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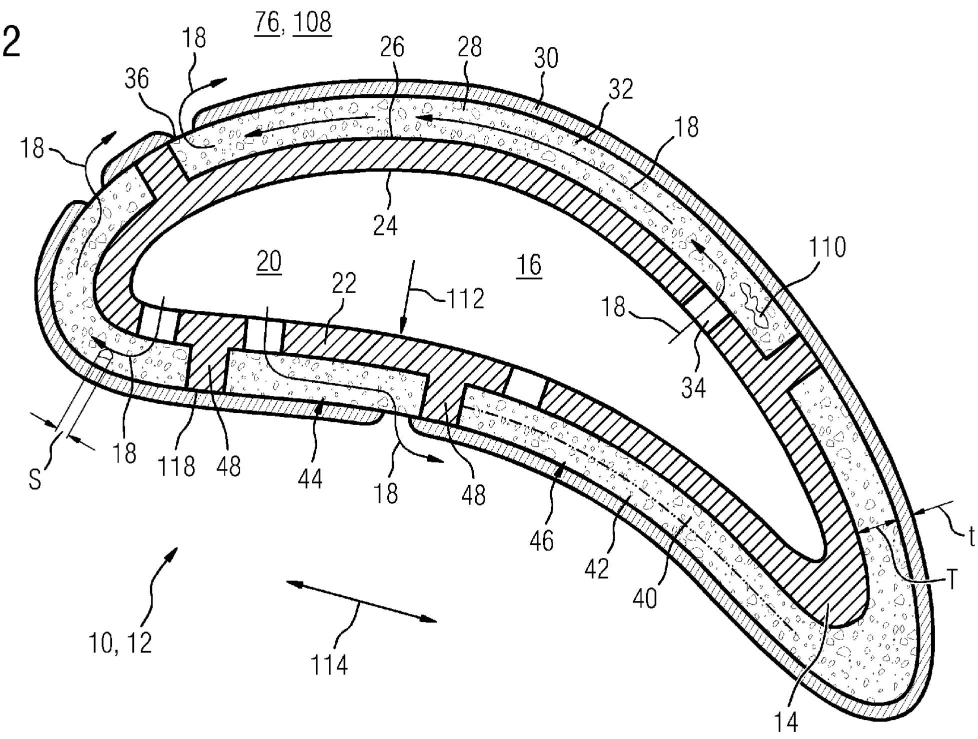
(54) COOLING CONCEPT FOR A TURBINE COMPONENT

(57) The present invention relates to a turbine component (12, 12a) comprising a base body (14, 14a) having at least one cavity (16) providing a flow path (18, 18a) for a cooling medium (20) and at least one wall (22, 22a) at least partially surrounding the at least one cavity (16) and wherein the at least one wall (22, 22a) has an inner surface (24) being oriented towards the at least one cavity (16) and an outer surface (26) being oriented opposed to the inner surface (24) of the at least one wall (22, 22a), wherein the at least one turbine component (12, 12a) further comprises at least one layer (28) of porous material and at least one thermal barrier coating (30), wherein the at least one layer (28) of porous material is arranged on the outer surface (26) of the at least one wall (22, 22a) of the base body (14, 14a) and the thermal barrier coating (30) is arranged at least partially on top of the at least one layer (28) of porous material.

Due to this a good cooling efficiency can be provided.

FIG 2



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Description

Field of the Invention

[0001] The present invention relates to a turbine component such as, for example, turbine rotor blades and stator vanes, and to a cooling concept useable in such components for cooling purposes. The present invention further relates to a method for manufacturing such a component.

Background to the Invention

[0002] Modern turbines often operate at extremely high temperatures. The effect of temperature acting on components of the turbine, like the turbine blades and/or stator vanes, can be detrimental to the efficient operation of the turbine and can, in extreme circumstances, lead to distortion and possible failure of the blade or vane. In order to overcome this risk, high temperature turbines may include hollow blades or vanes incorporating cooling channels, inserts and pedestals or other features for cooling purposes.

[0003] Internal cooling is designed to provide efficient transfer of heat between the gas washed surface and the flow of cooling air within. As heat transfer efficiency improves, for example, less cooling air may be necessary to adequately cool the aerofoils. Internal cooling typically includes structures to improve heat transfer efficiency including, for example, impingement tubes or pedestals (also known as pin fins). Moreover, the heat load acting on the aerofoil can be further reduced by adding a thermal barrier coating (TBC) on the outermost surface of the aerofoil being directly exposed to the hot gas path during operation of the turbine. However, in modern turbine systems the demand in regard to cycle efficiency is high. This results in the need to operate the turbines with even higher temperatures as today. Thus, solutions are needed that provide more efficient cooling concepts for turbine blades or vanes.

[0004] It is a first objective of the invention to provide an advantageous turbine component such as, for example, a turbine rotor blade and a stator vane with which the above-mentioned shortcomings can be mitigated, and especially, a high cooling efficiency can be realised and thus, for example, a more aerodynamic efficient component can be provided.

[0005] It is a second objective of the present invention to provide a method for manufacturing such a turbine component by which a more aerodynamic efficient turbine component and, for example, aerofoil component is facilitated.

Summary of the Invention

[0006] Accordingly, the present invention provides a turbine component comprising a base body having at least one cavity providing a flow path for a cooling medium and at least one wall at least partially surrounding the at least one cavity and wherein the at least one wall has an inner surface being oriented towards the at least one cavity and an outer surface being oriented opposed to the inner surface of the at least one wall, wherein the at least one turbine component further comprises at least one layer of porous material and at least one thermal barrier coating, wherein the at least one layer of porous material is arranged on the outer surface of the at least one wall of the base body and the thermal barrier coating is arranged at least partially on top of the at least one layer of porous material.

[0007] Due to the inventive matter the cooling efficiency can be enhanced in comparison with state of the art systems. This will enhance a component life and thus will ensure a reliable operation of a turbine engine comprising the inventive turbine component.

[0008] Due to its large relative heat transfer area a porous material is a very efficient way of creating a heat sink. If this advantage can be combined with the heat flux reduction from the gas flow to the turbine component using a thermal barrier coating an extremely efficient convective cooling system can be designed. Moreover, the arrangement of the thermal barrier coating directly on top of the layer of porous material would arrange the cooling system as close to the heat load at/in the turbine component as possible and thus avoid transport of heat through material, which would create thermal gradients in the metal. Furthermore, this arrangement provides the possibility to clean out the excess powder remaining in the structure after a for example employed additive manufacturing process. This also improves the reliability in the manufacturing process. Further, a lower amount of cooling medium may be needed in comparison with state of the art systems. Hence, the aerodynamic efficiency will be increased. Additionally, advantageous thermodynamics of the cycle can be realized. Due to the saving of cooling medium, it is possible to achieve higher "mixed" temperatures using a lower temperature in the flame of the combustor. This, in turn, decreases the NO_X emissions of the turbine.

