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(54) **Blade Cooling and Sealing System**

(57) A cooled component (60) for a gas turbine engine (10), for example a turbine rotor blade, is provided. The component has an internal cooling flow passage (70). Cooling air is passed through the internal cooling flow passage to remove heat from the component and thereby reduce its temperature. The cooling air is bled from the internal cooling passage after it has passed through a portion of the passage into an internal bleed flow passage (80). This bled air is then used in a seal. Thus, some of the cooling air that enters the internal cooling flow passage is used both to cool the component and to form a seal, for example with another component.

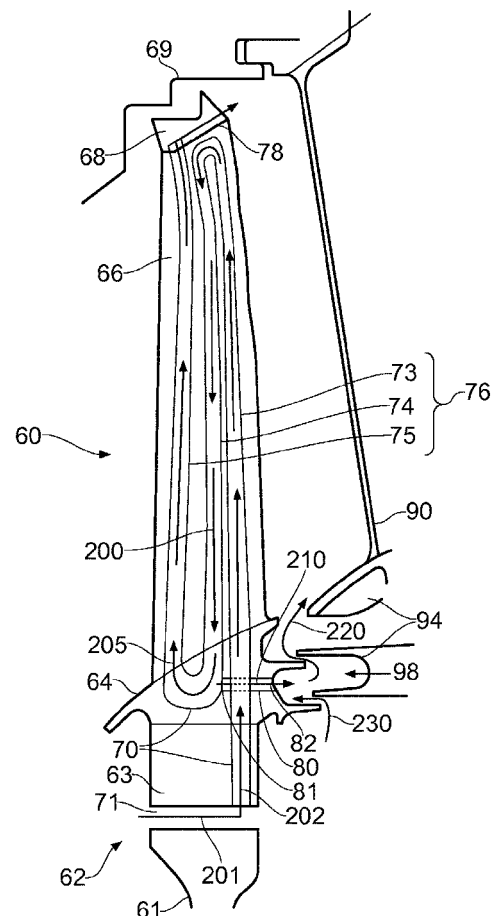


FIG. 3

Description

[0001] The present invention concerns cooling and sealing arrangements in a gas turbine engine. In particular, the present invention concerns a method and apparatus for using flow to cool components in a gas turbine engine and seal between components of a gas turbine engine.

[0002] The performance of a gas turbine engine, for example measured in terms of efficiency or specific output, may be improved by increasing the turbine gas temperature. It is therefore generally desirable to operate the turbine at the highest possible temperature. For a given gas turbine configuration (for example in terms of engine cycle compression ratio or bypass ratio), increasing the turbine entry gas temperature will produce more specific thrust (eg engine thrust per unit of air mass flow). However as turbine entry temperatures increases, the life of an uncooled turbine falls, necessitating the development of better materials and the introduction of internal air cooling.

[0003] Therefore, in order to cool turbine rotor blades and/or stator vanes (or at least to cool the aerofoil portions thereof), internal cooling passages are generally formed within the aerofoils. These internal cooling passages allow cooling air to be passed through the blades (or vanes) to remove heat through convection. The cooling air is typically taken from flow through a cooler part of the engine. For example, cooling air may be bled from the compressor, prior to combustion, for example from the HP compressor. Typical cooling air temperatures (for example at maximum take off condition) are between 700 and 900 K. Gas temperatures in the turbine can be in excess of 2100 K. Typically, cooling flow may enter the aerofoil via a fixture portion of the blade (at the radially inner end of the blade), pass through the blade in a substantially radial direction towards the tip, and then exit through the tip (at the radially outer end of the blade). Such cooling flow 100 is shown in the Fig. 2 arrangement, and discussed in greater detail below.

[0004] Generally, passing more cooling air through the blade allows the size (for example the mass) of the blade to be reduced. This has advantages in terms of engine mass and, in the case of a rotating blade, reduced centripetal loads. However, these advantages are at the expense of increasing the bleed flow, for example from the compressor prior to combustion. Any air that is bled from the compressor prior to combustion cannot be used to burn fuel in the combustor, and therefore cannot be used to generate useful work in the turbine. As such, increasing cooling flow to the blades generally allows their weight to be reduced and/or their life to be increased, at the expense of a reduction in engine performance.

[0005] In addition to cooling flow requirements, air is required for use in sealing various components in a gas turbine engine. For example, sealing flow is required to seal between rotor and stator rims. Once again, the sealing air is provided from elsewhere in the engine. Typically,

the sealing air may also be bled from the compressor. Such sealing flow 122, 124 is also shown in the Fig. 2 arrangement, and discussed in greater detail below.

[0006] As mentioned above, air that is bled from the compressor prior to combustion (i.e. air that has bypassed the combustor) cannot be used to generate useful work. Extracting air from the compressor, for example for cooling the hot turbine components or for providing a sealing flow, therefore has an adverse effect on engine performance, such as operating efficiency. Therefore, it is desirable to reduce the amount of air that is bled from the compressor. However, it is also desirable to retain or improve cooling to the turbine blades and/or vanes, for example to reduce their mass.

[0007] According to an aspect, there is provided a blade or vane for a gas turbine engine. The blade or vane comprises a cooling fluid inlet configured to allow cooling fluid into the blade or vane. The blade or vane comprises a cooling fluid outlet configured to allow cooling fluid out of the blade or vane. The blade or vane comprises an aerofoil that, in use, is gas-washed by a working fluid of the gas turbine engine. The blade or vane comprises a platform from which the aerofoil extends (in a direction that may be said to be radially outward). The blade or vane comprises an internal cooling flow passage between the cooling fluid inlet and the cooling fluid outlet configured to channel cooling fluid through the interior of the blade or vane (for example at least through the aerofoil portion of the blade or vane). The blade or vane comprises an internal bleed flow passage in fluid communication with the internal cooling flow passage at a bleeding position, and configured to bleed a portion of the cooling fluid from the internal cooling flow passage for use as a sealing flow. The blade or vane comprises a sealing flow outlet through which fluid from the internal bleed flow passage exits the blade or vane, the sealing flow outlet being positioned to allow the sealing flow to be used in a seal during operation. The internal cooling flow passage extends through at least a part of the aerofoil before the sealing flow is bled off at the bleeding position (for example during operation of the gas turbine engine). The cooling fluid inlet (71) is located radially inward of the platform (64) (for example in a location that is radially inward of the platform when the blade or vane is in use, i.e. when installed in the gas turbine engine). The sealing flow outlet (82) is located radially inward of the platform (64) (for example in a location that is radially inward of the platform when the blade or vane is in use, i.e. when installed in the gas turbine engine).

[0008] The aerofoil may include aerodynamic surfaces that, in operation, are gas-washed by the working fluid, such as a suction surface, a pressure surface, and a platform from which the suction surface and pressure surface extend. The blade or vane may comprise sealing features which may combine or interact with (for example interlock with, abut, or nearly abut) neighbouring surfaces (for example of neighbouring blades or vanes) of a gas turbine engine to form a seal when assembled. The blade or

vane may optionally include a shroud at its tip. The shroud may include sealing features to prevent or reduce over-tip leakage flow, for example in blade embodiments. The shroud may extend around the entire outer circumference of the blades. Alternatively, the blade or vane may have a partial shroud, or winglet, at its tip (for example extending around a portion of the segment between the blades). Alternatively still, the blade or vane may have no shroud at its tip.

