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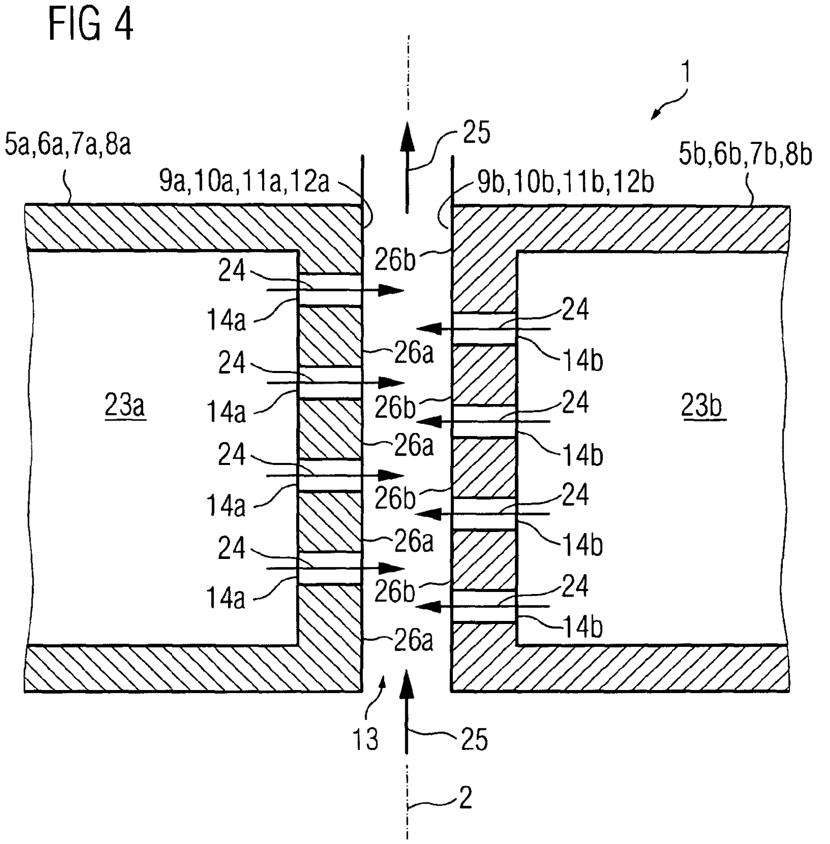
(54) **Turbine assembly having cooling arrangement and method of cooling**

(57) The present invention relates to cooling of blade or vane endwall or heat shields in a turbine assembly. As herein disclosed, a turbine assembly (1) comprises a first segment (5a,6a,7a,8a) and a second segment (5b, 6b,7b,8b) of blade or vane endwalls (5,6,8) or heat shields (7). The first segment (5a,6a,7a,8a) and the second segment (5b,6b,7b,8b) are arranged side by side along a circumferential direction with respect to a machine axis (1) with a separating gap (13) therebetween. The first segment (5a,6a,7a,8a) has a first sidewall (9a, 10a,11a,12a) and the second segment (5b,6b,7b,8b) has a second sidewall (9b,10b,11b,12b). The separating gap (13) extends from the first sidewall (9a,10a,11a,12a) to the second sidewall (9b,10b,11b,12b). In accordance with the present invention, the first sidewall (9a,10a,11a, 12a) has a perforation (14a) therethrough for conducting cooling air (24). The perforation (14a) is configured to eject the cooling air (24) into the separating gap (13) causing the cooling air (24) to impinge on the second sidewall (9b,10b,11b,12b).

FIG 4



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Description

[0001] The present invention relates turbomachines such as gas turbines. In particular, the present invention relates to cooling of blade or vane endwalls (platforms) and heat shields in turbomachines.

[0002] Modern gas turbines operate under extremely high gas temperature conditions. This requires the use of heavy cooling of the airfoils and the endwalls (also referred to as platforms) of turbine blades and vanes to ensure sufficient lifetime of these blades and vanes.

[0003] The efficiency of a gas turbine can be increased by optimizing certain parameters of the operation of the turbine. In particular, the most relevant parameters that affect the efficiency are temperature and pressure of the gas medium which forces the rotation of the turbine rotor, referred to herein as drive gas. The normal operating temperature of the drive gas nowadays, especially at the turbine inlet region is already significantly higher than the admissible material temperatures of the turbine components exposed to this fluid. Material temperatures that are too high lead to drop in strength of such heat exposed components. Temperature exceeding said limits cause melting and/or formation of cracks in the component, which may eventually cause local or complete destruction of the component. In order that the component temperatures do not exceed the admissible material temperatures, turbine components exposed to high temperatures are therefore cooled by a cooling medium.