[0009] Even if a chosen term is used in the singular or in a specific numeral form in the claims and the specification the scope of the patent (application) should not be restricted to the singular or the specific numeral form. It should also lie in the scope of the invention to have more than one or a plurality of the specific structure(s).

[0010] A turbine component is intended to mean a component provided for a turbine, like a gas turbine, like, for example, an aerofoil component embodied as a blade or vane or a part thereof, like an aerofoil, a root portion or an inner or outer platform. The turbine component may further be embodied as a turbine heatshield, a ring segment or a part of a combustor of the turbine, wherein such a combustor part is subjected to high temperatures (a so-called hot part). The turbine component may be a part of a turbine assembly.

[0011] Preferably, the turbine assembly may have a turbine cascade and/or wheel with circumferential arranged aerofoils. Moreover, the turbine component may further comprise an outer and an inner platform arranged at opposed ends of the aerofoil(s) or a shroud and a root portion arranged at opponent ends of the aerofoil(s). In case of the embodiment of the turbine component as an aerofoil it may be a basically hollow aerofoil, wherein a "basically hollow aerofoil" means an aerofoil with a casing, wherein the casing encases at least one cavity. A structure, like a rib, rail or partition, which divides different cavities in the aerofoil from one another and for example extends in a span wise direction of the aerofoil, does not hinder the definition of "a basically hollow aerofoil". Preferably, the aerofoil is hollow.

[0012] In this context a "base body" is intended to mean a structure that substantially imparts a shape and/or form of the turbine component or, for example, the aerofoil component. The cavity may, for example, be the cavity or be at least one of the cavities in a - hollow - aerofoil or a channel (channel system) for cooling medium in a root portion, a platform, a heatshield, a ring segment or a hot part. Further, a wall that "at least partially surrounds the at least one cavity" is intended to mean that at least one side of the cavity (channel) is restricted or closed off from an exterior, like a gas path, during operation of the turbine engine, by at least a section of the wall. The wall may, for example, be a wall building the leading or trailing edge of an aerofoil, a part of a suction or pressure side of an aerofoil or a wall of a root portion, platform, ring segment, heatshield or hot part.

[0013] The phrases that "the inner surface is oriented towards the at least one cavity" and that "the outer surface is oriented opposed to the inner surface" should be understood that the inner surface is facing the interior of the cavity (channel) and the outer surface is oriented towards an exterior of the turbine component or aerofoil component, like the gas path during operation. Moreover, the phrase "arranged on" should be also understood as "arranged at". Further, due to the described arrangement the at least one layer of porous material is arranged between the outer surface of the at least one wall of the base body and the at least one thermal barrier coating. The issue that "the thermal barrier coating is arranged at least partially on top of the at least one thermal barrier coating" should be understood in that there might be regions where the at least one layer of porous material is uncovered or the at least one thermal barrier coting has holes.

[0014] Furthermore, the term "porous material" should be understood in that the material has a channel system comprising or being out of communicating pores, wherein the channel system is embodied in such a way so that a flow medium, like a cooling medium or cooling air, can flow through the channel system or its communicating pores and thus through the layer of porous material.

[0015] In this context, a thermal barrier coating (TBC) is intended to mean a coating out of a material that has a low heat (thermal) conductivity, wherein "low heat conductivity" should be understood in that the heat conductivity of the TBC material is below 5 W/m/K and preferably even below 3 W/m/K. In comparison, the material of the porous layer should be a material having a high heat conductivity (higher than 7.5 W/m/K and preferably higher than 10 W/m/K. Hence, the basic property difference between the material of the TBC and that of the porous layer is the high heat conductivity for the base material of the base body and the porous layer, and the low conductivity of the TBC.

[0016] Furthermore, the at least one layer of porous material comprises a plurality of pores. Hence, the convective surface can be easily enhanced ensuring proper cooling. Hence, a lower total cooling air consumption in respect to state of the art systems can be realised. A pore size of each pore of the plurality of pores may be any size feasible for a person skilled in the art. The pore size may be between 0.005 millimetre (mm) and 2 mm, preferably between 0.0075 mm and 1 mm and most preferably, a majority of pores have a pore size between 0.01 mm and 0.5 mm. With these pore sizes a balance between a sufficient cooling surface and unrestricted flow path for the cooling medium can be provided. The mentioned limiting values should be integrated into the claimed range of pore sizes. In other words, a mean grain size distribution or mean pore size distribution, respectively, is between 0.01 mm and 0.5 mm. Each pore may have the same pore size (homogeneous pore size) or the pore sizes of the pores may be different from one another or may vary (inhomogeneous pore size).

[0017] A thickness of the at least one layer of porous material may be any thickness feasible for a person skilled in the art. The thickness may be between 0.1 mm and 10 mm, preferably between 0.5 mm and 7.5 mm and most preferably the at least one layer of porous material has a thickness between 1 mm and 5 mm. With such a layer thickness an efficient cooling can be provided. The mentioned limiting values should be integrated into the claimed range of the thickness. Moreover, the thickness of the at least one thermal barrier coating may be any thickness feasible for a person skilled in the art. The thickness may be between 0.05 mm and 5 mm, preferably between 0.1 mm and 2.5 mm and most preferably the at least one thermal barrier coating has a thickness between 0.2 mm and 1 mm. Consequently, structures, like the layer of porous material, beneath the thermal barrier coating can be protected easily against the hot temperature of the gas path. The mentioned limiting values should be integrated into the claimed range of the thickness.

[0018] A material of the at least one layer of porous material may be any material or material combination or alloy feasible for a person skilled in the art. Needed properties of the material are, for example, oxidation resistance at a high temperature and advanced mechanic properties, such as creep or thermo mechanic fatigue properties at high temperatures. Such super alloys may, for example, be <u>Hastelloy,</u> <u>Inconel,</u> <u>Waspaloy,</u> <u>Rene al-</u>

loys, Haynes alloys. The material of the at least one layer of porous material may be selected out of the group consisting of: nickel based super alloy, Inconel (In) 1792, In 939, In 738, Hastelloy or Alloy 247. With these materials a wide range of useable materials are available. This provides the possibility to choose the material which provides the needed properties for each specific application.

