IMPROVEMENT OF THE DESIGN AND METHODS OF DESIGNING CRITICAL COMPONENTS OF GAS FLOW PATH OF MODULAR AIRCRAFT GAS TURBINE

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Abstract

The results of the analysis of the structure of the critical elements of gas turbine air cooling systems are presented. The possibility of improving the effectiveness of film cooling of turbine stator and rotor blades and reduce of air leakages in this system through a highly effective complex of step labyrinth seals is shown.

1 Introduction

Development of the optimum design solutions of the most often damaged during operation modules high-pressure turbine (HPT) stator and rotor of current turbofan engines, such as block module of turbine nozzle guide vanes, which at current repair could be replaced to a new one without removing all subsequent assemblies of the rotor and stator hottest part of the generator, and the contour of the fan with the entire exhaust system, is of extremely important practical value and will contribute significantly to the reduction of life cycle cost of the entire power plant.

Another relevant and practically important problem now is to increase the efficiency of cooling of blades of high-temperature turbines, as nozzle guide vanes and turbine wheel blades, as well as the coefficient of performance of the turbine because of the increased airflow to cooling, with a significant increase in the turbine inlet gas temperature in the future engines of new generations, as well as the growth of the compressor pressure ratio, leads to a non-optimal from the standpoint of gas dynamic design of the airfoil blades, as well as a reduction in the height that significantly increases the amount of core and secondary loss of energy of the gas, the effect of radial tip clearance on the efficiency of turbine etc.

Very substantial increase in the efficiency of the turbine stages is possible with reducing gas inflow of HPT internal air cavity in her gas flow path, which occurs through the axial clearance between the nozzle assembly and the turbine wheel, as 1% of the inflowing air can reduce the efficiency level of about 3%. This problem, as will be shown below, can be solved by using the most effective labyrinth seal turning a flow on 360 degrees and choice when designing such a degree of reactivity level at which there is perhaps a small gas pressure gradient between the blade root and the turbine wheel adjacent air cavity.

Presented in this work constructive solutions equally may be used for designing both the turbines of aircraft engines and land-based gas turbines.

2 Design enabling technology to replace a damaged in operating unit of turbine nozzle guide vanes

Hot air part of aviation air-breathing engines has a resource in hours and cycles significantly smaller than the assemblies of its cold air part. Therefore, the design is particularly important to identify and to provide structurally and technologically replacing in operation of the most damaged engine components: combustion chamber, turbine rotor and stator. For example, during the operation take place burnouts and destruction of turbine nozzle vanes, and connected with a circumferential radial non-uniformity of the temperature field at the outlet from the combustion chamber, there are also thermal cracks on the outlet edge of the blade, at their junction with the airfoil shroud platforms, which arise from due to temperature differences and the thicknesses of these structural elements, etc. Modularity of HPT nozzle guide vanes in currently existing Russian and foreign engines, is provided constructively so that to replace, for example, having burnouts blocks of the turbine, it is necessary to remove all assemblies installed in the subsequent parts of the engine, turbine rotors modules including high and low pressure with their supports, because these modules are formed as annular elements, as for example (see Figure 1). When reassembling the engine has to be reduced, for example, the alignment of the rotors, the diametrical position of the contact surfaces mazes, etc., which is very difficult, with