[0004] A turbine assembly generally includes a plurality of stationary vanes and rotating blades. Each blade or vane includes an airfoil portion that extends into the flow path of the drive gas flowing axially through the turbine. The base of the airfoil is arranged on a platform or an endwall. Several such platforms are set side by side in an annular manner. In case of stationary vanes, such platforms or endwalls are also arranged to form an annular shroud at the tip of the blading. In case of rotating blades, structures commonly referred to as heat shields are often arranged annularly above the tip of the rotor blades to protect the turbine stator from high temperature of the turbine fluid.

[0005] Endwall cooling is gaining importance due to increasing turbine entry temperatures. In a commonly used method for endwall cooling, coolant is discharged through discrete holes in the endwalls of turbine blade or vane. After leaving the holes, the coolant forms a protective layer between the hot drive gas and the surface that is to be protected. However, in this method of film cooling, the ejected coolant interacts with the external flow near the endwall and generates aerodynamic and thermodynamic losses in the process. This reduces turbine stage efficiency and together with the consumption of cooling air is detrimental to the overall cycle efficiency.

[0006] Another known method for endwall cooling involves providing longitudinal holes through the endwall at the sides to allow cooling air to penetrate these holes, to effect a convective cooling. An example of such a technique is disclosed in the document US6309175B1. Disadvantageously, this method involves a risk of clogging of these holes by dust and exposure of the material around these holes to thermo-mechanical fatigue. Additionally, the drilling of such longitudinal holes also presents manufacturing complexities due to the inherent geometry of these components.

[0007] Further, due to the circumferentially adjacent arrangements of the endwalls and heat shields, a slit or a gap is naturally present between any two adjacent endwalls or heat shields. The drive gas, having higher pressure upstream than downstream along the axial flow path, penetrates through these gaps, thus causing significant heating of the tangential sides (sidewalls) of these endwalls and heat shields that leads to a reduction in the life of these components due to heating. Further, such parasitic leakage of the drive gas through these gaps leads to aerodynamic losses causing drop in turbine efficiency.

[0008] The object of the present invention is to provide an improved method for cooling blade or vane endwalls and heat shields that addresses the above mentioned problems posed by the existing state of the art.

[0009] The above object is achieved by the device according to claim 1 and the method according to claim 5.

[0010] Further embodiments of the present invention are disclosed in the dependent claims.

[0011] In summary, the present invention provides an improved cooling of blade and vane endwalls or heat shields that are circumferentially arranged side by side with a separating gap between the sidewalls any two adjacent segments of these endwalls or heat shields. The adjacent segments are referred to herein as first and second segments. The underlying idea of the present invention is to "close" the separating gap between the sidewalls by feeding the separating gap with cooling air, which, in turn, is used for cooling the sidewalls of adjacent endwall or heat shield segment. This is done by providing a perforation through at least one of the sidewalls, namely the first sidewall. The perforation conducts cooling air and ejects the same into the separating gap to impinge on the second sidewall.

[0012] One advantage of the present invention is that the parasitic leakage of the drive gas through the separating gap is eliminated or minimized due to the "closing" of this gap by the ejected cooling air. This improves the turbine stage efficiency and overall gas turbine efficiency.

[0013] A second advantage of the present invention is that the ejected cooling air provides impingement cooling of the second sidewall facing the first sidewall across the separating gap, thereby increasing the lifetime of the neighboring second segment.

[0014] A third further advantage of the present invention is that, a portion of the first sidewall is also cooled convectively by means of a heat sink thanks to the perforation penetrating through the material of the first sidewall.

[0015] A fourth advantage of the present invention is that the two-fold convective and impingement cooling provides a reduction in the amount of cooling air used for sidewall cooling, thus contributing to an increase in turbine stage and overall efficiency.

[0016] In accordance with a further embodiment, to provide uniform tangential cooling of the sidewalls, multiple such perforations are provided on both, the first and the second sidewall. These perforations on the first and second sidewalls have a relative arrangement, such that the cooling air ejected through the perforations on each sidewall impinges on spaces in between the perforations on the opposite sidewall.

