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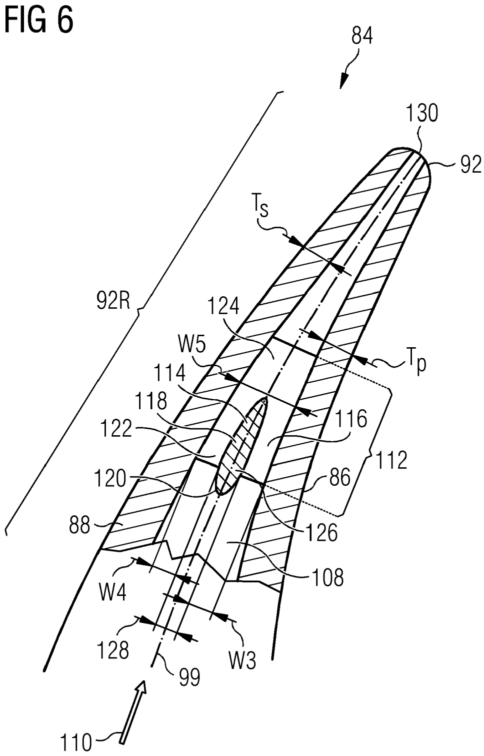
(54) AEROFOIL FOR A GAS TURBINE

(57) An aerofoil (84) for a gas turbine engine (10) comprising a suction side wall (68), a pressure side wall (66), a leading edge (70) and a trailing edge (72), a first radial end (74) and a second radial end (76). The suction side wall (68) and the pressure side wall (66) extend from the first radial end (74) to the second radial end (76) and meet to define the leading edge (70) and the trailing edge (72). The suction side wall (68) and the pressure side wall (66) form a cavity (108) therebetween for the flow of a coolant therethrough and a converge towards the trailing edge (72) to form a converging portion (112). Within the converging portion (112) is a rib (114) that extends radially between first radial end (74) and the second radial end (76) and defines a first passageway (116) between the rib (114) and the pressure side wall (66) and a second passageway (118) between the rib (114) and the suction side wall (68).

FIG 6



EP 3 875 735 A1

Description

FIELD OF INVENTION

[0001] The present invention relates to an internally cooled aerofoil for a gas turbine engine.

BACKGROUND OF INVENTION

[0002] It is necessary to cool aerofoils of a gas turbine engine to prevent overheating and prolong service life. Aerofoils have internal cooling passages through which coolant is channelled during engine operation. In highly optimised aerofoils it is necessary to maintain a constant flow of coolant and have a constant external wall thickness to ensure a constant heat transfer coefficient and hence a constant cooling effect.

[0003] In the trailing edge region of an aerofoil there can be a compromise between having a smooth and constant wall thickness and a constant cooling flow in the axial direction. If the smooth and constant wall thickness is compromised there will be abrupt changes in the wall thickness. These changes in local wall thickness can create problems during manufacturing as the metal in the casting solidifies and this can lead to unacceptable porosity, cracking hence reduced yield and increased component cost. If the constant cooling flow in the axial direction is compromised there will be uneven cooling of the aerofoil and unacceptable temperature gradients that reduce service life and even have performance problems particularly downstream of the trailing edge region. Further, such known aerofoils may require relatively high cooling flow to maintain the component at a necessary temperature to obtain component life requirements. This relatively high cooling air flow is detrimental to the overall engine efficiency.

STATEMENT OF INVENTION

[0004] In accordance with a first aspect of the present disclosure there is provided an aerofoil for a gas turbine engine. The aerofoil comprising a suction side wall, a pressure side wall, a leading edge and a trailing edge, a first radial end and a second radial end. The suction side wall and the pressure side wall extend from the first radial end to the second radial end and meet to define the leading edge and the trailing edge. The suction side wall and the pressure side form a cavity therebetween for the flow of a coolant therethrough and a converge towards the trailing edge to form a converging portion. Within the converging portion is a rib that extends radially between first radial end and the second radial end and defines a first passageway between the rib and the pressure side wall and a second passageway between the rib and the suction side wall.

[0005] In at least the converging portion the suction side wall and the pressure side wall may have a thickness that tapers towards the trailing edge.

[0006] The sum of the widths or sectional areas of the passageway between the rib and the suction side wall and the passageway between the rib and the pressure side wall may be approximately equal to the width or sectional area from the suction side wall to pressure side wall immediately on the trailing edge side of the rib.

[0007] The rib may have a tapering portion which tapers towards the trailing edge.

[0008] The tapering portion may taper at approximately the same rate as the converging portion converges.

[0009] The aerofoil may comprise at least one chordal rib that spans the cavity from the suction side wall to the pressure side wall and is generally perpendicular to the suction side wall and the pressure side wall. The chordal rib may extend in the chordal direction and may be at least partly within the converging portion.

[0010] The rib may have a leading portion, a main body potion and a trailing portion.

[0011] The leading portion may extend towards the leading edge from and beyond the chordal rib.

[0012] A mean camber line is defined as the locus of points mid-way between the suction side wall and pressure side wall. The rib may have a centre line. The rib may be curved along the mean camber such that its centre line may be along the camber line.