[0009] Such an arrangement of blade or vane (which may be a rotor blade or a stator vane, for example of a turbine) means that at least some of the flow that is used for cooling the blade or vane is also used as sealing flow in a seal. The seal may be any suitable seal in the gas turbine engine, for example between axially spaced components, such as a rotor blade or stage and a downstream or upstream stator blade or stage. For example, part of the cooling flow that enters the blade or vane may pass through part of the internal cooling flow passage to cool the blade, and then be bled from the internal cooling passage for use in a seal. For example, the cooling flow may pass into at least a part of the inside of the aerofoil portion of the blade or vane before being bled off at the bleeding position. This may mean that at least a part of the gas washed surface which may be exposed to the higher temperatures in the turbine may be cooled before the flow is bled off to be used as sealing flow. Even if the cooling flow passes through a part of the internal cooling passage within the aerofoil before being bled off, the bleeding position may still be within a different part of the blade or vane (i.e. the bleeding position need not be within the aerofoil portion of the blade or vane, although in some embodiments it may be).

[0010] The total amount (i.e. mass flow rate) of flow (for example air) that is bled, for example from the compressor, to achieve a given level of blade cooling may thus be reduced (for example minimized), for example compared with conventional arrangements. Additionally or alternatively, the level of blade cooling may be increased for a given amount of air that is bled from the compressor.

[0011] The internal cooling flow passage may be a multipass cooling flow passage. Each pass may be arranged to carry cooling fluid in either a substantially radially outward direction or a substantially radially inward direction. The bleeding position may be after at least one pass through the internal cooling passage from the cooling fluid inlet. For example, the bleeding position may be after two passes through the internal cooling passage, e.g. one pass in the radially outward direction and one pass in the radially inward direction.

[0012] The radially outward and radially inward directions may be aligned with a longitudinal direction, or chordwise direction, of the blade or vane (or the aerofoil portion thereof). The radially outward direction may correspond to a root-to-tip direction of the aerofoil. The radially outward and radially inward directions may refer to the directions relative to a gas turbine engine (for exam-

ple an axial flow gas turbine engine), for example when the aerofoil is installed in the gas turbine engine.

[0013] Arranging the aerofoil in this manner ensures that the sealing flow that exits through the sealing flow outlet has already passed through a part of the internal cooling flow passage that extends along the chord of the aerofoil, and thus removed heat from along the chord of the aerofoil, at least, before it is used as a sealing flow. This may be advantageous in removing heat evenly from the aerofoil, for example to avoid localised hot-spots being observed during operation.

[0014] The internal cooling flow passage may comprise an s-shape. Such a shape may be referred to as a serpentine shape.

[0015] The internal cooling flow passage may thus be a continuous passage. For example, the passes through the blade or vane may be joined by a bend (which may turn through substantially 180 degrees, and may or may not be in the aerofoil portion of the blade or vane) towards either the radially outward or radially inward direction. Such an arrangement may form the s-shape.

[0016] Arranging the internal cooling flow passage to be an s-shape may enable the cooling air to be directed inside the blade or vane as required. It may also provide convenient bleeding position to bleed the sealing flow from the internal cooling flow passage, for example at the bends in the passage.

[0017] The blade or vane may comprise a fixture for attaching the blade or vane to a gas turbine engine. The fixture may comprise an attachment portion, which may be a fir tree for attaching a blade to a disc, for example. The fixture may comprise a shank. The shank may be between the fir tree and the aerofoil. The fixture may comprise the platform. The aerofoil may extend from the platform. The surface of the platform from which the aerofoil extends may be exposed to the working fluid of the gas turbine engine during operation.

[0018] The cooling fluid inlet may be located in the fixture, for example in the shank of a rotor blade. The cooling fluid inlet is radially inward of the platform. This may be convenient, for example, if the cooling flow is provided from towards the axial centreline of the engine. Additionally or alternatively, it may be convenient if the blade or vane is a rotor blade.

[0019] The sealing flow outlet may be located in the fixture, for example in the shank of a rotor blade. The sealing flow outlet is radially inward of the platform. This may be convenient if, for example, the sealing flow is to be used in a seal towards the radially inner side of the vane or blade, for example a rim seal.

[0020] The cooling flow outlet may be located towards the radially outer end of the blade or vane. For example, the cooling flow outlet may be located towards the tip of the aerofoil. This may be convenient if, for example, the cooling fluid may be used for other purposes at the radially outer end of the aerofoil. For example, cooling fluid (such as air) exiting from the radially outer end of the aerofoil may be used to cool the tip of the aerofoil, for

example the tip of a rotor blade. By way of further example, cooling fluid exiting the radially outer end may be used to cool the casing shroud within which the rotor blades rotate.

[0021] The sealing flow may be removed (bled) from the internal cooling flow passage as close to the sealing flow outlet as possible. For example, if the sealing flow is used in a seal towards the radially inner end of the blade or vane, the bleeding position may be located in the radially inner end of the blade or vane, for example in the fixture. This may allow the maximum possible cooling potential to be extracted from the cooling flow. It may also simplify the arrangement of the internal cooling flow passage and internal bleed flow passage.

[0022] The bleeding position and the internal bleed flow passage may both be provided within the fixture. This may allow all of the features relating to the sealing flow to be located in close proximity, thereby allowing maximum cooling and/or a simple arrangement of features.

[0023] In use, the blade or vane may have an upstream side and a downstream side defined relative to a flow direction through the gas turbine engine. The upstream side may correspond to the leading edge of the blade or vane and the downstream side may correspond to the trailing edge of the blade. The sealing flow outlet may be located on the downstream side of the blade or vane. Alternatively, the sealing flow outlet may be provided on the upstream side of the blade or vane.

[0024] The blade or vane may comprise two sealing flow outlets. One of the sealing flow outlets may be located at the downstream side of the aerofoil, and the other may be located at the upstream side of the aerofoil. This may enable sealing flow to be provided to one or both of an upstream seal and a downstream seal. One or both of the seals may form at least a part of a circumferential seal.

[0025] The bleeding position may be at least 20% along the length of the internal cooling flow passage from the cooling fluid inlet. In an embodiment, the bleeding position is in the range of from 30% to 90% along the length of the internal cooling flow passage. In an embodiment, the bleeding position is in the range of from 40% to 80% along the length of the internal cooling flow passage. In an embodiment, the bleeding position is in the range of from 50% to 70% along the length of the internal cooling flow passage. In an embodiment, the bleeding position is on the order of 60% along the length of the internal cooling flow passage.

[0026] Providing the bleeding position at least 20% along the length of the internal cooling flow passage ensures that the flow that is to be used for sealing extracts a significant amount of heat from the blade or vane before it is bled from the internal cooling flow passage. The cooling air will tend to heat up as it moves through the internal cooling flow passage. Therefore, bleeding the cooling flow only after at least 20% of the distance from the inlet ensures that the cooling air is not bled off when it is at its

coolest, and thus its most effective, towards the start of the internal cooling flow passage.

[0027] In the range of from 10% to 70% of the mass flow rate of cooling flow entering the blade through the cooling flow inlet may be bled through the internal bleed flow passage and out through the sealing flow outlet. In some embodiments, this may be in the range of from 15% to 60%, or 20% to 50%, or 25% to 40%, or on the order of 30%. As such, a significant proportion of the flow passing into the blade or vane through the cooling flow inlet may be bled off to provide sealing flow. In some embodiments, more than 70% (for example substantially all) of the flow passing into the blade or vane through the cooling flow inlet may be bled off to provide sealing flow.

[0028] The internal bleed flow passage may have a dog-leg shape. The dog leg shape may be arranged such that the internal bleed flow passage passes to the side of the internal cooling flow passage when viewed in the radial direction. Such a portion of the bleed flow passage may be referred to as a gallery. Arranging the internal bleed flow passage to have a dog-leg may allow the sealing flow outlet to be positioned in the desired place, by allowing the internal bleed flow passage to pass to the side of the internal cooling flow passage.