[0017] The present invention is further described hereinafter with reference to illustrated embodiments shown in the accompanying drawings, in which:

FIG 1 schematically shows a longitudinal cross-sectional view of a turbine assembly,

FIG 2 schematically shows a cross-sectional view from an axial direction along a line A-A across a guide vane in FIG 1,

FIG 3 schematically shows a cross-sectional view from an axial direction along a line B-B across a rotor blade in FIG 1, and

FIG 4 schematically shows a cross-sectional view along the lines C-C across the endwall or heat shield segments in FIGS 2 and 3.

[0018] Embodiments of the present invention are described below referring to the accompanying drawings.

[0019] Referring to FIG 1 is illustrated an exemplary turbine assembly 1 in a high pressure turbine stage in a gas turbine engine. The assembly 1 is understood to be generally symmetrical in cross-sectional view about a longitudinal machine axis 2. The turbine assembly 1 includes a set of stationary guide vanes 3, one of which is shown in the cross-sectional view to the left of FIG 1. The turbine assembly 1 further includes a set of rotating blades 4, one of which is shown in the cross-sectional view to the right of FIG 1. The set of guide vanes 3 and the set of blades 4 are each mounted in annular formation around the machine axis 2 with each guide vane 3 and each blade 4 extending radially outwardly from the axis 2. Gases from a combustion process in a combustor (not shown) force their way into the guide vanes 3, whereupon they are expanded and imparted a spin in a direction of rotation of the blades 4. These gases, referred to herein as drive gas, then impact the blade 4, causing rotation of the blades 4 about the machine axis 2. The axial flow direction of the drive gas is denoted by the arrow 25.

[0020] Each guide vane 3 includes an airfoil 3a which extends radially into the axial flow path of the drive gas between an outer endwall 5 and an inner endwall 6. Multiple segments of outer 5 and inner 6 endwalls are arranged side by side along a circumferential direction to effect an annular formation of outer 5 and inner 6endwalls endwalls respectively.

[0021] Each blade 4 includes an airfoil 4a which extends radially into the axial flow path of the drive gas. A blade platform 8 extends circumferentially from the blade 4 at the base of the airfoil 4a. One or more blades 4 may be arranged on a single platform 8. Multiple such segments of endwalls are arranged side by side along a circumferential direction to effect an annular formation of endwalls around the axis 2.

[0022] To protect the stator portion surrounding the blades 4, segments 8 referred to as heat shields are annularly mounted above the tip of the blades 4. The heat shield segments 7, similar to the endwall 5,6,8 segments are also arranged side by side along a circumferential direction.

[0023] The circumferential arrangement of the outer 5 and inner 6 endwall segments of the guide vanes is illustrated in FIG 2, which is a cross-sectional view along the section line A-A through the guide vanes 3 in FIG 1. Referring to the top portion of FIG 2, adjacent segments of the outer endwall 5 are denoted as a first segment 5a and a second segment 5b. The segments 5a and 5b are arranged circumferentially side by side with a separating gap 13 between them. The first segment 5a has a first tangential sidewall 9a that faces a second tangential sidewall 9b of the second segment 5b, such that the gap 13 extends between these sidewalls 9a and 9b. In accordance with the present invention, one or more perforations 14a is provided through the first sidewall 9a for conducting cooling air, such that the cooling air is ejected from the perforations 14a into the gap 13 to impinge on the neighboring second sidewall 9b. This provides impingement cooling of the second sidewall 9b and convective cooling of the first sidewall 9a thanks to the perforations penetrating through the material of the first sidewall 9a. The two-fold convective and impingement cooling advantageously provides a reduction in the amount of cooling air used for sidewall cooling, thus contributing to an increase in turbine stage and overall efficiency.

[0024] Further, as shown, one or more perforations 14b are provided through the second sidewall 9b to conduct cooling air to likewise provide impingement cooling of the neighboring first sidewall 9a as well as convective cooling of the second sidewall 9b. In this example, the segments 5a and 5b each respectively includes a first face 15a and 15b exposed directly to the axial flow path of the drive gas, and a second face 19a and 19b radially opposite to the respective first surfaces 15a and 15b. The second faces 19a and 19b define respective cavities 23a and 23b wherein cooling air is supplied, for example, from the last stages of a compressor of the gas turbine engine. As illustrated in FIG 2, the one or more perforations 14a and 14b through the sidewalls 9a and 9b are configured such as to conduct the cooling air from the respective cavities 23a and 23b.