[0013] In use, a coolant flows through the converging portion in a generally chordal direction towards the trailing edge.

[0014] The trailing edge may comprise at least one aperture for the release of coolant from the cavity.

[0015] The aerofoil may be monolithically formed in a casting step or in an additive manufacturing process.

[0016] The aerofoil may be part of a vane. The vane may comprise radially inner and radially outer platforms, the aerofoil spanning between the platforms.

[0017] The aerofoil may be part of a blade. The blade may comprise a platform and a tip, the aerofoil spanning between the platform and the tip.

[0018] The tip comprises any one of the group a shroud, a squealer and a winglet.

BRIEF DESCRIPTION OF THE DRAWINGS

[0019] The above-mentioned attributes and other features and advantages of this invention and the manner of attaining them will become more apparent and the invention itself will be better understood by reference to the following description of embodiments of the invention taken in conjunction with the accompanying drawings, wherein

FIG. 1 shows part of a turbine engine in a sectional view and in which the present aerofoil is incorporated;

FIG.2 shows part of a turbine section in a sectional view and which the present aerofoil is incorporated;

FIG.3 is a radial section of the trailing edge region of a conventional first aerofoil having a constant wall thickness;

FIG.4 is a radial section of the trailing edge region of a conventional second aerofoil and showing an undesirable change in flow capacity (or area) through the passage;

FIG.5 is a radial section of the trailing edge region of a conventional third aerofoil and showing localised abrupt changes in wall thickness that can create casting problems;

FIG.6 is a radial section of the trailing edge region of an aerofoil according to the present invention and showing the addition of a span height rib to maintain flow conditions and aid avoidance of possible casting issues.

DETAILED DESCRIPTION OF INVENTION

[0020] **FIG.** 1 shows an example of a gas turbine engine 10 in a sectional view. The gas turbine engine 10 comprises, in flow series, an inlet 12, a compressor section 14, a combustor section 16 and a turbine section 18 which are generally arranged in flow series and generally about and in the direction of a longitudinal or rotational axis 20. The gas turbine engine 10 further comprises a shaft 22 which is rotatable about the rotational axis 20 and which extends longitudinally through the gas turbine engine 10. The shaft 22 drivingly connects the turbine section 18 to the compressor section 14.

[0021] In operation of the gas turbine engine 10, air 24, which is taken in through the air inlet 12 is compressed by the compressor section 14 and delivered to the combustion section or burner section 16. The burner section 16 comprises a burner plenum 26, one or more combustion chambers 28 and at least one burner 30 fixed to each combustion chamber 28. The combustion chambers 28 and the burners 30 are located inside the burner plenum 26. The compressed air passing through the compressor 14 enters a diffuser 32 and is discharged from the diffuser 32 into the burner plenum 26 from where a portion of the air enters the burner 30 and is mixed with a gaseous or liquid fuel. The air/fuel mixture is then burned and the combustion gas 34 or working gas from the combustion is channelled through the combustion chamber 28 to the turbine section 18 via a transition duct 17.

[0022] This exemplary gas turbine engine 10 has a cannular combustor section arrangement 16, which is constituted by an annular array of combustor cans 19 each having the burner 30 and the combustion chamber 28, the transition duct 17 has a generally circular inlet that interfaces with the combustor chamber 28 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 18.

[0023] The turbine section 18 comprises a number of blade carrying discs 36 attached to the shaft 22. In the present example, two discs 36 each carry an annular array of turbine blades 38. However, the number of blade carrying discs could be different, i.e. only one disc or more than two discs. In addition, guiding vanes 40, which are fixed to a stator 42 of the gas turbine engine 10, are disposed between the stages of annular arrays of turbine blades 38. Between the exit of the combustion chamber 28 and the leading turbine blades 38 inlet guiding vanes 44 are provided and turn the flow of working gas onto the turbine blades 38.

[0024] The combustion gas from the combustion chamber 28 enters the turbine section 18 and drives the turbine blades 38 which in turn rotate the shaft 22. The guiding vanes 40, 44 serve to optimise the angle of the combustion or working gas on the turbine blades 38.

[0025] The turbine section 18 drives the compressor section 14. The compressor section 14 comprises an axial series of vane stages 46 and rotor blade stages 48. The rotor blade stages 48 comprise a rotor disc supporting an annular array of blades. The compressor section 14 also comprises a casing 50 that surrounds the rotor stages and supports the vane stages 48. The guide vane stages include an annular array of radially extending vanes that are mounted to the casing 50. 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 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.

[0026] The casing 50 defines a radially outer surface 52 of the passage 56 of the compressor 14. A radially inner surface 54 of the passage 56 is at least partly defined by a rotor drum 53 of the rotor which is partly defined by the annular array of blades 48.

[0027] The present invention is described with reference to the above exemplary turbine engine having a single shaft or spool connecting a single, multi-stage compressor and a single, one or more stage turbine. 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.

[0028] The terms upstream and downstream refer to the flow direction of the airflow and/or working gas flow through the engine unless otherwise stated. The terms forward and rearward refer to the general flow of gas through the engine. The terms axial, radial and circumferential are made with reference to the rotational axis 20 of the engine.

[0029] The term aerofoil is intended to apply to a stator vane or a rotor blade.