[0029] The blade or vane may comprise effusion cooling flow outlets configured to use cooling flow, for example from the internal cooling flow passage, to supply surface cooling to the aerofoil. This may be an efficient way to cool the blade or vane, for example by forming a cooler air film over the surface of the blade or vane to minimize its interaction with the hot working gas. The cooling flow outlet itself may be an effusion cooling flow outlet.

[0030] According to an aspect, there is provided a compressor or turbine for a gas turbine engine having at least one rotor stage and at least one stator stage, and at least one of the rotor blades or stator blades is a blade or vane as described herein. According to one aspect, all blades in one or more rotor and/or stator stages of a compressor and/or turbine may be as described herein. The invention may be particularly advantageous in turbines, where the gas temperatures are higher than those in compressors. However, the invention described herein may apply to compressors as well as turbines.

[0031] According to an aspect, there is provided rotor-stator stage for a gas turbine, wherein: the rotor stage comprises at least one blade according to any one of the preceding claims and/or the stator stage comprises at least one vane according to any one of the preceding claims; and the sealing flow exiting from the sealing flow outlet is used as a seal between the rotor stage and the stator stage.

[0032] For example, the rotor stage may have blades as described herein that provide a sealing flow for a seal between the rotor stage and a neighbouring stator stage, for example the neighbouring downstream stator stage. The sealing flow may thus provide a gas (for example air) seal between a rotatable rotor stage and a stationary stator stage. The sealing flow may provide a rim seal.

[0033] According to an aspect, there is provided a method of using a flow in a gas turbine engine to cool a blade or a vane in a stage of the gas turbine engine and to seal between a rotor stage and a stator stage of the gas turbine engine. The blade or vane comprises an aerofoil that, in use, is gas-washed by a working fluid of the gas turbine engine. The method comprises using at least a part of the flow both to cool the aerofoil and to seal between the stages.

[0034] Such a method may provide any one or more of the advantages described herein.

[0035] The method may comprise cooling the aerofoil by passing a cooling flow into the blade or vane, through an internal cooling passage inside the aerofoil, and out of the blade or vane through a cooling fluid outlet. The method may comprise bleeding at least a part of the cooling flow passing through the internal cooling passage through a sealing flow outlet in the blade or vane before it reaches the cooling fluid outlet. The method may comprise using the flow passing out of the blade through the sealing flow outlet in a seal between the stages.

[0036] The flow that is used for cooling and sealing may be bled from a compressor of the gas turbine engine before entry into a combustor. Thus, using at least a portion of the flow for both cooling and sealing means that less air needs to be bled from the compressor and/or that more heat can be removed from a given blade or vane for a given amount of air bled from the compressor.

[0037] Embodiments of the invention will now be described by way of example only, with reference to the accompanying diagrammatic drawings, in which:

Figure 1 is a sectional side view of a gas turbine engine;

Figure 2 is a schematic cross-section through a turbine rotor blade and sealing features, showing internal cooling flow passages through the blade;

Figure 3 is a schematic cross-section through a turbine rotor blade and sealing features according to an embodiment, showing internal flow passages through the blade;

Figure 4 is a schematic cross-section through a part of a turbine blade and fixture according to an embodiment, showing an internal bleed flow passage;

Figure 5 is schematic cross-section through a part of a turbine blade and fixture according to an embodiment in a different plane to that of Figure 4, also showing an internal bleed flow passage; and

Figure 6 is a schematic cross-section through a turbine rotor blade and sealing features according to an embodiment, showing internal flow passages through the blade.

[0038] With reference to Figure 1, a ducted fan gas turbine engine generally indicated at 10 has a principal and rotational axis X-X. The engine 10 comprises, in axial flow series, an air intake 11, a propulsive fan 12, an intermediate pressure compressor 13, a high-pressure compressor 14, combustion equipment 15, a high-pressure turbine 16, an intermediate pressure turbine 17, a low-pressure turbine 18 and a core engine exhaust nozzle 19. A nacelle 21 generally surrounds the engine 10 and defines the intake 12, a bypass duct 22 and a bypass exhaust nozzle 23.

[0039] The gas turbine engine 10 works in a conventional manner so that air entering the intake 11 is accelerated by the fan 12 to produce two air flows: a first air flow A into the intermediate pressure compressor 13 and a second air flow B which passes through the bypass duct 22 to provide propulsive thrust. The intermediate pressure compressor 13 compresses the air flow A directed into it before delivering that air to the high pressure compressor 14 where further compression takes place.

[0040] The compressed air exhausted from the high-pressure compressor 14 is directed into the combustion equipment 15 where it is mixed with fuel and the mixture combusted. The resultant hot combustion products then expand through, and thereby drive, the high, intermediate and low-pressure turbines 16, 17, 18 before being exhausted through the nozzle 19 to provide additional propulsive thrust. The high, intermediate and low-pressure turbines 16, 17, 18 respectively drive the high and intermediate pressure compressors 14, 13 and the fan 12 by suitable interconnecting shafts.

[0041] As the air passes through the gas turbine engine 10 it is heated to high temperatures. In particular, the first airflow B reaches high temperatures as it passes, through the core of the engine. Typically, particularly high temperatures may be reached at the exit of the combustion equipment 15, and as the air subsequently passes through the high, intermediate and low-pressure turbines 16, 17, 18.

[0042] Gas temperatures in the turbine can be in excess of 2100 K. This may be higher than the melting point of the materials from which the turbine components are manufactured. Furthermore, as mentioned herein, it is desirable to operate the turbine at the highest possible temperature because generally, for a given gas turbine configuration, increasing the turbine entry gas temperature will produce more specific thrust.

[0043] Therefore, in order to cool turbine rotor blades and/or stator vanes (for example to cool the aerofoil portions thereof), internal cooling passages may be formed within the blades. These internal cooling passages allow cooling air to be passed through the blades (or vanes) to remove heat through convection.

[0044] A conventional arrangement of a turbine rotor blade having cooling passages 100 formed therethrough is shown in Figure 2. In the Figure 2 arrangement, three passages 32, 34 and 36 are formed through the turbine blade 30. The passages allow cooling air 102, 104, 106

to enter the blade 30 through a fixture portion 40, through a platform 42, and then through an aerofoil portion 44. The aerofoil portion 44 (and the radially outer surface of the platform 42) are exposed to the high temperature working gas flow during operation.

[0045] In the conventional arrangement shown in Figure 2, the cooling air 112, 114, 116 passes through the passages 32, 34, 36 in the aerofoil portion 44 of the blade 40, and then out 118 through the radially outer tip 46 of the blade.

[0046] Typically, cooling air may be bled from the compressor 13, 14, prior to combustion, for example from the HP compressor. Typical cooling air temperatures are between 700 and 900 K. This air that is bled from the compressor 13, 14 prior to combustion cannot be used to burn fuel in the combustor 15, and therefore cannot be used to generate useful work in the turbine 16, 17, 18. As such, providing cooling air 100 to the turbine blade 30 generally results in a reduction in engine efficiency.

[0047] In addition to the cooling air 100 that is provided to the turbine blade 44 from the compressor 13, 14 in the Figure 2 arrangement, more air 122, 124 is required for sealing purposes. Sealing air 122 is provided to seal an upstream (front) platform rim gap 51 between the neighbouring upstream stator vane platform 52 and the platform 42 of the rotor blade 30. Sealing air 124 is provided to seal a downstream (rear) platform rim gap 53 between the neighbouring downstream stator vane platform 54 and the platform 42 of the rotor blade 30.