[0025] A similar cooling arrangement may be advantageously used for cooling the guide vane inner endwalls

6. Referring to the bottom portion of FIG 2, two adjacent segments of the inner endwall 6 are denoted as a first segment 6a and a second segment 6b. The segments 6a and 6b are arranged side by side with a separating gap 13 between them. The first segment 6a has a first tangential sidewall 10a that faces a second tangential sidewall 10b of the second segment 6b, such that the gap 13 extends between these sidewalls 10a and 10b. Likewise to the earlier mentioned example, one or more perforations 14a and 14b are provided through the first sidewall 10a and second sidewall 10b for conducting cooling air therethrough, and ejecting the cooling air into the gap 13 to impinge on the respective neighboring sidewall 10b and 10a, for convective cooling of the respective sidewall 10a and 10b and impingement cooling of the neighboring sidewall 10b and 10a. The one or more perforations 14a and 14b conduct cooling air from a respective cavity 23a and 23b defined on faces 20a and 20b of the segments 6a and 6b that are radially opposite to faces 16a and 16b that are directly exposed to the drive gas.

[0026] A similar cooling arrangement may be also provided for blade 4 platforms or endwall 8 and heat shield 7. The circumferential arrangement of the heat shield 7 and the endwall or platform 8 of the blades 4 is illustrated in FIG 3, which is a cross-sectional view along the section line B-B through the guide vanes 3 in FIG 1. Referring to the top portion of FIG 3, two adjacent segments of the heat shield 7 are denoted as a first segment 7a and a second segment 7b. Referring to the bottom portion of FIG 3, two adjacent segments of the blade platform or endwall 8 are denoted as a first segment 8a and a second segment 8b. Referring to FIG 3 in general, the respective first segments 7a,8a and second segments 7b,8b are arranged side by side with a separating gap 13 between them. The respective first segment 7a,8a has a respective first tangential sidewall 11a,12a that faces a second tangential sidewall 11b,12b of the respective second segment 7b,8b, such that the gap 13 extends between the sidewalls 11a and 11b and between 12a and 12b. Likewise to the earlier mentioned example, one or more perforations 14a are provided through the first sidewall 11a, 12a and one or more perforations 14b are provided through the second sidewall 11b,12b for conducting cooling air therethrough. The perforations 14a and 14b eject cooling air into the gap 13 to impinge on the sidewalls of the neighboring segment. Referring to the top portion of FIG 3, the perforations 14a and 14b conduct cooling air from a respective cavity 23a and 23b defined on faces 21a and 21b of the segments 7a and 7b that are radially opposite to faces 17a and 17b that are directly exposed to the drive gas. Referring to the bottom portion of FIG 3, the perforations 14a and 14b conduct cooling air from a respective cavity 23a and 23b defined on faces 22a and 22b of the segments 8a and 8b that are radially opposite to faces 18a and 18b that are directly exposed to the drive gas.

[0027] Referring to FIG 4 is illustrated an arrangement of the perforations 14a and 14b on opposite sidewalls relative to each other. FIG 4 is a schematic illustration wherein the first segment depicted on the left side of FIG 4 could be any of the segments 5a,6a,7a,8a mentioned above. Accordingly the second segment depicted on the right side of FIG 4 would include any of the respective adjacent segments 5b,6b,7b,8b. The respective tangential sidewalls are shown as 9a,10a,11a,12a and 9b,10b, 11b,12b. The illustrated embodiment has multiple spaced apart perforations 14a through the sidewalls 9a, 10a,11a,12a and multiple spaced apart perforations 14b through the sidewalls 9b,10b,11b,12b. The relative arrangement of the perforations 14a and 14b are such that the streams of cooling air 24 ejected through the perforations 14a impinge on the sidewall 9b,10b,11b,12b on spaces between the perforations 14b on the sidewall 9b, 10b,11b,12b. Likewise, the streams of cooling air 24 ejected through the perforations 14b impinge on the sidewall 9a,10a,11a,12a on spaces between the perforations 14a on the sidewall 9a,10a,11a,12a. To achieve the same, the perforations 14a and 14b have a staggered arrangement relative to each other as shown, i.e., the perforations 14a and 14b are spaced apart with an axial shift relative to each other.

[0028] In addition to the above-mentioned advantages of the earlier described embodiments, having multiple perforations advantageously provides uniform tangential cooling of the sidewalls 9a,10a,11a,12a and 9b,10b,11b, 12b. Besides, filling of the tangential separating gap 13 by cooling air ejected through the perforations 14a and 14b helps minimize leakage of the drive gas through the gap 13 and hence improve turbine efficiency.