[0019] According to a preferred realisation of the invention the material of the at least one layer of porous material is the same material as a material of the at least one wall of the base body. In other words, the material of the at least one layer of porous material is of the same material, for example a metallic alloy material, as the internal load carrying structure. Hence, both components advantageously have the same properties and characteristics. A stable connection between the layer of porous material and the wall and thus the aerofoil component can be provided when the at least one layer of porous material is formed integrally with the at least one wall of the base body. Furthermore, the layer of porous material and the wall can be manufactured in one processing sequence or step. In this context, the wording "formed integrally" is intended to mean, that the layer of porous material and the wall are moulded out of one piece and/or that the layer of porous material and the wall could only be separate with loss of function for at least one of the parts.

[0020] A material of the at least one thermal barrier coating may be any material or material combination or alloy feasible for a person skilled in the art. Basically, the material of the at least one thermal barrier coating is a low conducting material with high oxidation resistance. By usage of such (a) material(s) the turbine component can be effectively protected from the detrimental effects of the operating environment. The material of the at least one thermal barrier coating may be, for example, a material selected out of the group consisting of: yttria-stabilized zirconia (YSZ), Mullite and Ceria. The list shows that a wide range of different materials can be used. Consequently, the most feasible can be selected in regard of the required necessities.

[0021] According to a further aspect of the invention the at least one wall of the base body may comprise at least one passage or aperture to allow the cooling medium to enter or flow into the at least one layer of porous material from the cavity or channel. Thus, an easy access can be realised. The cooling medium enters or flows - originating from the cavity - through the passage into the layer of porous material or its pores or channel system out of pores, respectively. Hence, the passage extends between the cavity and the at least one layer of porous material or "connects" the cavity with the layer of porous material. Moreover, the flow path of the cooling medium comprises a section, which is located downstream of the cavity or starts at the passage.

[0022] Consequently, the downstream section of the flow path starts at the passage of the at least one wall of the base body positioned or extends between the cavity and the at least one layer of porous material. In other words, the flow path starts in respect to the layer of porous material at the passage. Moreover, the flow path of the cooling medium terminates at at least one exit hole of the turbine component for the cooling medium. This can be done constructively easy when the exit hole is positioned at the at least one porous layer or at the at least one thermal barrier coating.

[0023] In a further realisation of the invention the at least one turbine component and/or the layer of porous material and/or the at least one wall of the base body comprises at least one chamber to collect the cooling medium. Preferably, the at least one chamber is positioned at an arbitrarily position along the flow path. Hence, the collection and the subsequent discharge of the cooling medium can be adjusted to cooling needs of the turbine component. A use of a chamber would be especially beneficial in such cases where, for example, a pressure in the gas path is too high, for example for the cooling medium to exit the porous material. In such a scenario the cooling medium can be collected by the chamber and can be ejected somewhere else where the pressure is low enough. In such a case the inlet and outlet of course have to be separate channels, otherwise there would be no pressure difference driving the flow. The chamber may also be positioned as a closed off chamber in respect to the cavity inside the cavity. Hence, an exit channel to connect the chamber with the outside of the turbine component may be provided and may, for example, extend through the at least one wall of the base body and/or through the at least one layer of porous material.

[0024] Different cooling requirements of different parts or areas of the turbine component, like a leading edge, a trailing edge, a suction side or a pressure side of the aerofoil, can be provided when the at least one layer of porous material comprises at least two sublayers varying in pore size and/or material composition and/or thickness and/or at least two sections varying in pore size and/or material composition and/or thickness. Additionally, the cooling system can be designed easier since there is a means to adjust the cooling medium distribution by choosing the thickness and porosity in the layer of porous material or its sections/sublayers.

[0025] In this context the subject of "two sublayer" is intended to mean that two layers are positioned one after the other in a direction pointing from the cavity through the at least one layer of porous material to an outside of the turbine component and the subject of "two section" is intended to mean that two sections are positioned one after the other in a direction basically perpendicular to the above mentioned direction or in case of an aerofoil component from the leading edge to the trailing edge (chord-wise direction) or in a span wise direction of the turbine component. Moreover, each section may also comprise at least two sublayers. Hence, the at least one layer of porous material may be composed and/or manufactured out of several different materials.

[0026] According to an advantageous embodiment of the invention the at least one wall of the base body comprises at least one rib extending into the at least one layer of porous material. Due to this the at least one porous layer can be divided constructively easily. The phrase "extending into" should be understood as oriented in the direction pointing from the cavity through the at least one layer of porous material to an outside of the turbine component and preferably an orientation of the rib basically perpendicular to the at least one wall of the base body, wherein in the scope of an arrangement as "basically perpendicular" should also lie a divergence of the orientations of the rib and the wall of about 30°.

[0027] The at least one porous layer can be sectioned in independent sections when the at least one rib is going through the at least one layer of porous material. In this context, the phrase "going through" should be understood as having no porous material on top of the rib (a surface of the rib located opposed to a part of the rib that is connected to the wall). In other words, an outer surface of the rib is in "contact" with the outside of the turbine component or is covered by the at least one thermal barrier coating.

[0028] Beneficially, the at least one thermal barrier coating comprises at least one exit hole for the cooling medium, wherein the at least one exit hole extends through the at least one thermal barrier coating. This provides an easy discharge of the cooling medium from the at least one layer of porous material or its pores. Preferably, the at least one exit hole is embodied as a film cooling hole. The exit hole or film cooling hole may be manufactured, for example, by drilling the hole into the thermal barrier coating after coating the porous layer with the TBC or the hole may be established during the coating process by leaving a gap in the coating. Properties and characteristics of the film cooling hole, like size, shape, orientation, incline etc., are the same as for film cooling holes known in the art. Thus, the cooling medium exits the at least one layer of porous material through at least one film cooling hole in the at least one thermal barrier coating.

[0029] As stated above, at least one layer of porous material comprises a plurality of pores and according to a further aspect of the invention the at least one passage in the at least one wall - that provides entry for the cooling medium into the at least one layer of porous material from the cavity - and the at least one exit hole for the cooling medium - extending through the at least one thermal barrier coating - are in flow communication with each other solely via the layer of porous material, and more specifically, via the plurality of flow communicating pores of the porous material. Hence, an efficient convective cooling can be provided.