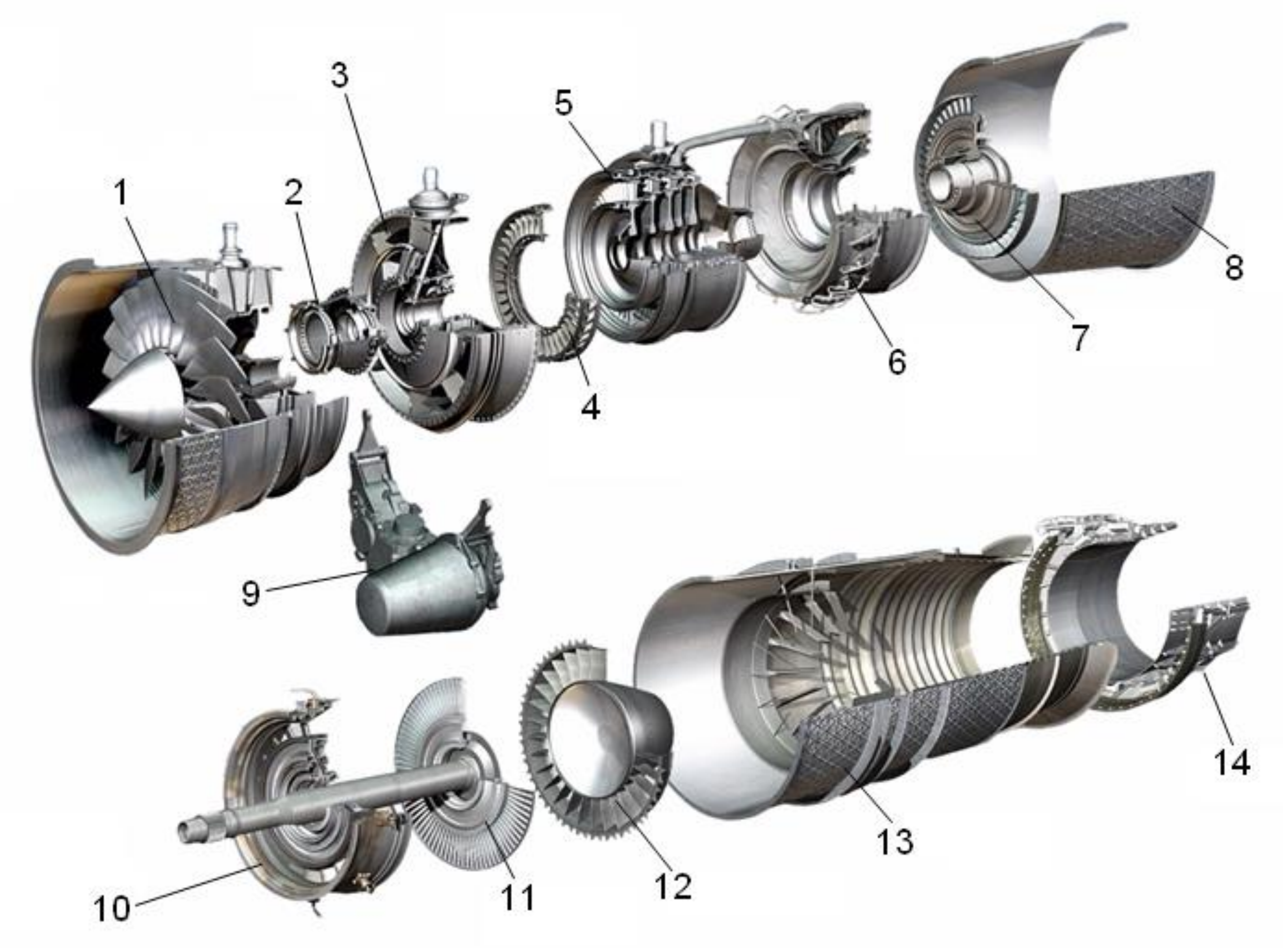

Fig. 1. Modules GTE EJ 200 [1]

1…5, 8, 9 – cold part; 6, 7, 10…14 – hot part

low rigidity of the aircraft GTE stator, associated with the requirements to minimize its weight in the design. In the design of the GTE, shown in Figure 1, the outer contour of the motor housing, designated under the number 8 has no horizontal joint and nozzle guide vanes assembly is combined into a single module 6 with annular combustion chamber, which greatly complicates the removal and replacement of the damaged unit if local repair is necessary.

In Figure 2 and 3 the original and the new, streamlined design of the module are shown, consisting of the assemblies of the flame tube and nozzle guide vanes of the turbofan with afterburning. This design was modified to allow the removal and replacement of the damaged unit in the direction, perpendicular to the longitudinal axis of the engine, which is of fundamental importance as in this case, when performing work on the replacement of the damaged unit does not require removal of the engine of a large number of the adjacent annular rotor and stator assemblies of HPT and LPT and of entire exhaust parts turbofans.