[0030] **FIG. 2** shows part of the turbine section 18 of the gas turbine engine 10. As mentioned above and in axial flow sequence of the working gas flow 34 through the turbine section 18 comprises an annular array of inlet guide vanes 44 and then an annular array of blades 38. The working gas flow 34 is channeled between a radially inner annulus 60 and a radially outer annulus 62 which is referred to as the 'gas path' and denoted 61.

[0031] The inlet guide vanes 44 comprise at least one aerofoil 64 having a suction side wall 68 and a pressure side wall 66, a leading edge 70 and a trailing edge 72, a first radial end 74 and a second radial end 76. The suction side wall 68 and the pressure side wall 66 extend from the first radial end 74 to the second radial end 76 and meet to define the leading edge 70 and the trailing edge 72. The first radial end 74 is attached to a radially inner platform 80 and the second radial end 76 is attached to a radially outer platform 82. The radially inner platform 80 and the radially outer platform 82 form part of the gas path 61.

[0032] The annular array of inlet guide vanes 44 comprises single vanes where there is a single aerofoil 64 mounted between the two platforms 80, 82. However, the annular array of inlet guide vanes 44 may comprise double, triple or more vane segments where there are two, three of more aerofoils 64 mounted between common platforms 80, 82.

[0033] The blades 38 comprise one aerofoil 84 having a pressure side wall 86 and a suction side wall 88, a leading edge 90 and a trailing edge 92, a first radial end 94 and a second radial end 96. The suction side wall 88 and the pressure side wall 86 extend from the first radial end 94 to the second radial end 96 and meet to define the leading edge 90 and the trailing edge 92. The first radial end 94 is attached to a radially inner platform 100 and the second radial end 76 is a blade tip 96 and is free. The blade tip 96 forms a tip gap 102 with a casing 104. The radially inner platform 100 and the casing 102 form part of the gas path 61. The tip 96 is a squealer configuration as is known in the art, but in other examples the blade tip 96 may comprise a shroud or a winglet as is known. The blade 38 comprises a root portion 106 which comprises a firtree configuration that engages a complimentary slot formed in the disc 36 for retaining the blades 38 in known manner. Alternatively, the root portion may be a dovetail configuration that engages a complimentary slot formed in the disc 36 for retaining the blades 38 in known manner.

[0034] The present invention is applicable to the aerofoils 64, 84 of the vane 44 and blade 38. In each of the vane 44 or blade 38, the suction side wall 68, 88 and the pressure side wall 66, 86 form a cavity 108 (see Fig. 6) therebetween. The cavity 108 is formed for the flow of a coolant therethrough. The cavity 108 may have many different configurations for optimal cooling depending on each application.

[0035] Before describing the present invention in detail, reference is now made to conventional aerofoil arrangements as shown in FIG.3 to FIG.5. Each of these figures show a portion of a conventional aerofoil 200 in cross-section and only the downstream portion including a trailing edge 202 formed by a suction side wall 204 and a pressure side wall 206 in conventional manner. A cavity 212 is formed between the suction side wall 204 and the pressure side wall 206. A coolant flow 208 flows through the cavity 212 and in the chordal direction towards the trailing edge 202 and out through an aperture or apertures 210.

[0036] In **FIG. 3** the cavity 212, the suction side wall 204 and the pressure side wall 206 all taper towards the trailing edge 202. In other words, the width W of the cavity and the thicknesses Ts and Tp of the suction side wall 204 and the pressure side wall 206 all reduce the nearer to the trailing edge. Here there is a smooth transition of the wall thicknesses Ts and Tp. For manufacturing there are no particular issues with casting this shape of the suction side wall 204 and the pressure side wall 206. However, this tapering causes the coolant flow 208 to accelerate towards the trailing edge 202 and therefore provide lessening cooling of the suction side wall 204 and the pressure side wall 206.

[0037] In **FIG. 4** the aerofoil 200 comprises the same configuration as described in FIG. 3 and comprises a chordal rib 214 running axially and at a prescribed pitch from hub to tip. The chordal rib 214 spans the cavity 212 from the suction side wall 204 to the pressure side wall 206 and is generally perpendicular to the suction side wall 204 and the pressure side wall 206. The chordal rib 214 extends in the chordal direction, that being generally in the plane of the figure and generally flowing the camber line of the aerofoil 200. The chordal rib 214 has a leading edge 216 and a trailing edge 218. In this example, there is a smooth transition of the wall thicknesses Ts and Tp. For manufacturing there are no particular issues with casting this configuration of aerofoil. However, at least from the leading edge 216 to the trailing edge 218 there is uneven coolant flow conditions because the cavity 212 is decreasing in width W. As shown in FIG.4 the cavity 212 has a width W2 at the leading edge 216 of the chordal rib 214 and a width W1 at the trailing edge 218. W2 is greater than W1 meaning the coolant flow must accelerate from the leading edge 216 of the chordal rib to its trailing edge 218. As the coolant flow accelerates it picks up less heat from the walls and leads to reduced cooling of the walls 204, 206 and the chordal rib 214 in the direction towards the trailing edge 202. In other words, there is an undesirable temperature gradient in the walls 204, 206 in the vicinity of and in the chordal rib 214.