[0048] In the Figure 2 arrangement, this sealing air 122, 124 is also bled from the compressor 13, 14 prior to combustion. This sealing air 122, 124 also cannot be used to burn fuel in the combustor 15 to generate useful work in the turbine 16, 17, 18.

[0049] Thus, in the Figure 2 arrangement, air is bled from the compressor for two separate, distinct purposes: to provide cooling air 100 to the turbine blade 30, and to provide sealing air 122, 124 to seal the platform rim gaps. These two separate streams of air (cooling air 100 and sealing air 122, 124) that are bled from the compressor 13, 14 prior to combustion may each account for a significant proportion of the air passing through the compressor, typically in the range of from 0.5% to 10% each. This has an appreciable detrimental impact on engine efficiency.

[0050] A turbine blade 60 according to an embodiment of the invention is shown in Figure 3. The turbine blade 60 may be any type of turbine blade. For example, the turbine blade 60 may be part of a high pressure turbine 16, an intermediate pressure turbine 17, or a low pressure turbine 18. The turbine blade 60 may be part of any type of gas turbine engine, for example a ducted fan gas turbine (turbofan) engine 10 such as that shown in Figure 1, a turbojet, a turboprop, a turboshaft, an open rotor engine, or any other gas turbine engine, for example axial flow or radial flow.

[0051] The turbine blade 60 has a fixture 62, a platform 64, an aerofoil (or aerofoil portion) 66, and a tip 68. An

internal cooling flow passage 70 passes through the inside (interior) of the blade 60. Cooling fluid (for example cooling air) 200 passes through the internal cooling flow passage 70, thereby cooling the turbine blade 60.

[0052] The fixture 62 may allow the blade 60 to be attached to a corresponding component of a gas turbine engine, for example to a turbine disc (not shown). The term fixture 62 may be used to refer to parts of the blade 60 that are radially inward of the platform 64. The fixture 62 may not be gas washed by the working fluid passing through the turbine 16, 17, 18. In the Figure 3 embodiment, the fixture 62 (part or all of which may be referred to as a root) has an attachment portion (which may be a fir tree) 61 (partially shown) and a shank 63. The attachment portion 61 may comprise the part of the fixture 62 that allows the blade 60 to be attached to the corresponding component, such as a turbine disc.

[0053] The tip 68 of the blade 60 is shaped to form a circumferential seal with the case 69. The blade 60 shown in Figure 3 may be said to be shrouded. However, different types of blades (or indeed different components, such as vanes) may be used, for example shroudless or partially shrouded turbine blades.

[0054] The internal cooling flow passage 70 has a cooling fluid inlet 71 to the blade 60. In the embodiment shown in Figure 3, the inlet 71 is provided in the shank 63 of the attachment portion 62. This allows the cooling air to enter at a radially inner (or inboard) side (or portion) of the turbine blade 60. In other embodiments, the inlet 71 to the internal cooling flow passage 70 may be in a different position in the blade 60. In embodiments relating to a stationary vane, for example, the inlet 71 may be in a radially outer (or tip) portion of the vane.

[0055] As mentioned above, the cooling fluid 201 entering the cooling fluid inlet 71 may be provided from any suitable source of relatively cool fluid, such as from the compressor 13, 14 prior to combustion. By way of example, in the range of from 0.2% to 10%, for example 0.5% to 5%, for example 0.75% to 2%, for example 1% to 1.5% of flow through the compressor may be bled off to enter the cooling fluid inlet(s) 71 for each stage that is cooled. The amount of compressor air that is bled may be dependent on the stage that is cooled, and the temperature/cooling requirements thereof. Purely by way of example only, an intermediate pressure turbine stage may require in the range of from 0.2% to 2% of compressor air to be bled to it, and a high pressure turbine stage may require in the range of from 1% to 10% of compressor air to be bled from it. Of course, other amounts of compressor air (i.e. outside of these ranges) may be required for a given stage.

[0056] In the Figure 3 embodiment, the cooling flow 201 passes radially outwardly 202 through a radial part of the internal cooling flow passage 70 in the shank 63. In the Figure 3 view (which may be said to be substantially along a local circumferential direction, or a side view of the blade) the cooling passage 70 passes through the inside of the blade 60, inside and through the platform

64, and into the aerofoil 66. This allows the surfaces of the aerofoil 66 (for example the pressure surface and the suction surface) that are exposed, in use, to the hot working gas, to be cooled by the cooling air.

[0057] The internal cooling flow passage 70 forms a serpentine, or s' shape 76 at least partially within the aerofoil portion 66 in the Figure 3 embodiment. In Figure 3, the serpentine internal passage 76 comprises a first radial passage 73 through which the cooling air flows in a radially outward direction. The first radial passage 73 is joined, via a bend or corner in the internal passage 70 at or towards the tip (or radially outer portion) 68, to a second radial passage 74 through which the cooling air flows in a radially inward direction. The second radial passage 74 is joined, via a bend or corner in the internal passage 70 at or towards the fixture (or radially inner portion) 62, to a third radial passage 75 through which the cooling air flows in a radially outward direction. The three radial internal passages 73, 74, 75, together with the bends that join them together, form the serpentine internal flow passage 76. By way of clarification, in operation, the cooling air 200 flows through the internal cooling flow passage 70 (which includes the serpentine passage 76) in the direction of the arrows through the passage 70 shown in Figure 3.

[0058] In alternative embodiments, the internal flow passage may take a different shape. Any suitable shape could be used. By example only, the internal cooling flow passage could have 1, 2, 3 (as in the case of the Figure 3 embodiment), 4, 5, 6, 7, 8, 9, 10 or more than 10 radial flow passages, each of which may be joined together by a bend to form a continuous passage 70. The internal flow passage 70 of some embodiments may comprise features other than radial flow passages. Some embodiments may have more than one internal flow passage, any number of which may have an associated internal bleed flow passage. Some embodiments may, for example, have more than one serpentine passage. Each serpentine passage may have its own inlet(s) and outlet(s). The serpentine passage may have an axial component (in relation to the engine axis X-X) that is configured to direct flow in a generally upstream (as in the example shown in Figure 3), or generally downstream, direction. In an embodiment with two (or more) serpentine-shaped internal flow passages 70, the direction of the axial component of the flow may be different for different passages, i.e. one passage may result in an upstream axial flow component, and another passage may result in a downstream axial flow component. In other words, the or each serpentine passage could flow either forwards or rearwards within the blade.

[0059] In the Figure 3 embodiment, the cooling air that has passed through the whole of the internal cooling flow passage 70 passes out of the blade 60 through a cooling fluid outlet 78. In the Figure 3 embodiment, the cooling fluid outlet 78 is formed in the tip 68 (or at a radially outer portion) of the blade 60. In other embodiments, the cooling fluid outlet 78 may be formed in other parts of the

blade 60, for example at a radially inner portion of the blade 60 (towards or in the fixture 62).

[0060] Although not shown in Figure 3, embodiments of the invention may comprise effusion (or film) cooling holes. Such effusion cooling holes may take cooling fluid from the internal cooling flow passage 70 to provide a film of relatively cool air to the aerodynamic surfaces of the blade 60. This may further help to prevent the surface of the blade 60 (and in particular the surface of the platform 64 and aerofoil 66 that are exposed to the working fluid during use) from being heated to temperatures above allowable limits.