[0030] In a further advantageous embodiment the turbine component is a turbine blade or stator vane, for example, a nozzle guide vane, or a part thereof, like an aerofoil, a root portion or a platform.

[0031] The present invention also provides a method for manufacturing the beforehand described turbine component. The method comprises at least the step of: manufacturing the at least one layer of porous material by additive manufacturing.

[0032] Due to the inventive method a turbine component can be provided that has an increased cooling efficiency in comparison with state of the art systems. Moreover, a reliable manufacturing process can be provided.

[0033] It is further provided that the method comprises the following steps: manufacturing the base body and the at least one layer of porous material by additive manufacturing using powder, removing the residual powder out of pores of the at least one layer of porous material and adding the at least one thermal barrier coating on top of the at least one layer of porous material.

[0034] The arrangement of the thermal barrier coating directly on top of the layer of porous material provides the possibility to clean out the excess powder remaining in the structure after, for example, an employed additive manufacturing process before the TBC is added. This results in a more robust manufacturing process in comparison with state of the art systems.

[0035] The addition of the at least one thermal barrier coating may be done by any method feasible for a person skilled in the art. Preferably, the at least one thermal barrier coating may be applied by electron beam-physical vapour deposition (EB-PVD) or atmospheric plasma spraying. Thus, established methods can be applied.

[0036] The previously given description of advantageous embodiments of the invention contains numerous features which are partially combined with one another in the dependent claims. Expediently, these features can also be considered individually and be combined with one another into further suitable combinations. Furthermore, features of the method, formulated as apparatus features, may be considered as features of the assembly and, accordingly, features of the assembly, formulated as process features, may be considered as features of the method.

[0037] The above-described characteristics, features and advantages of the invention and the manner in which they are achieved can be understood more clearly in connection with the following description of exemplary embodiments which will be explained with reference to the drawings. The exemplary embodiments are intended to illustrate the invention, but are not supposed to restrict the scope of the invention to combinations of features given therein, neither with regard to functional features. Furthermore, suitable features of each of the exemplary embodiments can also be explicitly considered in isolation, be removed from one of the exemplary embodiments, be introduced into another of the exemplary embodiments and/or be combined with any of the appended claims.

Brief Description of the Drawings

[0038] The present invention will be described with reference to drawings in which:

FIG 1: shows a schematically and sectional view of a gas turbine engine comprising several inventive turbine components,

FIG 2: shows a sectional view of a turbine component of FIG 1 and,

FIG 3: shows a sectional view of an alternatively embodied turbine component.

Detailed Description of the Illustrated Embodiments

[0039] The present invention is described with reference to an exemplary turbine engine 50 having a single shaft 62 or spool connecting a single, multi-stage compressor section 54 and a single, one or more stage turbine section 58. However, it should be appreciated that the present invention is equally applicable to two or three shaft engines and which can be used for industrial, aero or marine applications.

[0040] The terms upstream and downstream refer to the flow direction of the airflow and/or working gas flow through the engine 50 unless otherwise stated. If used, the terms axial, radial and circumferential are made with reference to a rotational axis 60 of the engine 50.

[0041] FIG 1 shows an example of a gas turbine engine 50 in a sectional view. The gas turbine engine 50 comprises, in flow series, an inlet 52, a compressor section 54, a combustion section 56 and a turbine section 58, which are generally arranged in flow series and generally in the direction of a longitudinal or rotational axis 60. The gas turbine engine 50 further comprises a shaft 62 which is rotatable about the rotational axis 60 and which extends longitudinally through the gas turbine engine 50. The shaft 62 drivingly connects the turbine section 58 to the compressor section 54.

[0042] In operation of the gas turbine engine 50, air 64 which is taken in through the air inlet 52 is compressed by the compressor section 54 and delivered to the combustion section or burner section 56. The burner section 56 comprises a burner plenum 66, one or more combustion chambers 68 defined by a double wall can 70 and at least one burner 72 fixed to each combustion chamber 68. The at least one combustion chambers 68 and the burners 72 are located inside the burner plenum 66. The compressed air passing through the compressor section 54 enters a diffuser 74 and is discharged from the diffuser 74 into the burner plenum 66 from where a portion of the air enters the burner 72 and is mixed with a gaseous or liquid fuel. The air/fuel mixture is then burned and the combustion gas 76 or working gas from the combustion is channelled via a transition duct 78 to the turbine section 58.

[0043] This exemplary gas turbine engine 50 has a cannular combustor section arrangement 80, which is constituted by an annular array of combustor cans 70 each having the burner 72 and the combustion chamber 68, the transition duct 78 has a generally circular inlet that interfaces with the combustion chamber 68 and an outlet in the form of an annular segment. An annular array of transition duct outlets form an annulus for channelling the combustion gases to the turbine section 58.

[0044] The turbine section 58 comprises a number of turbine assemblies 10 embodied as blade carrying discs 82 or turbine wheels 84 attached to the shaft 62. In the present example, the turbine section 58 comprises two discs 82 each carry an annular array of turbine components 12 or aerofoil components 12, embodied as a turbine blade, which each comprises several sub-components, like an aerofoil or a root portion. However, the number of blade carrying discs 82 could be different, i.e. only one disc 82 or more than two discs 82. In addition, turbine assemblies 10 embodied as turbine cascades 86 are disposed between the turbine blades. Each turbine cascade 86 carries an annular array of turbine components 12 or aerofoil components 12, which each comprises an aerofoil in the form of guiding vanes, which are fixed to a stator 88 of the gas turbine engine 50, and an inner and outer platform. Between the exit of the combustion chamber 56 and the leading turbine blades inlet guiding vanes or nozzle guide vanes 90 are provided. They turn and accelerate the flow of working gas 76 onto the turbine blades.

[0045] The combustion gas 76 from the combustion chamber 68 enters the turbine section 58 and drives the turbine blades which in turn rotate the shaft 62. The guiding vanes 90 serve to optimise the angle of the combustion or working gas 76 on to the turbine blades. The turbine section 58 drives the compressor section 54. The compressor section 54 comprises an axial series of turbine assemblies 10 embodied as guide vane stages 92 and rotor blade stages 94. The rotor blade stages 94 comprise a rotor disc 82 supporting turbine components 12 or aerofoil components 12 or an annular array of turbine blades.