[0038] In **FIG. 5** the aerofoil 200 comprises the same configuration as described in FIG. 4 except that the suction side wall 204 and the pressure side wall 206 have thickened portions 220, 222, respectively, in the region of the chordal rib 214. The thickening portions 220, 222 create a constant width portion 224 of the cavity 212. Coolant flow 208 passing through the constant width portion 224 has a constant velocity and therefore cools the walls 204, 206 and the chordal rib 214 more evenly and has a reduced temperature gradient in the walls 204, 206 in the vicinity of and in the chordal rib 214. Furthermore, the relatively abrupt increase in thickness of the thickened portions 220, 222, particularly in zones 226, causes relatively poor solidification during the casting process. This can lead to a relatively high scrap rate, shortened service life and even failure.

[0039] Referrence is now made to **FIG. 6** which depicts one exemplary embodiment of the present invention. As has been mentioned the inventive aerofoil will now be described with reference to a turbine blade 38, but it is equally applicable to the vane 44. The aerofoil 84 comprises the suction side wall 88, the pressure side wall 86, the leading edge 90 and the trailing edge 92 and configured as previously described. The suction side wall 88 and the pressure side wall 86 form the cavity 108. FIG. 6 shows a trailing edge portion 92R which extends from the trailing edge 92 towards the leading edge 90 a distance 10-30%, preferably 15-20%, of the length of a mean camber line 99 in the case of a blade and a distance 25-50%, preferably 35-40%, of the length of a mean camber line 99 in the case of a vane. The mean camber line 99 is defined as the locus of points halfway between the external surfaces of the suction side wall 88 and the pressure side wall 86.

[0040] Within at least a part of the trailing edge portion 92R, the suction side wall 88 and the pressure side wall 86 converge towards one another in a direction towards the trailing edge to form a converging portion 112. The cavity 108 decreases in width W in the direction towards the trailing edge 92 in the converging portion 112. The cavity 108 decreases in sectional area i.e. the area of the cavity 108 through which the coolant flow passes through or the area of the plane perpendicular to the coolant flow direction and between the suction side wall 88, the pressure side wall 86 and radially inner and out boundaries of the cavity. The converging portion 112 extends over a part of the trailing edge portion 92R but may extend over all the trailing edge portion 92R. Within the converging portion 112 is a rib 114 that extends radially between the first radial end 94 and the second radial end 96 of the aerofoil and preferably extends radially from the first radial end 94 to the second radial end 96. The rib 114 is not directly in contact with the suction side wall 88 or the pressure side wall 86. A first passageway 116, having a width W3, is defined between the rib 114 and the pressure side wall 86 and a second passageway 118, having a width W4, is defined between the rib 114 and the suction side wall 88. In the converging portion 112 the pressure side wall 86 and the suction side wall 88 have respective thicknesses Tp and Ts that preferably taper towards the trailing edge 92. In use, coolant 110 flows through the cavity 108 in the direction of the arrow 110 and portions of the coolant 110 pass through the first passageway 116 and second passageway 118.

[0041] The width W3 of the first passageway 116 is equal to the width W4 of the second passageway 118 in this exemplary embodiment. However, the rib 114 may be offset from the mean camber line 99 towards one or the other of the pressure side wall 86 or the suction side wall 88 and therefore W3 may be greater than W4 and vice versa. Nonetheless, the sum of the widths W3 and W4 of the first and second passageways is approximately equal to the width W5 from the suction side wall 88 to pressure side wall 86 immediately on the trailing edge side 92 of the rib 114 or immediately downstream of the rib 114. It should be appreciated that the sectional areas of the passageways 116, 118 and the cavity 108 for the coolant to flow through, in a direction from the leading edge to the tailing edge of the aerofoil, is proportional to the widths such that the widths W3, W4 and W5 are also indicative of the sectional areas at the indicated positions. In other words, the radial height of the cavity 108 is constant; however, in other embodiments the radial height of the cavity may vary and therefore the sectional areas indicated at the widths W3, W4 and W5 are relevant and here W3, W4 and W5 are indicative of the sectional areas rather than merely the widths.

[0042] The rib 114 has a leading edge portion 120 and a tapering portion 122 which tapers towards the trailing edge 92. The tapering portion 122 tapers at approximately the same rate as the converging portion 112 converges such that the width W3 of the first passageway 116 and the width W4 of the second passageway 118 remain approximately constant over the length of the tapering portion 122.

[0043] The aerofoil 84 comprises a chordal rib 124 that spans the cavity 108 from the suction side wall 88 to the pressure side wall 86 and is generally perpendicular to the suction side wall 86 and the pressure side wall 88. The chordal rib 124 extends in the chordal direction, i.e. the direction from the leading edge 90 towards the trailing edge 92 of the aerofoil 86. The chordal rib 124 is relatively thin, i.e. its thickness is relatively small into the figure or radially, and is approximately the same thickness as the pressure or suction side walls but may be less. The chordal rib 124 is at least partly located within the converging portion 112.