[0061] The blade 60 comprises an internal bleed flow passage 80 in fluid communication with the internal cooling flow passage 70 at a bleeding position 81. At the bleeding position 81, the internal cooling flow passage 70 and the internal bleed flow passage 80 are fluidly interconnected. At the bleeding position 81, some of the air 205 flowing through the internal cooling flow passage 70 continues through the rest of the internal cooling flow passage 70, and some of the air 210 flowing through the internal cooling flow passage 70 before the bleeding position 81 enters and flows through the internal bleed flow passage 80.

[0062] The air 210 that is bled off the cooling passage 70 at the bleeding position 81 may be used in a seal. Thus, the bled air 210 may be referred to as sealing air 210. The sealing air 210 may be used, for example, to seal one or more gaps between the turbine blade 60 and neighbouring components, such as neighbouring stator vanes. By way of example, in the embodiment of Figure 3, the sealing air 210 exits the blade 60 through a sealing flow outlet 82 and is used to seal the gap between the blade platform 64 and the platform (or root) 94 of a downstream stator vane 90. As such, the sealing air 210 in Figure 3 may be used as rear platform rim sealing flow, for example in a rear platform rim sealing flow 98. In other embodiments, the sealing air 210 that is bled through the internal bleed flow passage 80 may be used in other seals.

[0063] In Figure 3, the bleeding position 81 is at or near the radially inner portion at the end of the second radial passage 74 of an internal cooling flow passage 70. For example, the bleeding position 81 may be positioned on the bend between the second radial passage 74 and the third radial passage 75. This means that the cooling air 200 has already passed through a significant portion of the internal cooling flow passage 70 at the position 81 where a proportion of it is bled to become sealing air 210. Air exiting the internal cooling flow passage 70 at the bleeding position 81 to be bled off into the internal bleed flow passage 80 has already passed through at least a portion 73, 74 of the internal cooling flow passage 70 that is within, or inside, the aerofoil 66 of the blade 60. Thus, the air entering the internal bleed flow passage 80 has already removed heat from the blade 60, for example from the aerodynamic surfaces of the aerofoil 66.

[0064] In Figure 3, the bleeding position 81 is after two

passes (which may be referred to as radial passes) through the blade 60. In other embodiments, the bleeding position 81 may be at a different position along the internal cooling flow passage 70. Indeed, as mentioned herein, the internal cooling flow passage 70 may take a substantially different shape to that shown in Figure 3. It is desirable for the internal flow passage 70 to have passed into (and optionally back out of) the aerofoil portion 66 of the blade 60 before the bleeding position 81. This may allow heat to be removed particularly effectively from the gas washed surfaces of the blade 60 for example from the aerofoil portion 66.

[0065] The proportion of the flow 210 that enters the internal bleed flow passage 80 may be determined by, for example, the relative size (for example cross-sectional area, or effective flow area) of the internal bleed flow passage 80 and the cooling flow passage 75 downstream of the bleeding position 81. Additionally or alternatively, the proportion of the flow 210 that enters the internal bleed flow passage 80 may be determined by the connection geometry between the internal cooling flow passage 70 and the internal bleed flow passage 80 at the bleeding position 81. For example, the angle between the internal cooling flow passage 70 and the internal bleed flow passage 80 may have an impact, as may the internal wall shape.

[0066] The proportion 210 of the flow 200 in the internal cooling flowing flow passage 74 upstream of the bleeding position 81 that enters the bleed flow passage 80 may be in the range of from 5% to 75%, for example on the order of 30%.

[0067] According to the arrangements described herein, at least a portion of the air that is used to cool a turbine component (such as a turbine blade 60) may also be used to provide a sealing flow 210. This means that, for a given mass flow rate of sealing flow (which may be determined by the geometry to be sealed), and a given level of component (for example turbine blade) cooling (which may be determined by, for example, the maximum component operating temperature for a given service life), the total mass flow that needs to be bled from the compressor 12, 13, 14 (for example) may be reduced. In turn this means that more air can pass through the combustor to burn fuel, and thus more useful work can be extracted by the turbine 17, 18.

[0068] Additionally or alternatively, the total amount of air that is bled from the compressor 12, 13, 14 may be kept substantially the same as arrangements where the cooling flow and sealing flow are separate (such as that shown in Figure 2). In that case, the total mass flow rate of cooling flow may be increased in the turbine blade 60, at least in the portion of the internal cooling flow passage 70 before the bleeding position 81. This may therefore result in greater blade cooling effectiveness for a given mass flow of air bled from the compressor 12, 13, 14. In turn, this may allow the blade weight to be reduced, for example for a given component life. For example, the combined cooling/sealing flow arrangement shown in

Figure 3 may result in a blade 60 that is in the range of from 2% to 25%, for example 5% to 15%, for example 8% to 12%, for example around 10% lighter than a blade of equivalent external geometry (for example having the same external aerofoil geometry) with the separate cooling and sealing flows such as that shown in Figure 2. This may result in further advantages, such as weight reductions in other components, such as the turbine rotor disc.

[0069] In some embodiments, the advantages described herein may be combined, such that both less mass flow rate of air is bled from the compressor 12, 13, 14 and more cooling air is provided to the blade 60, and thus its mass may be lower.

[0070] In some embodiments, additional sealing flow 230 may be provided to augment the bled sealing flow 210. This is represented in the Figure 3 embodiment by the arrow 230. Such additional sealing flow 230 may be bled from compressor 12, 13, 14 in the conventional manner. In some embodiments, no additional sealing flow 230 may be required, such that the entire sealing flow 220 is provided by the sealing flow 210 bled from the internal cooling flow passage 70. In other embodiments, the mass flow rate of additional sealing flow 230 that is required may be less than the mass flow rate of the conventional sealing flow 124 that would be required in the absence of the bled sealing flow 210.

[0071] The rear platform rim seal 98 that the bled sealing air 210 is used to seal in the Figure 3 embodiment has a double-seal arrangement, i.e. there is a radially inner seal and a radially outer seal. The seal 98 may be a labyrinth seal. This may reduce the possibility of the sealing air 210 being ingested into the disc cavity. Thus, it may reduce the risk of the sealing air moving radially inward, away from the seal in which it is to be used. However, the sealing air 210 may be used in any suitable seal, for example a single seal arrangement between two surfaces (which may be configured to rotate relative to each other).

[0072] The bled sealing air 210 may exit through the sealing flow outlet 82 in any suitable position relative to the seal 98. For example, the sealing flow outlet 82 may be upstream of the seal 98 and/or within or into the seal 98. By way of example, in the Figure 3 embodiment, the bled sealing flow 210 exits within the seal 98, and additional sealing flow 230 is used in an upstream (radially inner) part of the seal 98. Other embodiments may have different arrangements of seal and/or sealing flow outlet.

[0073] The bled sealing air 210 is shown in the Figure 3 embodiment as exiting into the seal 98 through the sealing flow outlet 82 in a direction that has substantially no radial component. However, in other embodiments, this may not be the case. For example, the angle between the axial direction and the bled sealing air 210 may be in the range of from 0 degrees to 45 degrees, for example 1 degree to 35 degrees, for example 2 degrees to 25 degrees, for example 3 degrees to 15 degrees, for example 4 degrees to 10 degrees, for example around 5

degrees (positive angles being defined as being towards the radially outer direction). However, any suitable exit angle for the bled sealing air 210 may be chosen depending on, for example, the geometry of the seal. The direction of the bled sealing air 210 that exits through the flow outlet 82 may additionally or alternatively have a circumferential component.