Claw connection of blocks of turbine stator vanes with ring support is shown in Figure 2 on the right. It allows the removal of these blocks from the assembly of turbine guide vanes during repair and the mount them to the original place as at the modernized module the case 3 and the support ring 4 is designed detachable. Of course, the complete solution to this problem also includes designs for sealing: the mating faces, such as shrouds of separate blocks of CA; releasable connecting of the flame tube combustor and the turbine stator, and to provide the required flow passage area of guide vane bucket system, replacing of individual modules of sets of units, etc. However, the constructive solutions to ensure these requirements are known, so the designers of turbine nozzle guide vanes can be recommended to actively work in this direction.

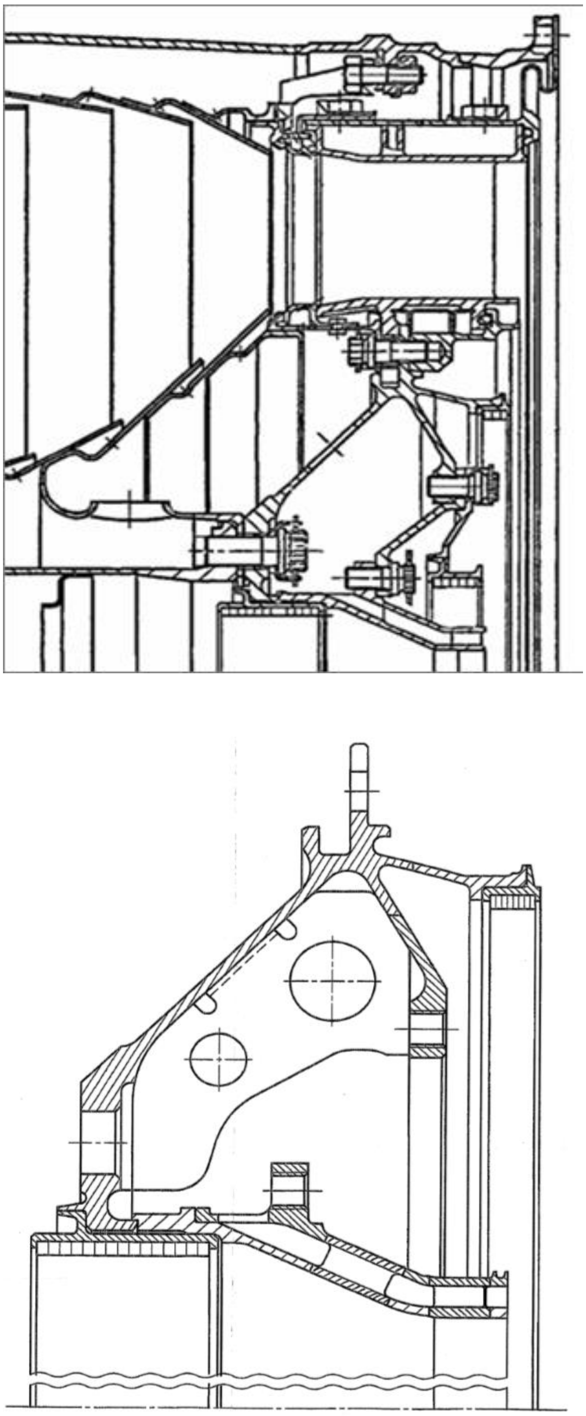

Fig. 2. Module design, including components of the combustion chamber and of the turbofan turbine nozzle vanes: at the top - the original design of the module; bottom – assembly of ring bearing blocks