[0029] While this invention has been described in detail with reference to certain preferred embodiments, it should be appreciated that the present invention is not limited to those precise embodiments. Rather, in view of the present disclosure which describes the current best mode for practicing the invention, many modifications and variations would present themselves, to those of skill in the art without departing from the scope and spirit of this invention. The scope of the invention is, therefore, indicated by the following claims rather than by the foregoing description. All changes, modifications, and variations coming within the meaning and range of equivalency of the claims are to be considered within their scope.

Claims

1. A turbine assembly (1), comprising:

a first segment (5a,6a,7a,8a) and a second segment (5b,6b,7b,8b) of a blade or vane endwall (5,6,8) or heat shield (7), said first segment (5a, 6a,7a,8a) and said second segment (5b,6b,7b, 8b) arranged side by side along a circumferential direction with respect to a machine axis (1) with a separating gap (13) therebetween, said first segment (5a,6a,7a,8a) having a first sidewall

(9a,10a,11a,12a) and said second segment (5b,6b,7b,8b) having a second sidewall (9b,10b,11b,12b), said separating gap (13) extending from said first sidewall (9a,10a,11a,12a) to said second sidewall (9b, 10b,11b,12b),

characterized in that

said first sidewall (9a,10a, 11a,12a) has a perforation (14a) therethrough for conducting cooling air (24), said perforation (14a) being configured to eject said cooling air (24) into said separating gap (13) causing said cooling air (24) to impinge on said second sidewall (9b,10b,11b, 12b).

2. The turbine assembly (1) according to claim 1, **characterized in that** each of said first (9a,10a, 11a,12a) and second (9b,10b,11b,12b) sidewalls has multiple spaced apart perforations (14a,14b) therethrough for conducting cooling air (24), the perforations (14a, 14b) through each of said sidewalls (9a,10a,11a,12a and 9b,10b,11b,12b) being configured to eject the cooling air (24) into said separating gap (13) causing the cooling air (24) to impinge on spaces (26b,26a) between the perforations (14b,14a) on the other of said sidewalls (9b,10b,11b,12b and 9a,10a,11a, 12a).

3. The turbine assembly (1) according to claim 2, **characterized in that** the perforations (14a) through said first sidewall (9a,10a,11a,12a) and the perforations (14b) through said second sidewall (9b,10b,11b, 12b) have a staggered arrangement relative to each other.

4. The turbine assembly (1) according to any of clams 2 and 3, wherein each said segment (5a,6a,7a,8a and 5b,6b,7b,8b) further includes a first face (15a-b,16a-b,17a-b,18a-b) exposed directly to an axial flow of a turbine drive gas (25) and a second face (19a-b,20a-b,21a-b,22a-b) radially opposite to said first face (15a-b,16a-b,17a-b,18a-b), the respective sidewall (9a,10a,11a,12a and 9b,10b,11b,12b) of said segment (5a,6a,7a,8a and 5b,6b,7b,8b) being generally perpendicular to said first face (15a-b,16a-b,17a-b,18a-b) and said second face (19a-b,20a-b, 21a-b,22a-b), **characterized in that** said perforations (14a,14b) conduct cooling air (24) from a cavity defined by said second face (19a-b,20a-b,21a-b, 22a-b).

5. A method for cooling blade or vane endwalls (5,6,8) or heat shields (7) of a turbine, comprising:

- arranging a first segment (5a,6a,7a,8a) and a second segment (5b,6b,7b,8b) of said blade or vane endwalls (5,6,8) or said heat shields (7) adjacent to each other, wherein said first segment (5a,6a,7a,8a) has a first sidewall (9a,10a, 11a,12a) and said second segment (5b,6b,7b, 8b) has a second sidewall (9b,10b,11b,12b), said arrangement being made such that a separating gap (13) exists between said first sidewall (9a,10a,11a,12a) and said second sidewall (9b,10b,11b,12b),
- providing a perforation (14a) through said first sidewall (9a,10a,11a,12a),
- directing cooling air (24) through said perforation (14a) such that said cooling air (24) is ejected into said separating gap (13) to impinge on said second sidewall (9b,10b,11b,12b).