[0074] In the Figure 3 embodiment, the bleed position 81 is in the fixture portion 62 of the blade 60. In other embodiments, however, the bleed position 81 may be elsewhere in the blade. The location of the bleed position 81 may be dependent on, for example, the location of the required sealing flow 210 and/or the shape of the internal cooling flow passage 70.

[0075] The internal cooling flow passage 70 and the internal bleed flow passage 80 may be arranged so as not to interact with each other, for example at any position other than the bleed flow position. In the Figure 3 arrangement, the dashed line along the internal bleed flow passage 80 represents the internal bleed flow passage 80 passing to the side of the internal cooling flow passage 70 so as to avoid the two passages interacting. In the Figure 3 embodiment, the internal bleed flow passage 80 is shown as passing to the side of the internal cooling flow passage 70 within the shank 63. This may be convenient because the shank may have a sufficient thickness for the two passages to pass side-by-side, for example in the local circumferential plane (that is, a plane perpendicular to the local radial direction). However, the two passages 70, 80 may pass each other side-by-side in other places depending on the desired arrangement, for example of the bleed position 81 and/or the sealing flow outlet 82.

[0076] Details of the internal passages 70, 80 in the shank 63 of the Figure 3 embodiment are shown in Figures 4 and 5. Figures 4 and 5 show that the internal bleed flow passage 80 (which may be referred to as a gallery, shank gallery, or bleed flow gallery) of the Figure 3 embodiment has a dogleg 85. This dogleg 85 allows the internal bleed flow passage 80 to pass to the side of the internal cooling flow passage 70, without the two internal passages 70, 80 interacting. In the case of the Figure 3, 4 and 5 embodiment, the dogleg 85 has a component in the local circumferential direction to allow the internal bleed flow passage 80 to pass to the side of the internal cooling flow passage 70 when viewed along a radial direction (which direction may be referred to as a longitudinal axis of the blade 60 itself). In this embodiment, the dogleg 85 is arranged such that the internal bleed flow passage is taken towards the pressure surface of the blade 60. Other embodiments may have alternative arrangements which allow the two internal flow passages 70, 80 to pass each other, which may or may not comprise dogleg structures. Some embodiments may not comprise a dogleg structure. In some embodiments, the sealing flow outlet 82 may be provided at a location that does not require the internal bleed flow passage 80 to pass to the side of the internal cooling flow passage 70. For ex-

ample, the sealing flow outlet 82 may be provided on a circumferential side surface of the shank 63. Such embodiments, amongst others, may not require a dogleg arrangement.

[0077] Figure 6 shows an alternative blade 60. The blade 60 shown in Figure 6 shares many features with the blade 60 of the Figure 3 embodiment described in detail herein. Like features between the Figure 3 and Figure 6 embodiments have the same reference numerals, and will not be described in detail again in relation to the Figure 6 embodiment. By way of clarification, any features described in relation to Figures 3, 4, or 5 may also apply to the Figure 6 embodiment, or any other embodiment of the invention.

[0078] Figure 6 comprises a second internal bleed flow passage 86. The second bleed flow passage 86 may be substantially the same as the internal bleed flow passage 80 in construction. However, the second internal bleed flow passage 86 may be configured (for example positioned and/or shaped and/or sized) to provide a second sealing flow 240 to a different seal to that of the sealing flow 210 from the internal bleed flow passage 80. In the Figure 6 embodiment, the second internal bleed flow passage 86 is designed to provide second sealing flow 240 to a seal upstream of the blade 60. For example, the second sealing flow 240 may be used in a forward platform rim sealing flow, for example to form a circumferential seal between the rotor platform 64 and neighbouring upstream vane platforms. In other embodiments, the second sealing flow 240 could be used to provide air to a seal in other positions.

[0079] At least some of the examples described herein refer to rotor blades, for example high pressure, low pressure or intermediate pressure turbine rotor blades. The invention may be embodied in other types of components, for example stator vanes, for example high pressure, low pressure or intermediate pressure turbine stator vanes. The invention may, by way of example only, be embodied in any component that has internal cooling passages (or components in which internal cooling flow passages may be desirable). For example, embodiments of the invention may include those components that do not currently include internal cooling flow passages, but in which internal cooling flow passages may be desirable if, for example, the bleed flow (for example from an upstream compressor) may be reduced by using the bleed flow both for cooling purposes and for sealing in or between components. Any feature described and/or claimed herein may combined with any other compatible feature described in relation to the same or another embodiment.

Claims

1. A blade or vane (60) for a gas turbine engine (10) comprising:

a cooling fluid inlet (71) configured to allow cooling fluid into the blade or vane;
 a cooling fluid outlet (78) configured to allow cooling fluid out of the blade or vane;
 an aerofoil (66) that, in use, is gas-washed by a working fluid of the gas turbine engine, the aerofoil extending from a platform (64);
 an internal cooling flow passage (70) between the cooling fluid inlet and the cooling fluid outlet configured to channel cooling fluid through the interior of the blade or vane;
 an internal bleed flow passage (80) in fluid communication with the internal cooling flow passage at a bleeding position (81), and configured to bleed at least a portion of the cooling fluid from the internal cooling flow passage for use as a sealing flow (210); and
 a sealing flow outlet (82) through which fluid from the internal bleed flow passage exits the blade or vane, the sealing flow outlet being positioned to allow the sealing flow to be used in a seal (98) during operation, wherein:

the internal cooling flow passage extends through at least a part of the aerofoil before the sealing flow is bled off at the bleeding position;
 the cooling fluid inlet (71) is located radially inward of the platform (64); and
 the sealing flow outlet (82) is located radially inward of the platform (64).

2. A blade or vane (60) according to claim 1, wherein:

the internal cooling flow passage (70) is a multipass cooling flow passage, each pass (73, 74, 75) being arranged to carry cooling fluid in either a radially outward direction or a radially inward direction; and
 the bleeding position (81) is after at least one pass through the internal cooling passage from the cooling fluid inlet (71)

3. A blade or vane (60) according to claim 1 or claim 2, wherein the internal cooling flow passage (70) comprises an s-shape.

4. A blade or vane (60) according to any one of the preceding claims, further comprising a fixture (62) for attaching the blade or vane to a gas turbine engine (10), wherein:

the cooling fluid inlet (71) is located in the fixture (62); and
 the sealing flow outlet (82) is located in the fixture (62).

5. A blade or vane (60) according to any one of the

preceding claims, wherein the cooling flow outlet (78) is located towards the radially outer end (68) of the blade or vane.

6. A blade or vane (60) according to claim 4 or claim 5, when dependent on claim 4, wherein the bleeding position (81) is located in the fixture (62).

7. A blade or vane (60) according to any one of the preceding claims, wherein:

the blade or vane, in use, has an upstream side and a downstream side defined relative to a flow direction through the gas turbine engine; and
 the sealing flow outlet (82) is located on the downstream side of the blade or vane.

8. A blade or vane (60) according to any one of the preceding claims, wherein the bleeding position (81) is at least 20% along the length of the internal cooling flow passage (70) from the cooling fluid inlet (71).

9. A blade or vane (60) according to any one of the preceding claims, wherein in the range of from 10% to 70% of the mass flow rate of cooling flow entering the blade through the cooling flow inlet (71) is bled through the internal bleed flow passage (80) and out through the sealing flow outlet (82).

10. A blade or vane (60) according to any one of the preceding claims, wherein the internal bleed flow passage (80) has a dog-leg shape (85) such that it passes to the side of the internal cooling flow passage (70) when viewed in the radial direction.

11. A compressor (13, 14) or turbine (16, 17, 18) for a gas turbine engine (10) having at least one rotor stage having rotor blades and at least one stator stage having stator vanes, wherein at least one of the rotor blades or stator vanes is a blade or vane (60) according to any one of the preceding claims.