[0044] The leading edge portion 120 of the rib 114 extends and tapers towards the leading edge 90 of the aerofoil 84 from the leading edge of the chordal rib 114. In other applications the leading edge portion 120 may be aligned with or start from the leading edge of the chordal rib 114. The tapering portion 122 extends approximately 50% of the axial length of the chordal rib 114. Alternatively, the tapering portion 122 may extend anywhere up to the trailing edge of the chordal rib 114. The design of the rib 114 is based upon maintaining a desired velocity of the coolant passing through the cavity 108 thereby maximising cooling and minimising any temperature gradients in the aerofoil.

[0045] The rib 114 and chordal rib 124 are integral with one another such that the rib 114 is supported to prevent bending, vibrations and lateral movements towards either the suction or pressure side walls. Although only one chordal rib 124 is shown, there may be more than one chordal rib 124. The chordal ribs 124 may be equally spaced over the radial extent of the aerofoil or otherwise positioned to optimise stability of the rib 114.

[0046] As mentioned, the mean camber line 99 is defined as the locus of points mid-way between the suction side wall 88 and pressure side wall 86 and in particular their external surfaces. The rib 114 has a centre line 126 and as seen the rib 114 is curved along the mean camber 99 such that its centre line 126 is along or coincident with the mean camber line 99.

[0047] Although not shown, some aerofoils 84 may have a constant overall thickness in any radial line or the overall thickness may taper in the radial direction. Where the overall thickness tapers in the radial direction, the width W5 may also taper radially and therefore either one or both the widths W3 and W4 are intended to taper and by virtue of the sectional thickness 118 of the rib 114 also tapering in the radial direction.

[0048] The trailing edge 92 of the aerofoil comprises at least one aperture 130 for the release of coolant from the cavity108. The at least one aperture may be a series of spaced apart apertures along the radial extend of the trailing edge. The at least one aperture may be a circular or elliptical hole or may be a slot. The at least one aperture 130 is shown as being intersected by the mean camber line 99, but in other examples the at least one aperture 130 may be positioned in the pressure or suction side walls close to the trailing edge.

[0049] It should be appreciated that the aerofoil 84 and host vane or blade is monolithically formed in a casting step during manufacturing. Alternatively, the aerofoil is monolithically formed in an additive manufacturing process.

[0050] While the present invention has been described above by reference to various embodiments, it should be understood that many changes and modifications can be made to the described embodiments. It is therefore intended that the foregoing description be regarded as illustrative rather than limiting, and that it be understood that all equivalents and/or combinations of embodiments are intended to be included in this description.

[0051] It should be noted that the term "comprising" does not exclude other elements or steps and "a" or "an" does not exclude a plurality. 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. 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. An aerofoil (84) for a gas turbine engine (10),
the aerofoil (84) comprising a suction side wall (68), a pressure side wall (66), a leading edge (70) and a trailing edge (72), a first radial end (74) and a second radial end (76),
the suction side wall (68) and the pressure side wall (66) extend from the first radial end (74) to the second radial end (76) and meet to define the leading edge (70) and the trailing edge (72),
the suction side wall (68) and the pressure side wall (66) form a cavity (108) therebetween for the flow of a coolant therethrough and a converge towards the trailing edge (72) to form a converging portion (112), wherein
within the converging portion (112) is a rib (114) that extends radially between first radial end (74) and the second radial end (76) and defines a first passageway (116) between the rib (114) and the pressure side wall (66) and a second passageway (118) between the rib (114) and the suction side wall (68).

2. An aerofoil (84) as claimed in claim 1 wherein
in at least the converging portion (112) the suction side wall (68) and the pressure side wall (66) have a thickness that tapers towards the trailing edge (72).

3. An aerofoil (84) as claimed in claim 1 wherein
the sum of the widths (W3, W4) of the passageway between the rib (114) and the suction side wall (68) and the passageway between the rib (114) and the pressure side wall (66) are approximately equal to the width (W5) from the suction side wall (68) to pressure side wall (66) immediately on the trailing edge (72) side of the rib (114).

4. An aerofoil (84) as claimed in any one of claims 1-3 wherein
the rib (114) has a second passageway (122) which tapers towards the trailing edge (72).

5. An aerofoil (84) as claimed in claim 4 wherein
the second passageway (122) tapers at approximately the same rate as the converging portion (112) converges.

6. An aerofoil (84) as claimed in any one of claims 1-5 wherein
the aerofoil (84) comprises at least one chordal rib (114) that spans the cavity (108) from the suction side wall (68) to the pressure side wall (66) and is generally perpendicular to the suction side wall (68) and the pressure side wall (66),
the chordal rib (114) extends in the chordal direction and is at least partly within the converging portion (112).

7. An aerofoil (84) as claimed in any one of claims 1-6 wherein
the rib (114) has a leading edge portion (120) and a tapering portion (122).

8. An aerofoil (84) as claimed in claim 7 wherein
the leading edge portion (120) extends towards the

leading edge (70) from and beyond the chordal rib (114).

9. An aerofoil (84) as claimed in any one of claims 1-8 wherein
a mean camber line (99) is defined as the locus of points mid-way between the suction side wall (68) and pressure side wall (66),
the rib (114) has a centre line (126) and the rib (114) is curved along the mean camber such that its centre line is along the camber line.