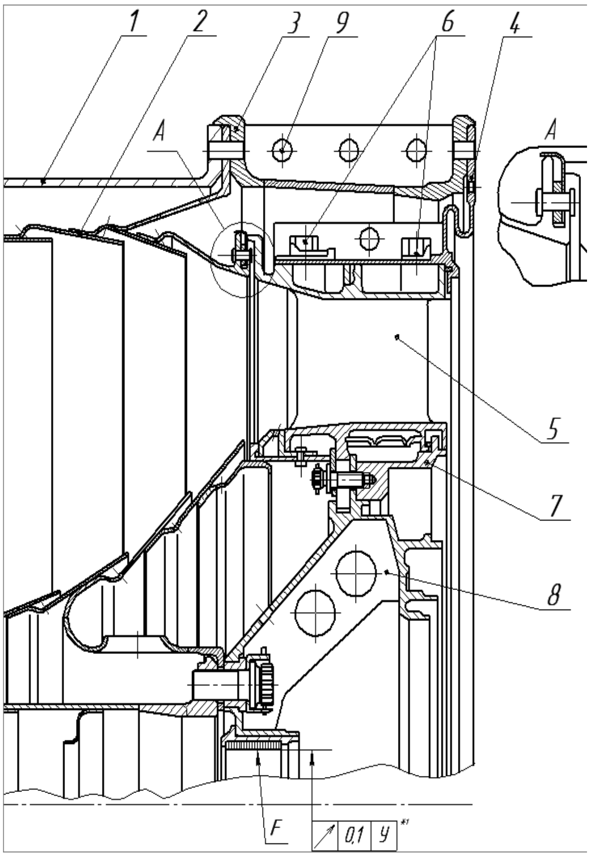

Fig. 3. New design output of the combustion chamber and of the turbofan turbine nozzle vanes:
1, 2 - casing and the flame tube of the combustion chamber interconnected by a flange; 3 - split casing; 4 - detachable carrier ring blocks; 5 – turbine nozzle guide vanes block; 6 - mounting bolts of block 5 to split ring 4; 7, 8 - one-piece annular bearing parts of turbine stator vanes.

3 Improved film cooling of turbine stator guide vanes and disk blades

Figure 4 shows the modern HPT nozzle vanes and moving blades with the film-cooling, in which cooling air is fed from the internal cavity to the outer surface of the profile through a channel of small diameter, about 0.3 mm. At the exit of the channel it has an expanding cone portion disposed at substantially smaller angle to the surface of the profile, which must contribute to possible closely aligned stream of cooling air to the blade surface. The small diameter of the bore reduces the throw distance of the cooling air, which, due to technological limitations, is usually directed in these constructions at large angles to the cooling surface of the airfoil, of 25 ... 35 degrees minimally. Particularly, that is why for cooling of modern HPT rotor blades requires a large amount of cooling air, about 4.0 ... 4.5% of the total air flow through the inner contour of the turbofans.

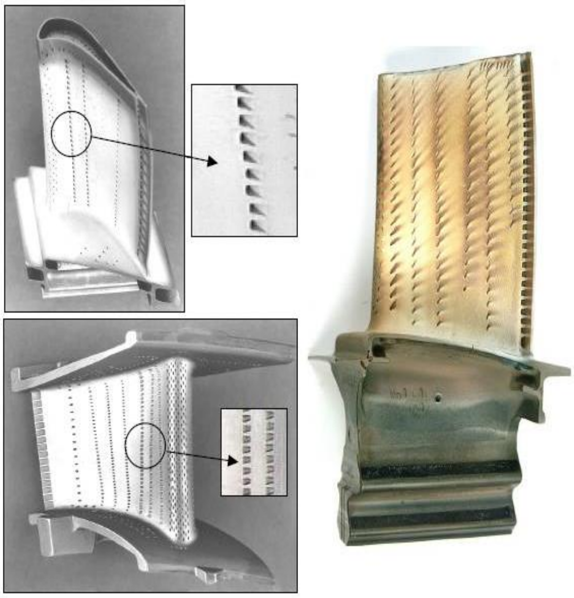

Fig. 4. Rotor and stator HPT blades with film cooling of the airfoil

3.1 HPT rotor blades with a slot cooling of profile surface

Obviously, that the discrete film cooling channels it is appropriate to replace with the slot, wherein the angle may significantly reduce the yield of the cooled air and orient the jet of cooling air in the wall layer of the hot gas and thereby increase the effectiveness of film cooling of high-temperature turbine blades. In Figure 5 shows a sectional view of the airfoil of the turbine rotor blade with two slotted channels 2, positioned on the concave surface of the blade.
The bearing capacity of the structure is provided with jumpers 1, compounding both sides of the profile. In Figure 5 is also shown that the gap can be made as in the lintel area

and anywhere profile for which it is necessary to arrange in the inner cavity wall of the extension. Considered design blade with slot cooling can be cast on the currently available technologies. Minimum number of slots on the concave surface of the blade profile must be two, as shown in Figure 5.
Input edge of the blade has a cyclone cooling system.