6. The method according to claim 5, further comprising

- providing multiple spaced apart perforations (14a,14b) through each of said first (9a,10a,11a, 12a) and second (9b,10b,11b,12b) sidewalls, and
- directing cooling air (24) through the perforations (14a,14b) such that said cooling air (24) is ejected said separating gap (13) to impinge on spaces (26a,26a) between the perforations (14b,14a) on the other of said sidewalls (9b,10b, 11b,12b and 9a,10a,11a,12a).

7. The method according to claim 6, comprising arranging the perforations (14a) through said first sidewall (9a,10a,11a,12a) and the perforations (14b) through said second sidewall (9b,10b,11b,12b) in a staggered manner relative to each other.

8. The method according to any of claims 6 and 7, wherein each said segment (5a,6a,7a,8a and 5b,6b, 7b,8b) further includes a first face (15a-b,16a-b,17a-b,18a-b) exposed directly to an axial flow of a turbine drive gas (25) and a second face (19a-b,20a-b,21a-b,22a-b) radially opposite to said first face (15a-b, 16a-b,17a-b,18a-b), the respective sidewall (9a, 10a,11a,12a and 9b,10b,11b,12b)of said segment (5a,6a,7a,8a and 5b,6b,7b,8b) being generally perpendicular to said first face (15a-b,16a-b,17a-b,18a-b) and said second face (19a-b,20a-b,21a-b,22a-b), wherein said method further comprises directing said cooling air (24) through said perforations (14a, 14b) from a cavity (23a,23b) defined by said second face (19a-b,20a-b,21a-b,22a-b).