[0046] The compressor section 54 also comprises a stationary casing 96 that surrounds the rotor stages 94 in circumferential direction 98 and supports the vane stages 92. The guide vane stages 92 include an annular array of radially extending turbine components 12 or aerofoil components 12 embodied as vanes that are mounted to the casing 96. The vanes are provided to present gas flow at an optimal angle for the blades at a given engine operational point. Some of the guide vane stages 92 have variable vanes, where the angle of the vanes, about their own longitudinal axis, can be adjusted for angle according to air flow characteristics that can occur at different engine operations conditions.

[0047] The casing 96 defines a radially outer surface 100 of a passage 102 of the compressor section 54. A radially inner surface 104 of the passage 102 is at least partly defined by a rotor drum 106 of the rotor which is partly defined by the annular array of blades.

[0048] FIG 2 shows a sectional view of a turbine component 12 or aerofoil component 12 embodied in this exemplary embodiment as a vane of the gas turbine engine 50. The turbine component may also be a heatshield, a ring segment or a hot part of the combustor section (not shown). The turbine component 12 or aerofoil component 12 is embodied as a basically hollow aerofoil with a base body 14 having in this exemplary embodiment one cavity 16 providing a flow path 18 for a cooling medium 20. The base body 14 further comprises a wall 22 surrounding the cavity 16. Moreover, the wall 22 has an inner surface 24 being oriented towards or facing the cavity 16 and an outer surface 26 being oriented opposed to the inner surface 24 of the wall 22 or towards a gas path 108 of the combustion gas 76.

[0049] To provide an efficient cooling of the turbine component 12 or aerofoil component 12 it further comprises a layer 28 of porous material and a thermal barrier coating 30. The layer 28 of porous material is arranged on the outer surface 26 of the wall 22 of the base body 14 and the thermal barrier coating 30, in turn, is arranged at least partially on top of the layer 28 of porous material. In other words, the layer 28 of porous material is arranges between the wall 22 and the thermal barrier coating 30. The layer 28 of porous material has a thickness T being between 1 mm and 5 mm and the thermal barrier coating 30 has a thickness t being between 0.2 mm and 1 mm (dimensions in FIG 2 are not shown true to scale).

[0050] The material of the layer 28 of porous material is a material selected out of the group consisting of: nickel based super alloy, and is for example Inconel (In) 1792, In 939, In 738, Hastelloy or Alloy 247. In this exemplarily and preferred embodiment the material of the layer 28 of porous material is the same material as a material of the wall 22 of the base body 14 and is formed integrally with the wall 22 of the base body 14 (An hatching for the metal material of the porous layer 28 is not depicted in FIG 2.). Moreover, the thermal barrier coating is, for example, yttria-stabilized zirconia (YSZ).

[0051] The layer 28 of porous material comprises a plurality of pores 32 forming a channel system 110 so that the cooling medium 20 can flow through the channel system 110 or its communicating pores 32 and thus through the layer 28 of porous material. The channel system 110 is only shown in FIG 2 symbolically for three pores 32. A majority of the pores 32 have a pore size S between 0.01 mm and 0.5 mm (again dimensions in FIG 2 are not shown true to scale). The pore size S of each pore 32 may be different, as shown in this exemplary embodiment, or the pores may all have the same pore size (not shown).

[0052] To allow the cooling medium 20 to enter the layer 28 of porous material or its pores 32 from the cavity 16 the wall 22 of the base body 14 comprises several passages 34. At each passage 34 start - at least in respect to the layer 28 of porous material - a section 33 or portion of the flow path 18 of the cooling medium 20, wherein the section is arranged downstream of the cavity 16. Furthermore, the flow path 18 terminates at at least one exit hole 36 of the turbine component 12 or aerofoil component 12 for the cooling medium 20. Each passage 34 and the respective exit hole(s) 36 for the cooling medium 20 are in flow communication with each other solely via the plurality of flow communicating pores 32 of the porous material. The exit holes 36 represent uncovered areas of the layer 28 of porous material (not covered by the TBC). In other words, the thermal barrier coating 30 comprises the exit hole(s) 36 for the cooling medium 20, wherein the exit hole 36 extends through the thermal barrier coating 30. The exit hole(s) 36 is/are embodied as (a) film cooling hole(s). By traveling the channel system 110 of pores 32 of the layer 28 of porous material the cooling medium 20 can provide an effective convective cooling of the turbine component 12 or aerofoil component 12.

[0053] The layer 28 of porous material or sections 46 thereof may comprises in selected parts sublayers 40, 42. The sublayers 40, 42 are, for example, positioned one after the other in a direction 112 pointing from the cavity 16 through the layer 28 of porous material to an outside (gas path 108) of the turbine component 12. These sublayers 40, 42 may vary in pore size S and/or material composition and/or thickness T. In FIG 2 two sublayers 40, 42 are symbolically shown by the broken line, wherein for better representability the pattern of pores 32 are the same for both sublayers 40, 42.

[0054] Moreover, the layer 28 of porous material may comprise several sections 44, 46 (only two of the five sections shown in FIG 2 are marked with reference numerals). These sections 44, 46 are positioned one after the other around the outer surface 26 of the turbine component 12 or aerofoil component 12, for example, in chord-wise direction 114 or in a span wise direction 116 of the turbine component 12 (see FIG 1). The sections 44, 46 may also vary in pore size S and/or material composition and/or thickness T. Each section 44, 46, comprising a passage 34 from the cavity 16, a part of the layer 28 of the porous material and an exit hole 36, may be a separate sub-system. Such a sub-system needs to be designed to match the local heat load from the outside, supply pressure, temperature of the cooling medium 20 and pressure at the place where the cooling medium 20 exits.

[0055] The sections 44, 46 are separated from one another by a rib 48 of the wall 22 of the base body 14. Thus, the rib 48 extends into and is going through the layer 28 of porous material. An outer surface 118 of the rib 48 is covered by the thermal barrier coating 30.

[0056] A manufacturing sequence of the turbine component 12 comprises the following step:

- manufacturing the base body 14 and the layer 28 of porous material by additive manufacturing using powder.

[0057] In short, material is added as powder over a complete area. Then the powder is melted (welded) at the places where the wanted structure (wall, 22, layer 28 etc.) should be formed. The turbine component 12 or aerofoil component 12 is built up by adding layer by layer, the welding pattern is individual for each layer to create the wanted shape. When the process is finished a solid turbine component 12 or aerofoil component 12 embedded in powder where it was not melted is built.