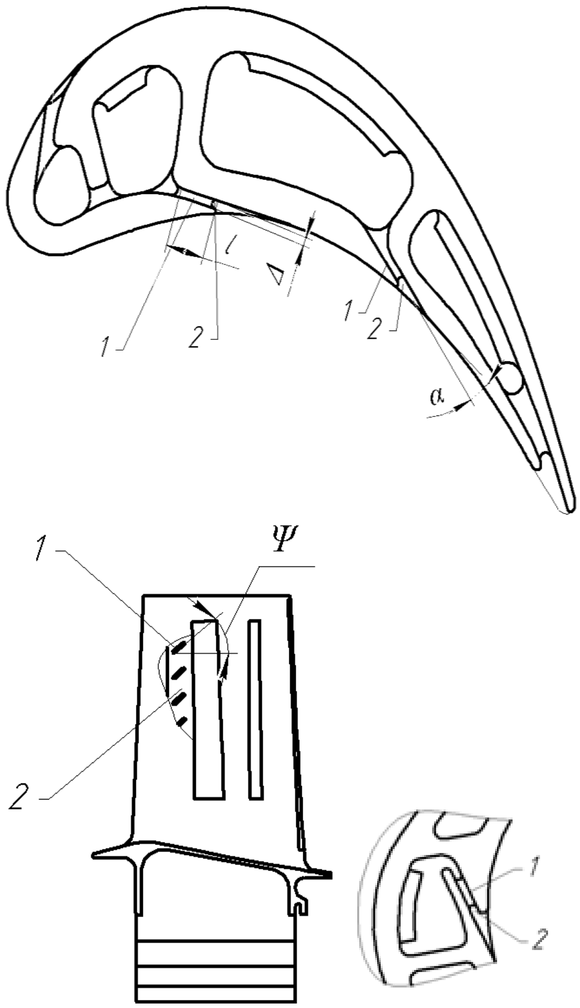

Fig. 5. HPT rotor blades with a slot film cooling of concave side of the blade

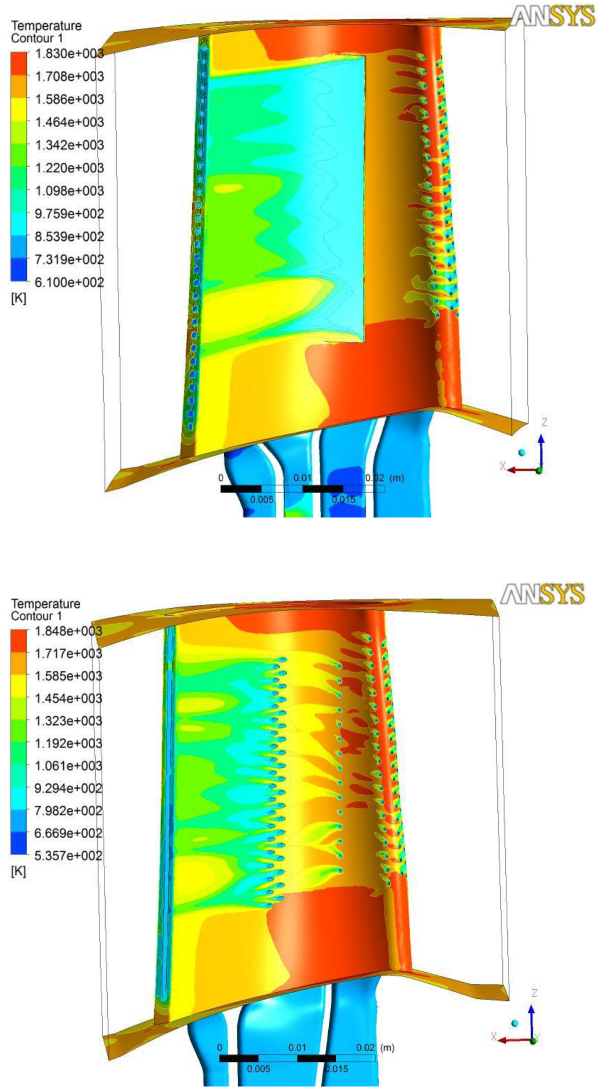

Fig. 6. HPT rotor blades with film cooling, formed by a slot (on the top) and by a single channel system (below), located on the concave side of the blade HPT rotor blade

3.2 Improvement of shroud film cooling of high-temperature HPT guide vanes

The film cooling of guide vanes of the high-temperature HPT nozzle currently consumes more than 10% of the cooling air of the compressor bleed. Therefore, the problem of its rational use is highly relevant. In some designs of the turbine guide vanes, the critical element in determining capacity for its work are tip shrouds and, in particular, the joints on the shrouds at the outlet, which forms metal burnouts, reaching the critical section of the profiles, as shown in Figure 7.