12. A rotor-stator stage for a gas turbine, wherein:

the rotor stage comprises at least one blade (60) according to any one of the preceding claims and/or the stator stage comprises at least one vane according to any one of the preceding claims; and
 the sealing flow (210) exiting from the sealing flow outlet (82) is used in a seal (98) between the rotor stage and the stator stage.

13. A method of using a flow in a gas turbine engine (10) to cool a blade or a vane (60) in a stage of the gas turbine engine and to seal between a rotor stage and a stator stage of the gas turbine engine, wherein:

the blade or vane (60) comprises an aerofoil (66)
that, in use, is gas-washed by a working fluid of
the gas turbine engine; and
the method comprising using at least a part of
the flow both to cool the aerofoil (66) and to seal 5
between the stages.

14. A method according to claim 13, comprising:

cooling the aerofoil (66) by passing a cooling 10
flow into the blade or vane (60), through an in-
ternal cooling passage (70) inside the aerofoil
(66), and out of the blade or vane through a cool-
ing fluid outlet (78);
bleeding at least a part of the cooling flow pass- 15
ing through the internal cooling passage (70)
through a sealing flow outlet (82) in the blade or
vane before it reaches the cooling fluid outlet
(78); and
using the flow (210) passing out of the blade 20
through the sealing flow outlet in a seal (98) be-
tween the stages.

15. A method according to claim 13 or claim 14, wherein
the flow (210) that is used for cooling and sealing is 25
bled from a compressor (13, 14) of the gas turbine
engine (10) before entry into a combustor (15).

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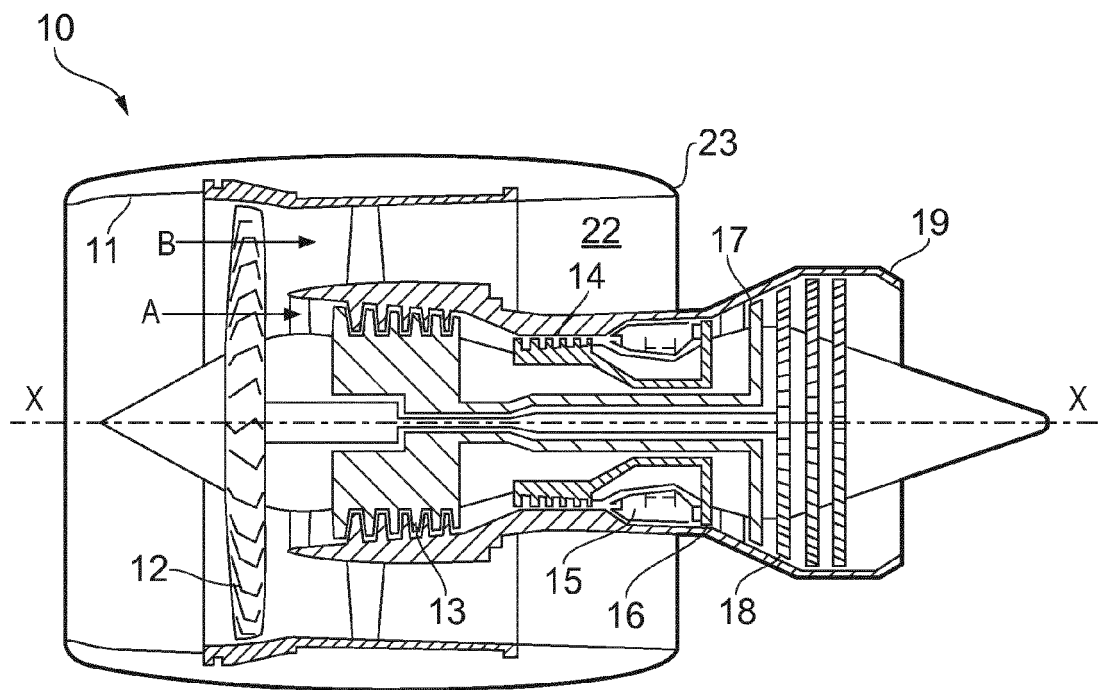