[0058] In a subsequent step the residual powder is removed out of the pores 32 of the layer 28 of porous material, for example by washing. In the next step the thermal barrier coating 30 is added on top of the layer 28 of porous material, for example, by electron beam-physical vapour deposition (EB-PVD) or atmospheric plasma spraying.

[0059] In FIG 3 an alternative embodiment of the turbine component 12 is shown. Components, features and functions that remain identical are in principle substantially denoted by the same reference characters. To distinguish between the embodiments, however, the letter "a" has been added to the different reference characters of the embodiment in FIG 1 and 2. The following description is confined substantially to the differences from the embodiment in FIG 1 and 2, wherein with regard to components, features and functions that remain identical reference may be made to the description of the embodiment in FIG 1 and 2.

[0060] FIG 3 shows an alternative turbine component 12 of a turbine assembly 10a. The embodiment from FIG 3 differs in regard to the embodiment according to FIG 1 and 2 in that the flow path 18a of the cooling medium 20 comprises a chamber 38 to collect the cooling medium 20 before it exits the turbine component 12a or aerofoil component 12a, respectively. The flow path 18a comprises a section 33a, which is located downstream of the cavity 16 or starts at a passage 34 in the wall 22a of the base body 14a. Thus, the section 33a of the flow path 18a of the cooling medium 20 starts at the passage 34 and terminates at an exit hole 36 (e.g. a film cooling hole) for the cooling medium 20 of the turbine component 12a or aerofoil component 12a or the thermal barrier coating 30, respectively. The chamber 38 is positioned at an arbitrarily position along the flow path 18a in section 33a. In this exemplary embodiment the chamber 38 is a subcavity of cavity 16. To allow the cooling medium 20 to enter the chamber 38 from the layer 28 of porous material the wall 22a comprises an entry channel 122 positioned between the layer 28 of porous material and the chamber 38. Moreover, to allow the cooling medium 20 to exit the turbine component 12a or aerofoil component 12a the wall 22a comprises an exit channel 122 arranged between or connecting the chamber 38 and one exit hole 36 and spanning through the wall 22a and the layer 28 of porous material. In other words, the flow path 18a starts in respect to the layer 28 of porous material at the passage 34 and ends at the exit hole 36 located downstream to the chamber 38.

[0061] Alternatively, the chamber may be located in the layer of porous material itself (not shown). Moreover, the cooling medium may be collected inside the cavity and may be ejected either at some other part of the turbine component or extracted outside the turbine engine for cooling and increasing the pressure in a closed loop (not shown).

[0062] It should be noted that the term "comprising" does not exclude other elements or steps and "a" or "an" does not exclude a plurality. Also elements described in association with different embodiments may be combined. It should also be noted that reference signs in the claims should not be construed as limiting the scope of the claims.

[0063] Although the invention is illustrated and described in detail by the preferred embodiments, the invention is not limited by the examples disclosed, and other variations can be derived therefrom by a person skilled in the art without departing from the scope of the invention.

Claims

1. A turbine component (12, 12a) comprising a base body (14, 14a) having at least one cavity (16) providing a flow path (18, 18a) for a cooling medium (20) and at least one wall (22, 22a) at least partially surrounding the at least one cavity (16) and wherein the at least one wall (22, 22a) has an inner surface (24) being oriented towards the at least one cavity (16) and an outer surface (26) being oriented opposed to the inner surface (24) of the at least one wall (22, 22a),
wherein the at least one turbine component (12, 12a) further comprises at least one layer (28) of porous material and at least one thermal barrier coating (30), wherein the at least one layer (28) of porous material is arranged on the outer surface (26) of the at least one wall (22, 22a) of the base body (14, 14a) and the thermal barrier coating (30) is arranged at least partially on top of the at least one layer (28) of porous material.

2. Turbine component according to claim 1,
wherein the at least one layer (28) of porous material comprises a plurality of pores (32) and/or wherein a majority of pores (32) have a pore size (S) between 0.01 mm and 0.5 mm.

3. Turbine component according to claim 1 or 2,
wherein the at least one layer (28) of porous material has a thickness (T) being between 1 mm and 5 mm and/or wherein the at least one thermal barrier coating (30) has a thickness (t) being between 0.2 mm and 1 mm.

4. Turbine component according to any one of the preceding claims,
wherein the material of the at least one layer (28) of porous material is a material selected out of the group consisting of: nickel based super alloy, Inconel

(In) 1792, In 939, In 738, Hastelloy or Alloy 247.

5. Turbine component according to any one of the preceding claims, wherein the material of the at least one layer (28) of porous material is the same material as a material of the at least one wall (22, 22a) of the base body (14, 14a).

6. Turbine component according to any one of the preceding claims, wherein the at least one layer (28) of porous material is formed integrally with the at least one wall (22, 22a) of the base body (14, 14a).

7. Turbine component according to any one of the preceding claims, wherein the at least one wall (22, 22a) of the base body (14, 14a) comprises at least one passage (34) to allow the cooling medium (20) to enter the at least one layer (28) of porous material from the cavity (16).

8. Turbine component according to any one of the preceding claims, wherein the flow path (18, 18a) of the cooling medium (20) comprises at least one section (33, 33a), which starts at a passage (34) in the at least one wall (22, 22a) of the base body (14, 14a) and terminates at at least one exit hole (36) of the at least one turbine component (12, 12a) for the cooling medium (20) and/or wherein the at least one turbine component (12a) comprises at least one chamber (38) to collect the cooling medium (20) and/or wherein the at least one chamber (38) is positioned at an arbitrarily position along the flow path (18a) in section (33a).

9. Turbine component according to any one of the preceding claims, wherein the at least one layer (28) of porous material comprises at least two sublayers (40, 42) varying in pore size (S) and/or material composition and/or thickness (T) and/or at least two sections (44, 46) varying in pore size (S) and/or material composition and/or thickness (T).

10. Turbine component according to any one of the preceding claims, wherein the at least one wall (22, 22a) of the base body (14, 14a) comprises at least one rib (48) extending into the at least one layer (28) of porous material.

11. Turbine component according to any one of the preceding claims, wherein the at least one wall (22, 22a) of the base body (14, 14a) comprises at least one rib (48) going through the at least one layer (28) of porous material.

12. Turbine component according to any one of the preceding claims, wherein the at least one thermal barrier coating (30) comprises at least one exit hole (36) for the cooling medium (20), wherein the at least one exit hole (36) extends through the at least one thermal barrier coating (30) and/or wherein the at least one exit hole (36) is embodied as a film cooling hole.