Fig.7. Block of guide vanes of high-temperature film-cooled platforms, formed by a system of single cylindrical channels

Cause of violations of film cooling gas are secondary currents, generated in the boundary layer due to the gas pressure gradient between the concave and convex surface of inter-blade channel, as can be seen in Figure 8.

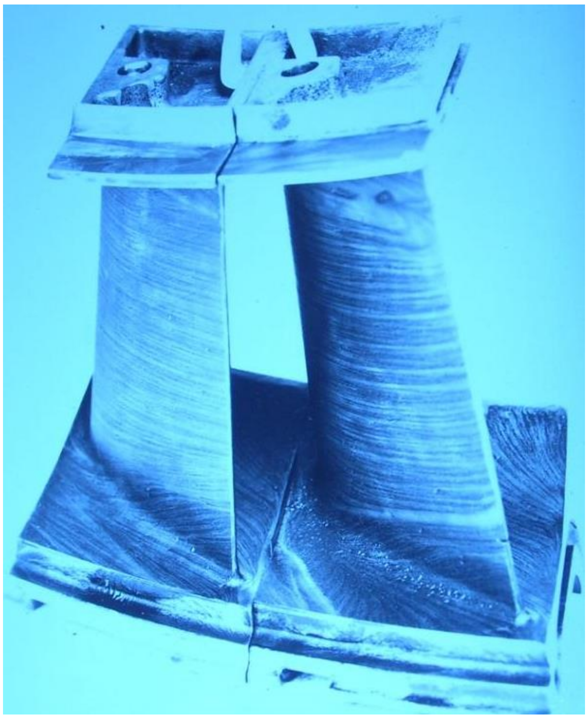

Fig.8. Experimental streamlines on the surface of vanes and parietal layer of the internal plateform.

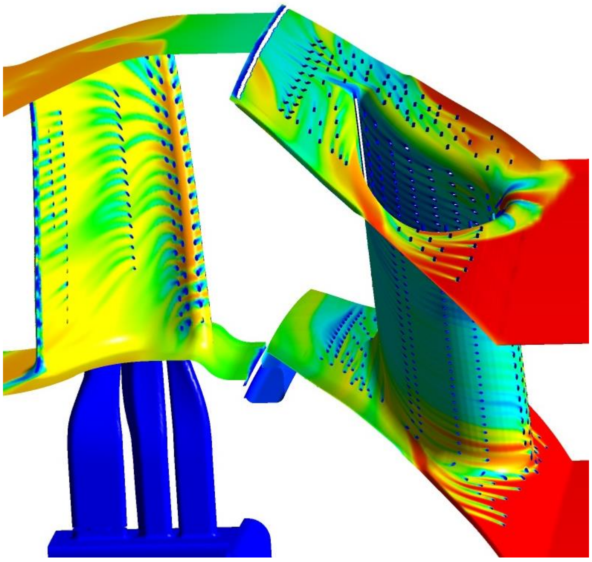

Fig.9. Calculated streamlines on the surfaces of the nozzle stator and working rotor blades with intense film cooling, formed by a system of single cylindrical channels