FIG 1

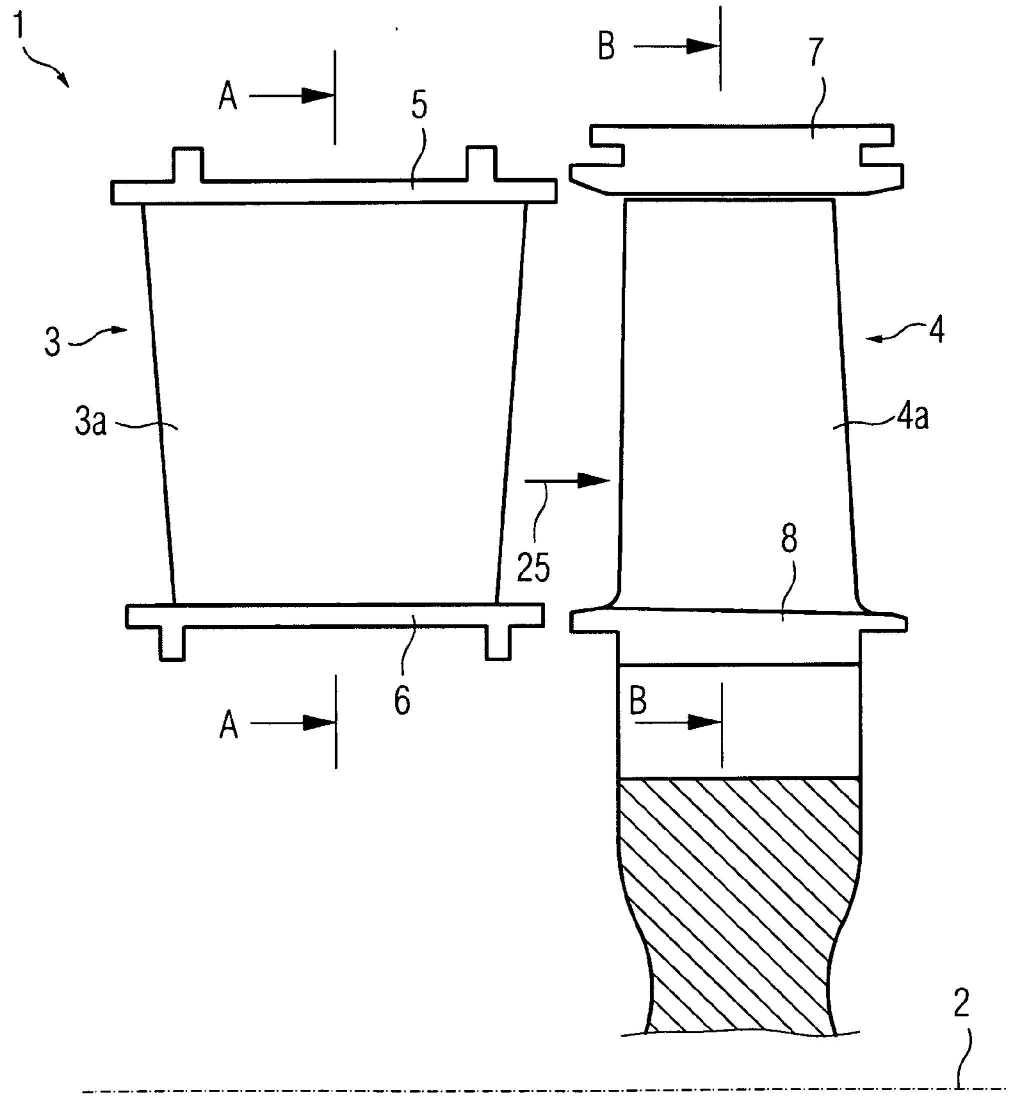
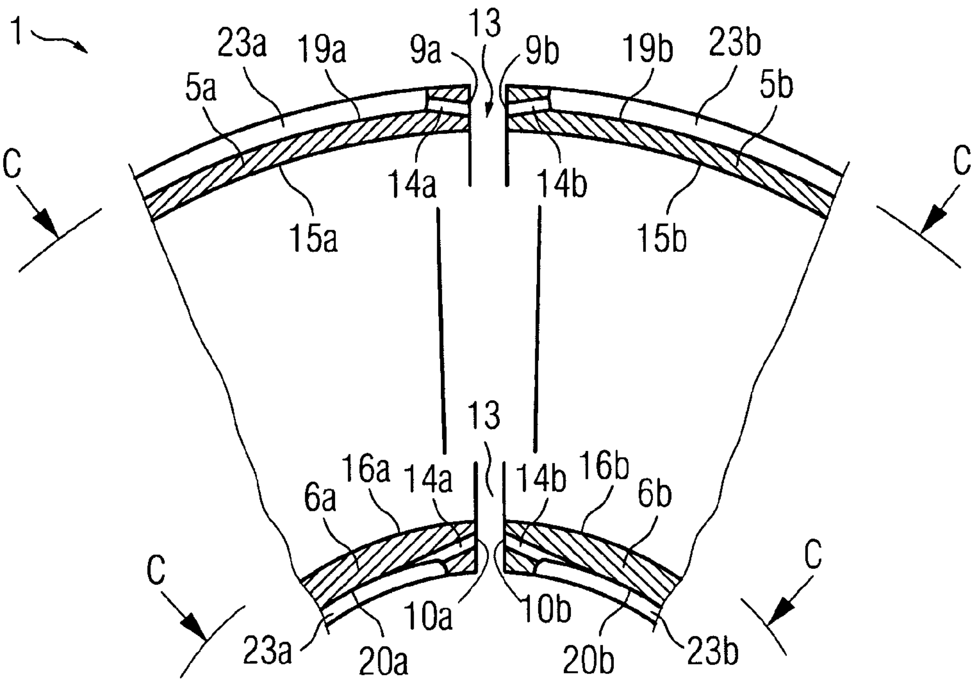
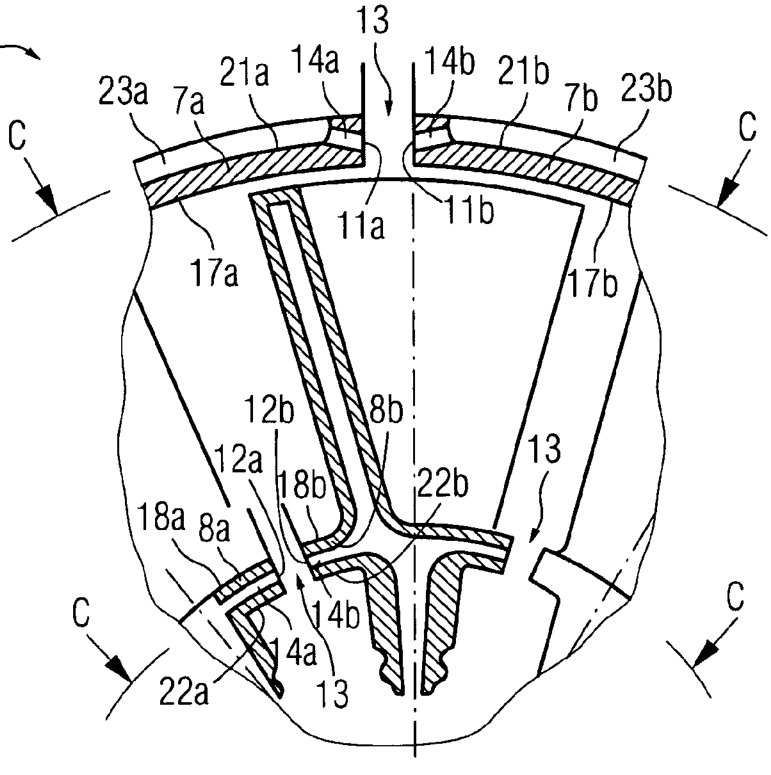