10. An aerofoil (84) as claimed in any one of claims 1-9 wherein
in use, a coolant flows through the converging portion (112) is a generally chordal direction towards the trailing edge (72).

11. An aerofoil (84) as claimed in any one of claims 1-10 wherein
the trailing edge (72) comprises at least one aperture (130) for the release of coolant from the cavity (108).

12. An aerofoil (84) as claimed in any one of claims 1-11 wherein
the aerofoil (84) is monolithically formed in a casting step or in an additive manufacturing process.

13. An aerofoil (84) as claimed in any one of claims 1-12 wherein
the aerofoil (84) is part of a vane (44),
the vane (44) comprises a radially inner platform (80) and a radially outer platform (82), the aerofoil (84) spanning between the radially inner platform (80) and the radially outer platform (82).

14. An aerofoil (84) as claimed in any one of claims 1-13 wherein
the aerofoil (84) is part of a blade (38),
the blade (38) comprises a platform (100) and a tip (102),
the aerofoil (84) spanning between the platform (100) and the tip (102).

15. An aerofoil (84) as claimed in claim 14 wherein
the tip (102) comprises any one of the group a shroud, a squealer and a winglet

FIG 1

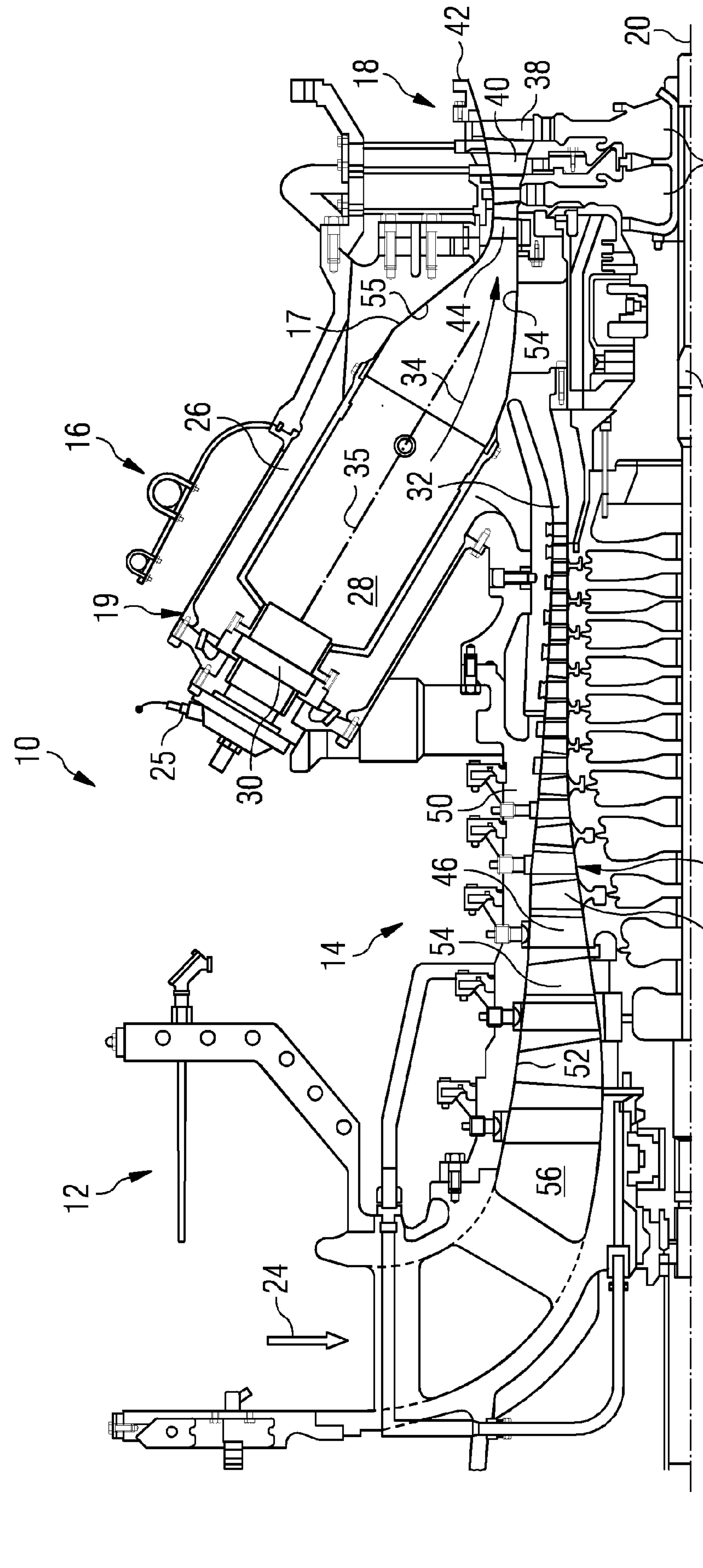

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FIG 2

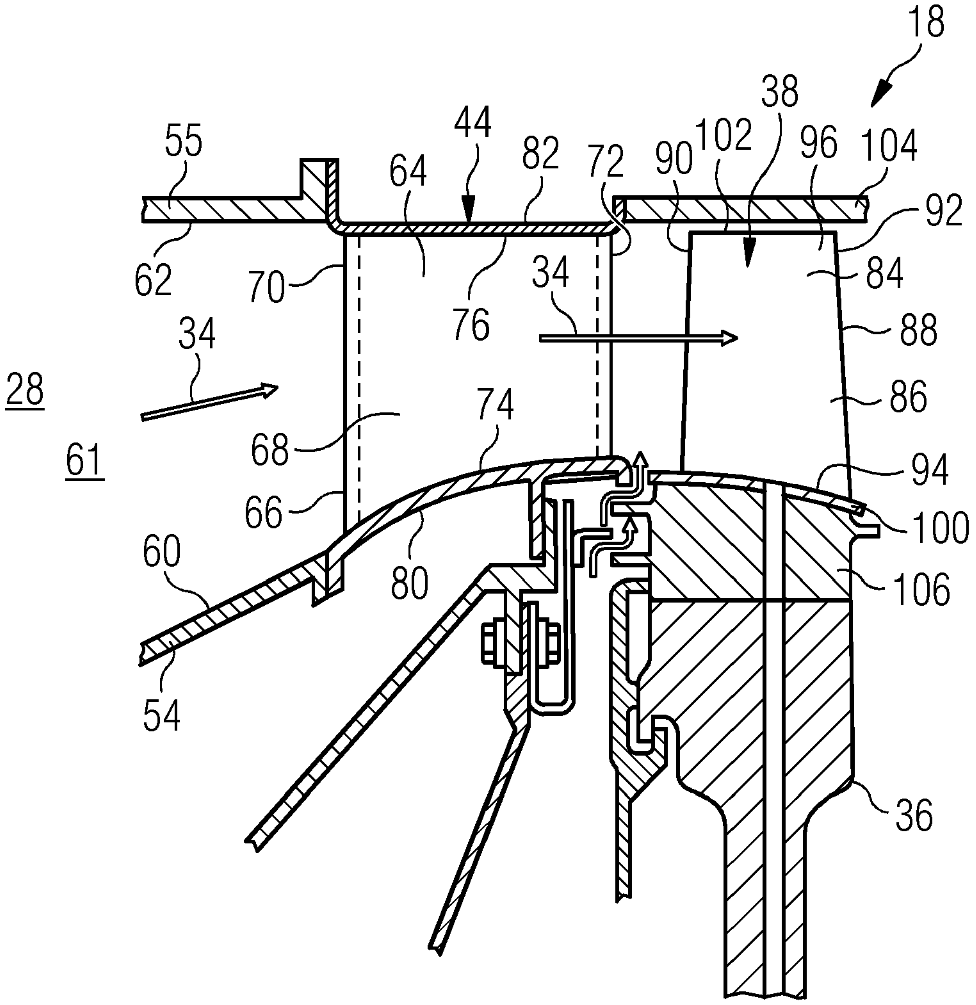
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FIG 3