They prevent the uniform cooling of the platforms in the wall layer of the inter-blade channel due to changes in the trajectory of the cooling air jets that focus on the direction of the back of the profile, and not along the channel, as shown in Figure 9 for the internal platform. This paper considers the task of creating for platforms such a meridian form of turbine guide vane profils, which would promote a uniform, not a deformed distribution of streamlines in the cross-section of inter-blade channel. It is not advisable to secondary currents of gas to go back side of the profile and caused separated flows induced gas flow causing additional loss of energy of the gas, reducing the efficiency of the turbine stages. This problem is particularly actual for small-sized gas turbine engines and advanced turbofans, which total value of the pressure ratio of the compressor increased currently about 1.5 ... 2.0 times and is closer to the values of 50 ... 60.

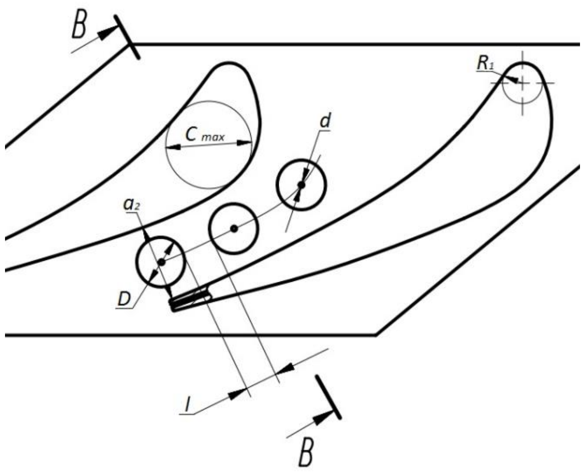

Fig.10. Constructive experimental arrangement of periodic spherical indentations on the face of the platform channel inter-blade

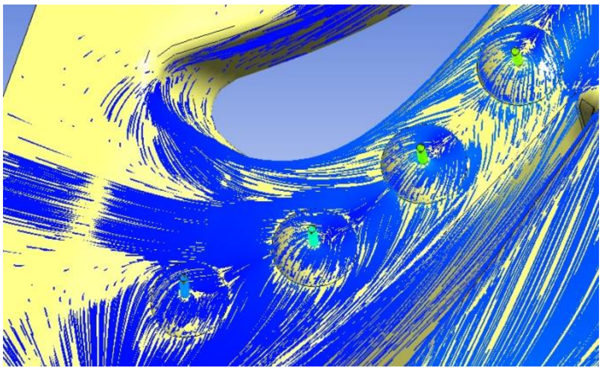

Fig.11. Streamlines on the platform in the presence of periodic spherical indentations

Calculated streamlines obtained in the package Ansys CFX, confirms the absence of secondary gas flows on the surface of platform of guide vane and the possibility of effective film cooling.

4 Decrease in of air at the inlet to the root zone of the rotor blade turbine HPT

As is known, the leak of the cooling air from the cooling cavity through gaps between the rotor and the stator of the turbine stage in a transverse direction relative to the main gas flow leads to a significant reduction in the efficiency of the step. Most important to ensure minimum air ingression through the axial clearance between the rotor wheel and the stator blades as 1% of the air reduces the efficiency level of about 3%. Currently the high-effective brush seals are widely used. However, their use is limited by high peripheral speed of the rotor. On Figure 12 shows the step labyrinth seal-ring with turning on 180 degrees of flow in the swirl lattice of theater of EJ 200 turbofan engine. Moreover, attention is drawn to the fact that the top, most responsible seal works in worse conditions than the lower, because it is not a gas leak occur in the direction of the step, but vice versa. Experimental studies of such seals carried by the authors have shown that their effectiveness depends on the flow direction and the distance from the step to the ridge can be varied by about 50%.

