cooling design (where only internal cooling was considered) were initially selected from literature suggestions in a similar way as described in the work of Salpingidou et al. (2017) with the maximum allowable metal temperature being 1,100

K, since a conservative material was assumed. At the next step, this cooling mass flow was applied to the CFD computations. Typical views of the CFD computations are presented in Figures 5 and 6.

Table 1: Main thermodynamic cycle conditions for the investigations

Parameter	Value
Hot-gas temperature	17,20 K
Cold-air temperature	580 K



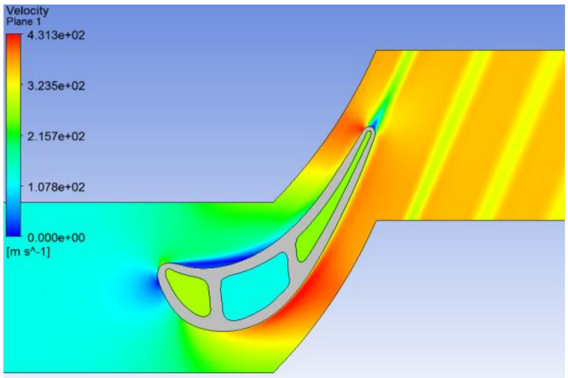


Figure 5: Temperature distribution at middle plane

Figure 6: Velocity distribution at middle plane

At the next stage the turbine blade geometry was split into 3 main regions, each one corresponding to each inner flow channel, as shown in Figure 7. Each one of these subregions was treated as a separate sub-unit in which the effect of turbine blade local geometrical features, such as inner channels length and diameter, on heat transfer and pressure losses was incorporated through the use of literature based correlations, presented in Table 2. In the overall analysis the turbine blade was modelled as a heat exchange unit composed of a number of these interconnected sub-units. The development of the surrogate tool was performed in the CAPE-OPEN/COCO free software, (COCO - the CAPEOPEN simulator, 2018), as shown in Figure 8.

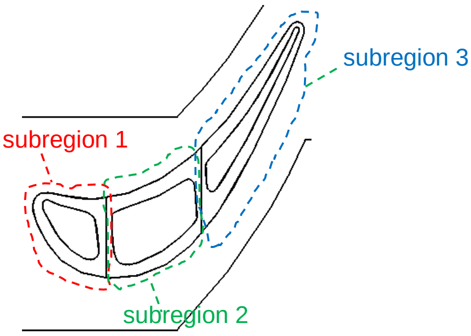


Figure 7: The turbine blade geometry main regions

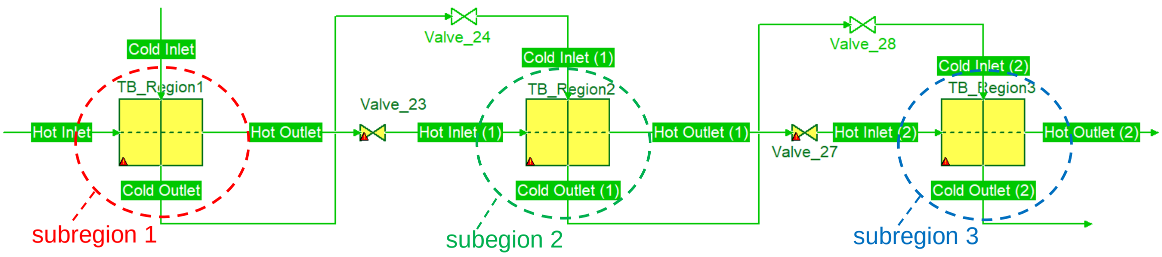


Figure 8: The surrogate tool modelling the turbine blade geometry as implemented in CAPE-OPEN/ COCO software