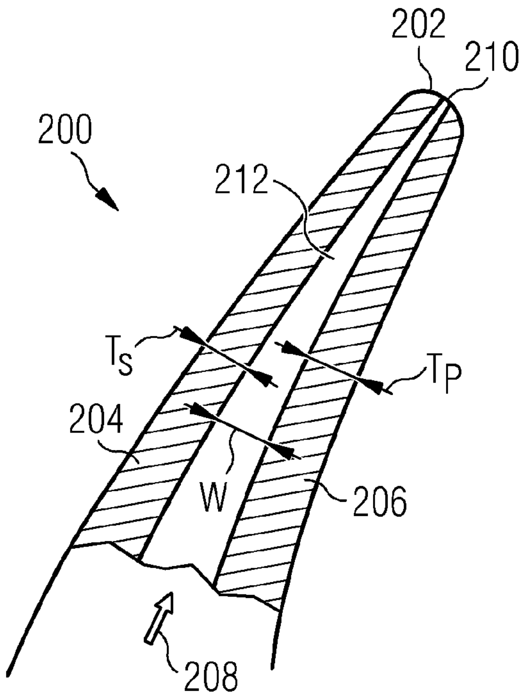


FIG 4

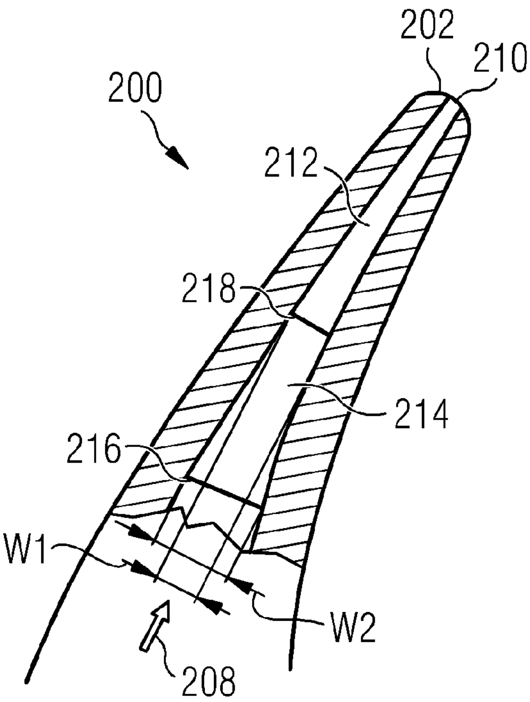


FIG 5

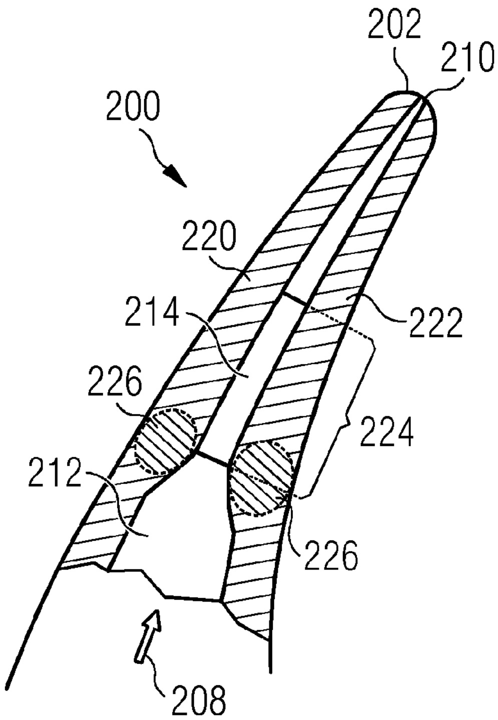
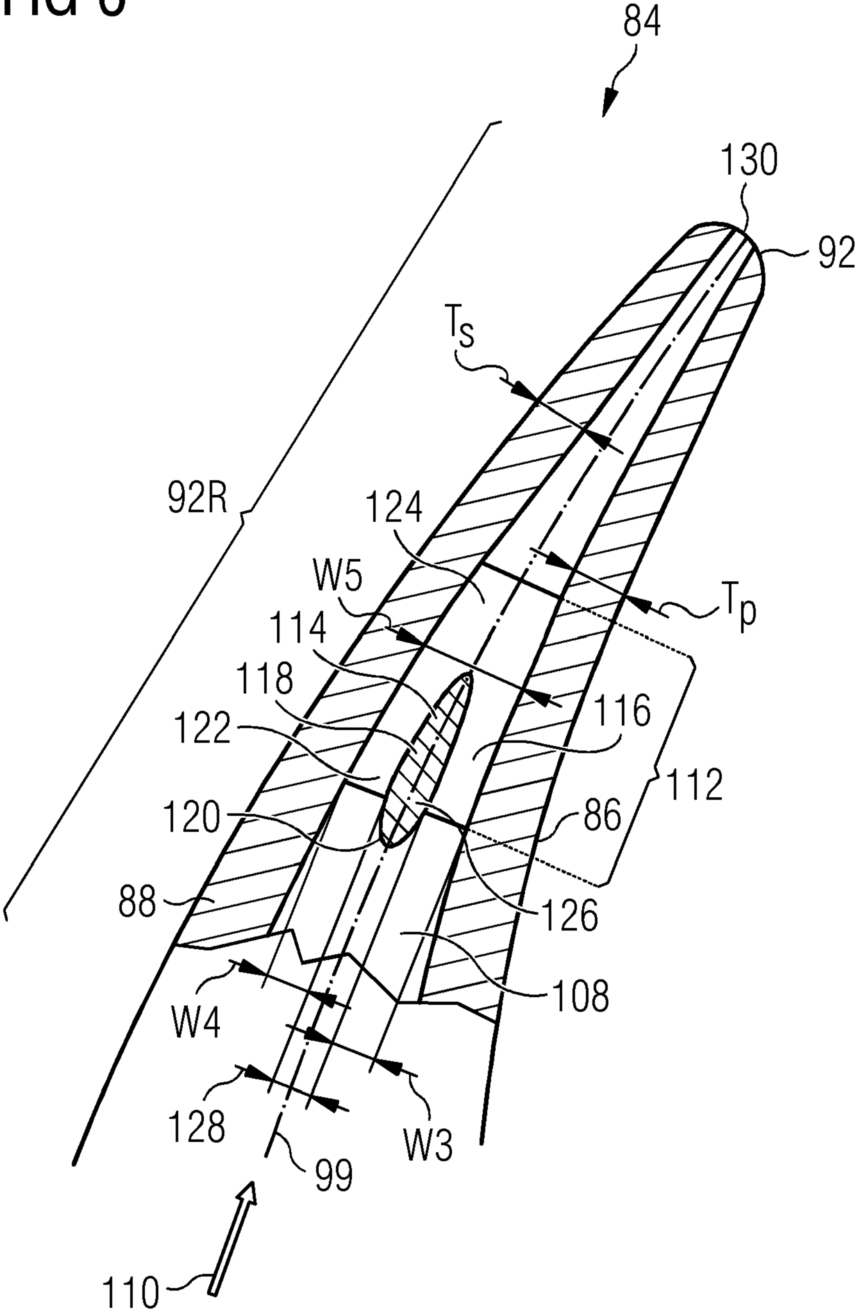


FIG 6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EUROPEAN SEARCH REPORT

Application Number
EP 20 16 1261

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Place of search	Date of completion of the search	Examiner
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