13. Turbine component according to claims 7 and 12, wherein the at least one layer (28) of porous material comprises a plurality of pores (32) and wherein the at least one passage (34) in the at least one wall (22, 22a) and the at least one exit hole (36) for the cooling medium (20) are in flow communication with each other solely via the plurality of flow communicating pores (32) of the porous material.

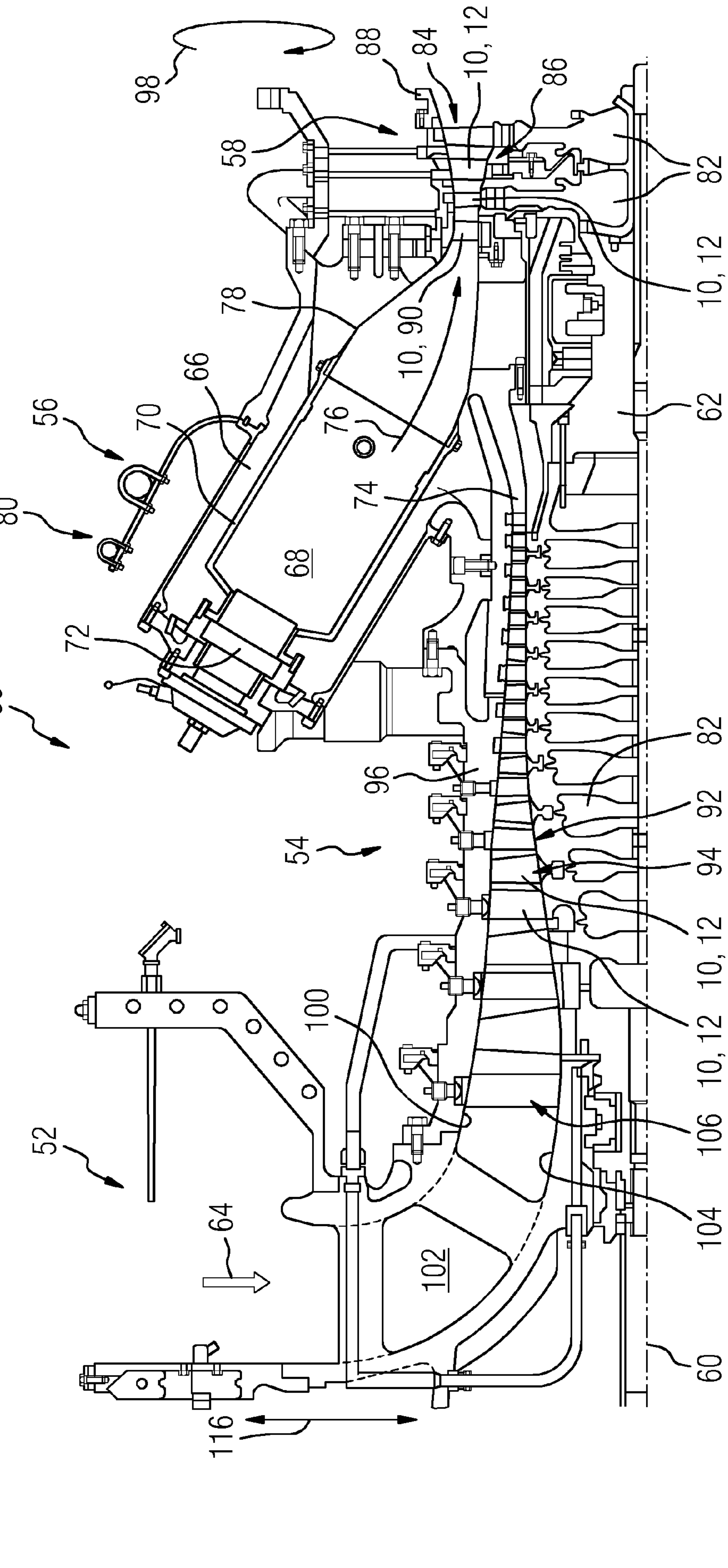
14. Method for manufacturing the turbine component (12, 12a) according to at least one of claims 1 to 13, **characterised in** the step of:

- manufacturing the at least one layer (28) of porous material by additive manufacturing.

15. Method according to claim 14, wherein the method comprises the following steps:

- manufacturing the base body (14, 14a) and the at least one layer (28) of porous material by additive manufacturing using powder,
- removing the residual powder out of pores (32) of the at least one layer (28) of porous material and
- adding the at least one thermal barrier coating (30) on top of the at least one layer (28) of porous material.