FIG 2



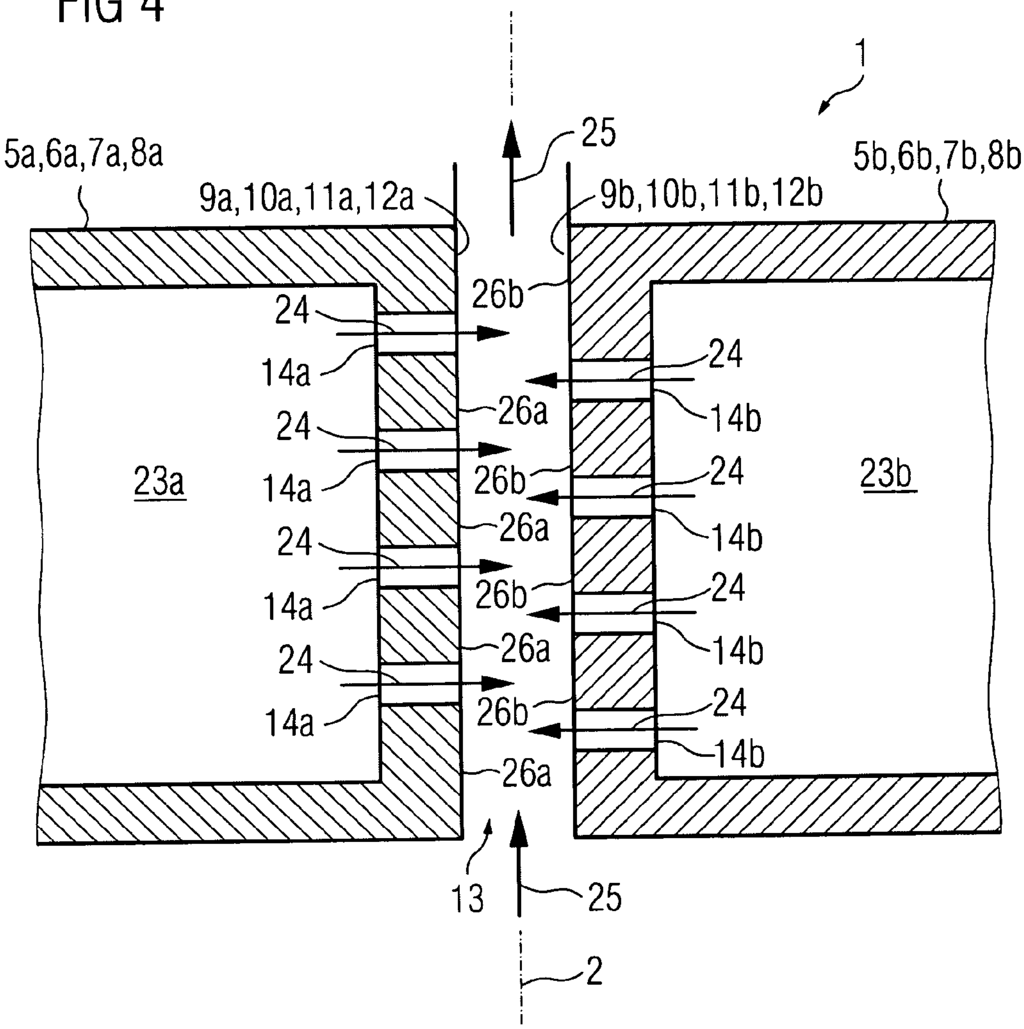
A - A

FIG 3



B - B

FIG 4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EUROPEAN SEARCH REPORT

Application Number
EP 10 00 4074

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The present search report has been drawn up for all claims

Place of search	Date of completion of the search	Examiner
Munich	14 September 2010	Rapenne, Lionel

CATEGORY OF CITED DOCUMENTS

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For more details about this annex : see Official Journal of the European Patent Office, No. 12/82

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