Table 2: Applied assumptions for surrogate tool and comparison with CFD results

	Hot-gas side	Cold-air side
Heat transfer	• Standard flat plate general correlation for turbulent flow (as presented in various textbooks) adapted with shape factor C_1 to include local velocity acceleration effects. Reynolds number was based on mean flow distance on turbine blade surface (similar to flat plate length definition). $$Nu{=}C_1\ 0.037Re^{0.8}Pr^{1/3}$$	• Gnielinski correlation for heat transfer, (Gnielinski, 1976). $$Nu{=}\frac{\left(\frac{f}{8}\right)(Re-1000)Pr}{1+12.7\left(\frac{f}{8}\right)^{0.5}\left(Pr^{\frac{2}{3}}-1\right)}$$
Pressure loss	• Adapted correlation for turbulent flow in smooth surfaces, (Petukhov, 1970), was applied with the assumption that the flow is developed similarly to flow between parallel plates. Reynolds number was based on hydraulic diameter calculation for wetted area. Shape factor C_2 was included to incorporate local velocity acceleration effects. $$f{=}C_2(0.790lnRe-1.64)^{-2}$$	• Correlation for turbulent flow in smooth pipes, (Petukhov, 1970). Shape factor C_3 was included to incorporate the effect of inner flow channels shape in relation to circular pipes. $$f{=}C_3(0.790lnRe-1.64)^{-2}$$
Surrogate tool input	• Turbine blade geometry (length, distance between blades, mean flow distance on turbine blade surface), outer flow temperature, pressure and mass flow conditions, Hot-gas side mean flow velocity	• Inner channel length and area, hydraulic diameter, inner flow temperature, pressure and mass flow conditions
Deviation in relation to CFD results	• Pressure loss deviation: -0.4 % • Temperature difference deviation: -1.0 K	• Pressure loss deviation: -1.3 % • Temperature difference:+0.6 K

As it can be seen in Figure 8, the turbine blade overall geometry was split into three subregions, the first one covering the turbine blade leading edge region, the second one covering the turbine blade middle region and the third one covering the trailing edge turbine blade region, as shown also in Figure 7, These regions were interconnected with each other in the same manner that the inner flow was circulating inside the turbine blade geometry in order to properly model the turbine blade thermal operation as a heat exchange unit consisting of a combination of three smaller heat exchange sub-units. During the development of the surrogate tool, the calibration of the coefficients reflecting the effect of the turbine blade geometrical characteristics of each subregion on pressure losses and heat transfer was performed in relation to the CFD results.

Table 3: Effect of heat transfer enhancers on turbine blade cooling

Heat transfer enhancer types Alam and Kim (2018)	Nu/Nu_o Nusselt number relative increase in relation to smooth surfaces (for inner flow)	f/f_o Friction losses relative increase in relation to smooth surfaces (for inner flow)	Q / Q_o Heat transfer relative increase in relation to smooth surfaces (for overall turbine blade)
Multiple arc ribs	5.85	4.96	1.37
Multiple arc ribs with gap	5.07	3.71	1.35
Smooth surfaces	1	1	1

At the next step, the effect of incorporating heat transfer enhancers on inner flow channels of turbine blade geometry was numerically assessed for two heat transfer augmentation surfaces, such as the ones presented in the work of Alam and Kim (2018), using experimental data available in literature describing their heat transfer and pressure loss characteristics. More specifically, the characteristics of surfaces with multiple arc ribs and multiple ribs with gap as described in Alan and Kim (2018) were used for the analysis. The application of these tube enhancers was numerically introduced through appropriate multipliers in relation to the smooth surface correlations for pressure loss and heat transfer. The results were assessed with the use of the surrogate tool (after its validation in relation to the CFD computations) and are summarized in Table 3 in order to identify configurations with promising effect on aero engine performance. In this respect in Table 3, the relative increase of heat transfer increase in relation to smooth surfaces (for overall turbine blade) was used as the comparison criterion.

3. Conclusions

In the present paper the development of a surrogate tool for the numerical investigation of turbine blade cooling for aero engine applications is presented. The tool was developed with the use of detailed CFD computations on a CFD model of a turbine blade with internal cooling channels which was used as the reference geometry for the investigation. The turbine blade was modelled as a heat exchange unit consisting of a number of interconnected sub-units. In each sub-unit appropriate heat transfer and pressure loss correlations from literature with necessary adaptations in relation to the CFD computations, were applied in order to include the flow conditions and geometrical characteristics effect on heat transfer and pressure loss. After the numerical calibration of the surrogate model in relation to the CFD computations the effect of heat transfer enhancers was numerically introduced by the application of appropriate experimentally derived coefficients available from literature. In this respect the effect of the characteristics of surfaces with multiple arc ribs and multiple ribs with gap, as described in Alan and Kim (2018), were used for the analysis having as the main comparison criterion the relative increase of heat transfer increase in relation to smooth surfaces (for the overall turbine blade). The analysis showed that both types of heat transfer enhancers can provide significant heat transfer increase of ~35% in relation to the case were no heat transfer enhancers were applied and thus, their introduction in turbine blade cooling channels can be of practical interest. For this reason, a more detailed investigation, incorporating the effect of heat transfer and pressure loss on the overall aero engine thermodynamic cycle, is planned for future research actions which will be also combined with the numerical derivation of dedicated correlations heat transfer and pressure loss correlations for turbine blade geometries and also extend the surrogate tool to include turbine blade film cooling.

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