FIG 1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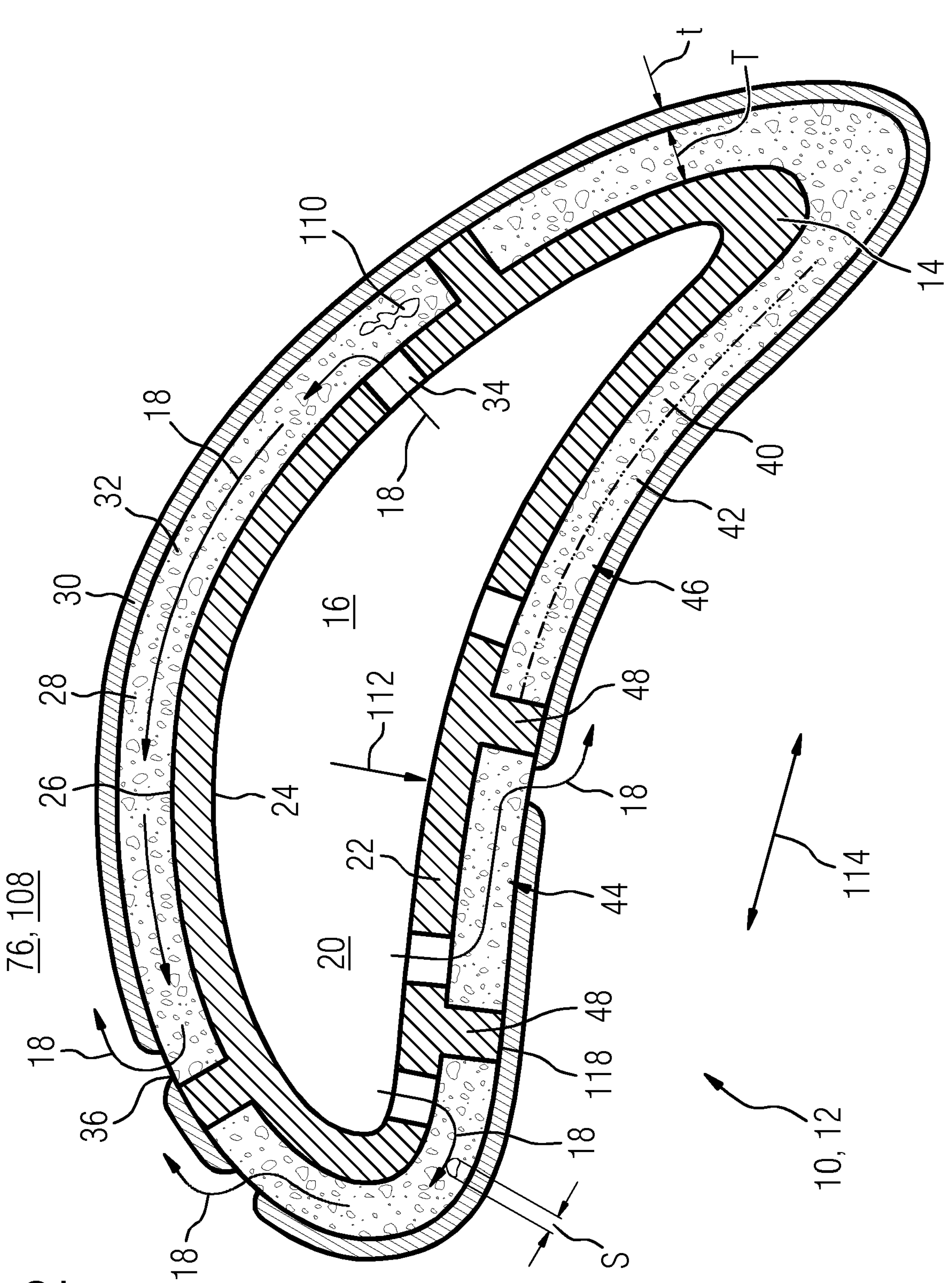
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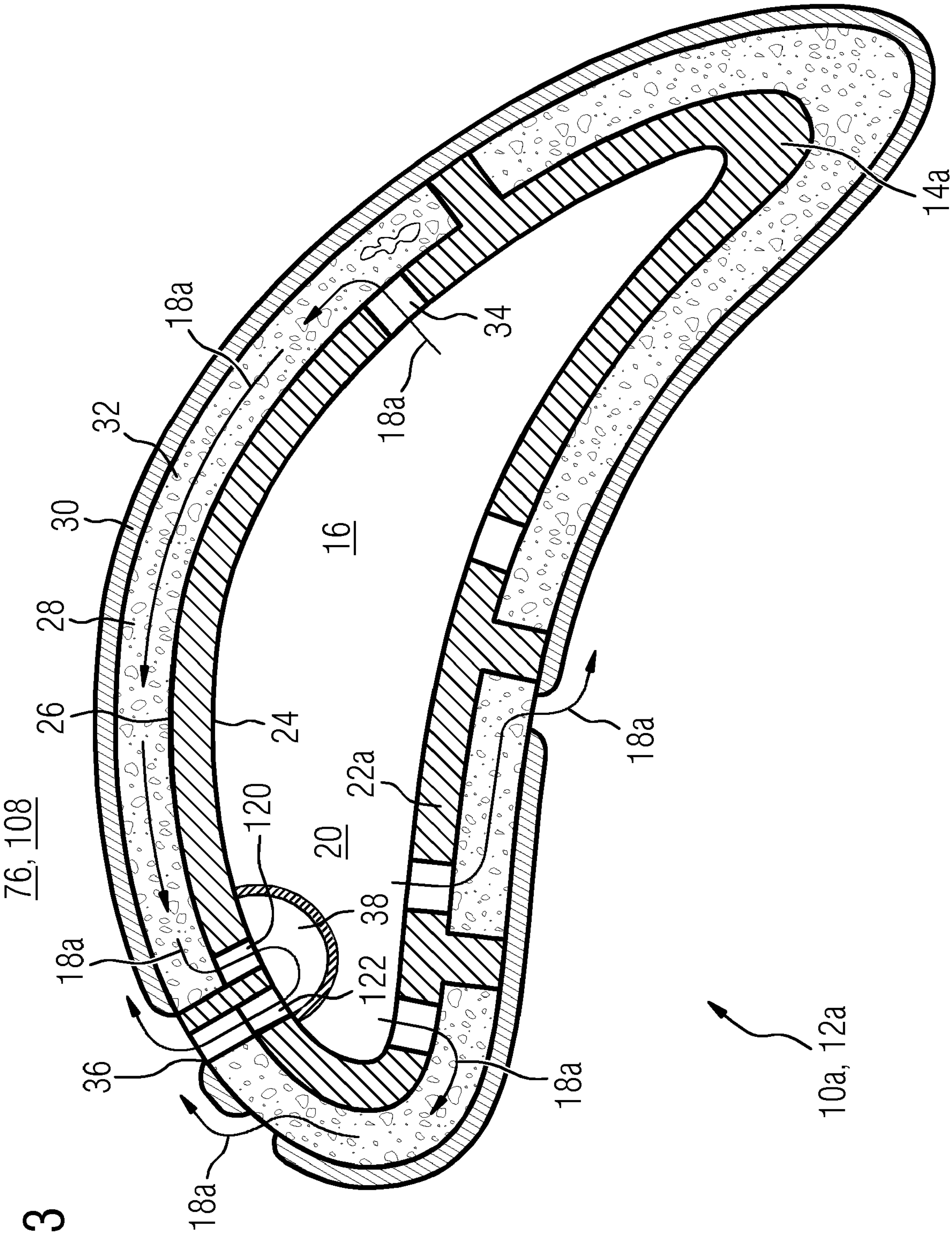
FIG 2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10a, 12a

FIG 3

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EUROPEAN SEARCH REPORT

Application Number
EP 18 15 2326

DOCUMENTS CONSIDERED TO BE RELEVANT

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X Y A	EP 1 475 567 A1 (SIEMENS AG [DE]) 10 November 2004 (2004-11-10) * paragraphs [0019], [0021], [0022], [0030]; claim 15; figure 2 * -----	1,2,4-13 3 14,15	
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			F01D

The present search report has been drawn up for all claims

1

Place of search	Date of completion of the search	Examiner
Munich	18 June 2018	Georgi, Jan

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