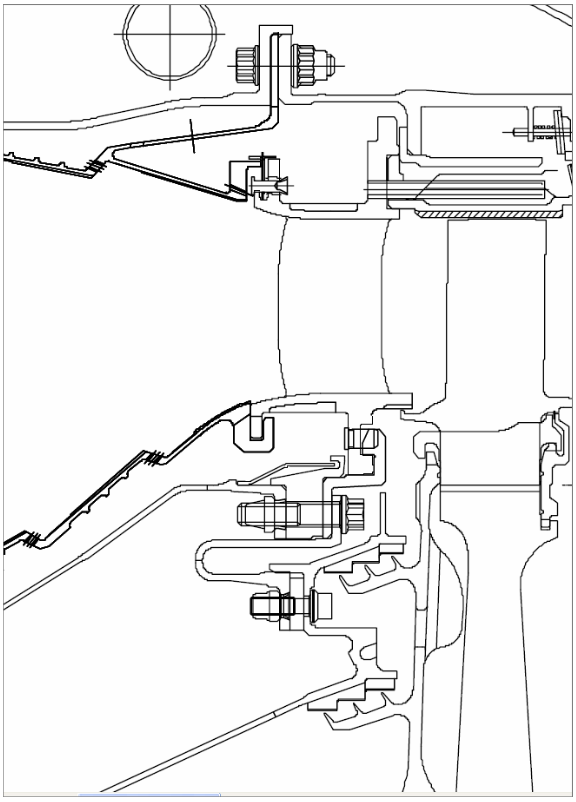

Fig.12. The structural layout of the HPT with a system of stepped labyrinth seals

However, as presented in Figure 12 design labyrinth seals is not the optimal. More efficiently work the seals with the flow turning to 360 degrees. This type of combined stepped labyrinth seal shown in Figure 13.

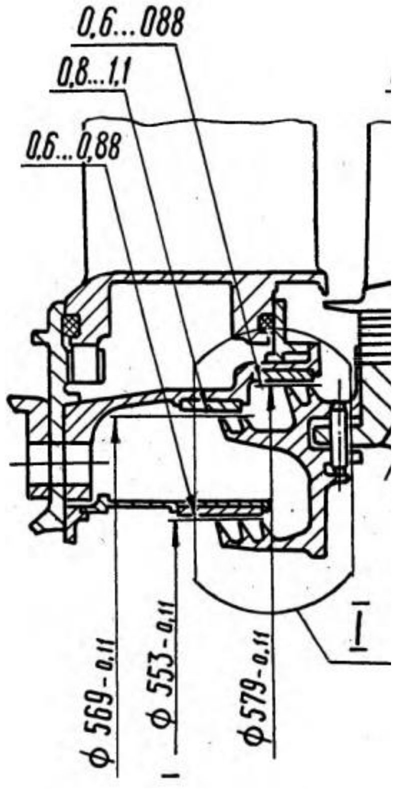

Fig. 13. The combination of seals with turning of flow on 360 and 180 degrees

For its estimate of the flow rate and pressure drop of gas on the maze, one must perform all hydraulic calculation of the air system, bleed air from the space in the cooling system. Most uncertain quantities determining the results are diametrical clearance between the ridge and the wall of the maze. In the case where the applied cell and ridges labyrinths work for plunging, gas leakage characteristics become more ambiguous and depend mainly on the value of gas pressure differential between the cavities at the entrance and exit of the maze. However, this problem requires additional information, since the twist of the outlet flow from the labyrinth greatly reduces the amount of gas pressure increases the pressure differential, i.e. gas flow rate. Therefore, the actual values of pressure and temperature in the gas turbine air cavities are determined experimentally, in a study of the entire engine.

5 Conclusions

The results of the theoretical and experimental research, presented in this paper, should be checked in each particular design of aircraft turbine engine, taking into account the dimensions and actual operating conditions. However, certainly such scientific and technical studies are relevant and necessary, that would analyze and summarize currently available design experience, as well as new designs are to be considered to facilitate the development of more advanced gas turbine engines.

References

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