FIG. 1

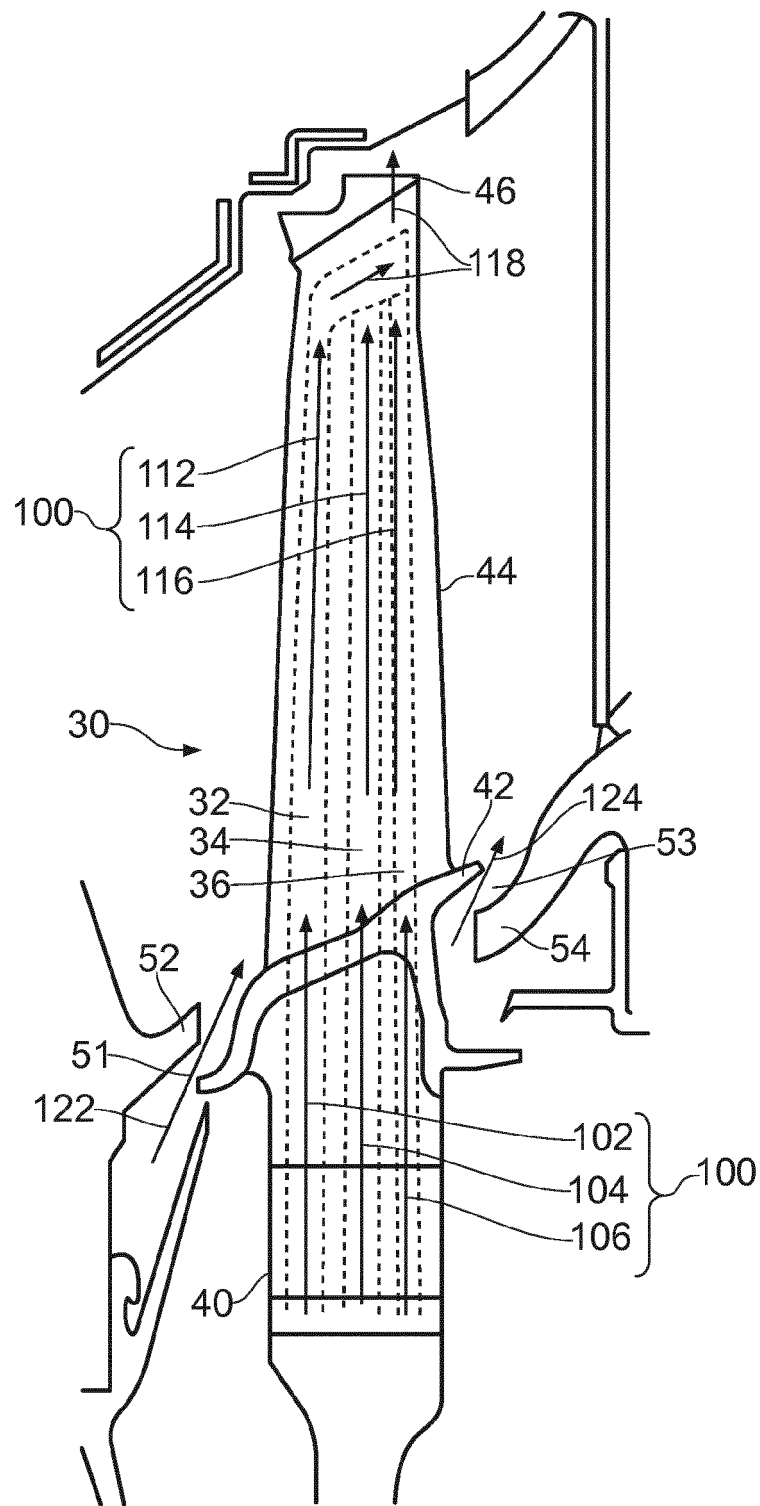


FIG. 2

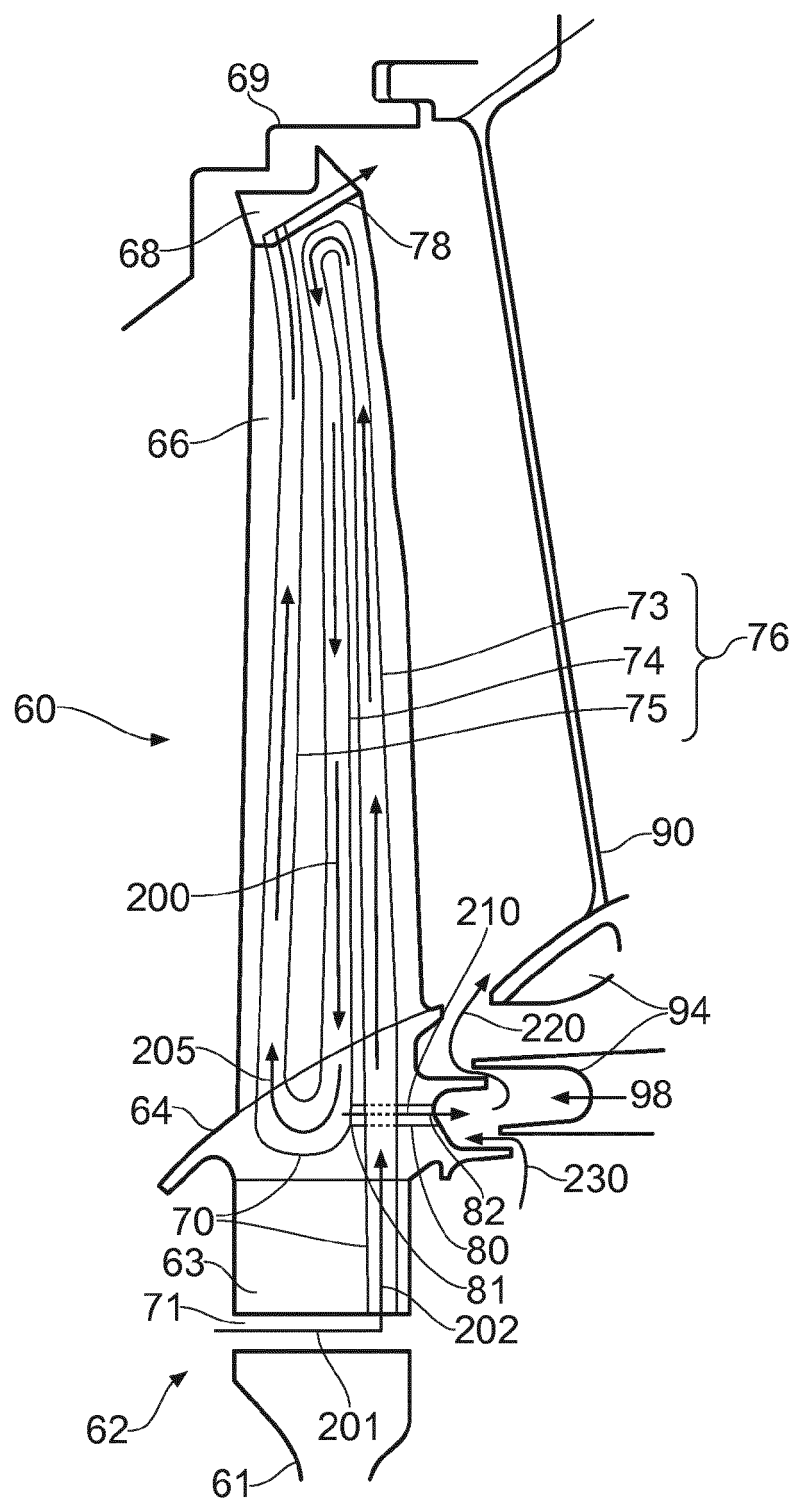


FIG. 3

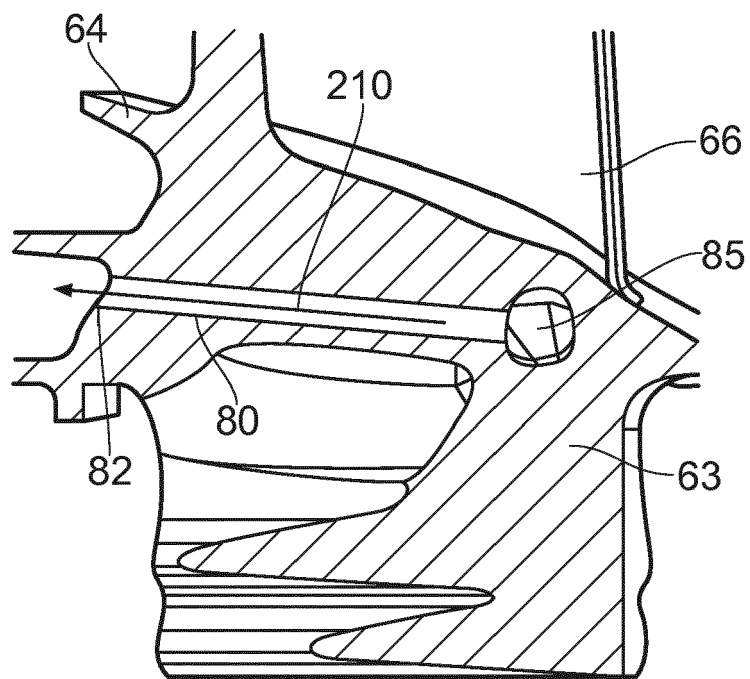


FIG. 4

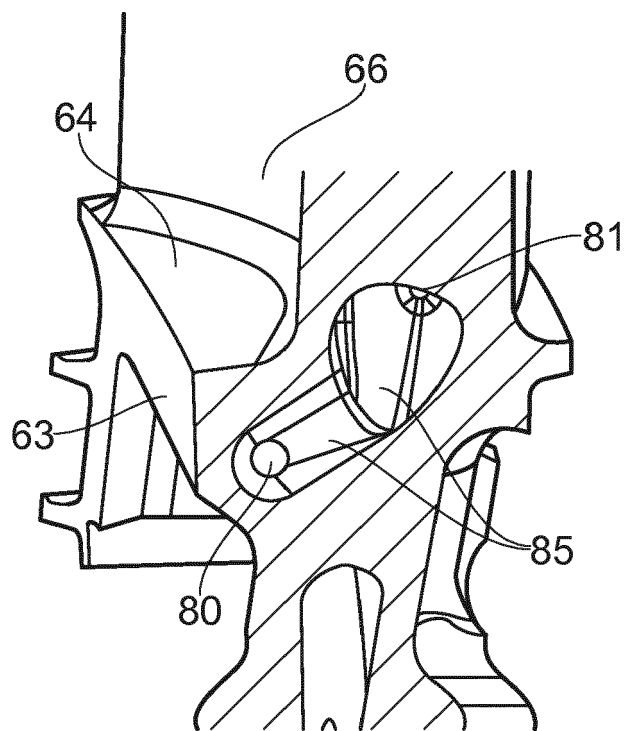


FIG. 5

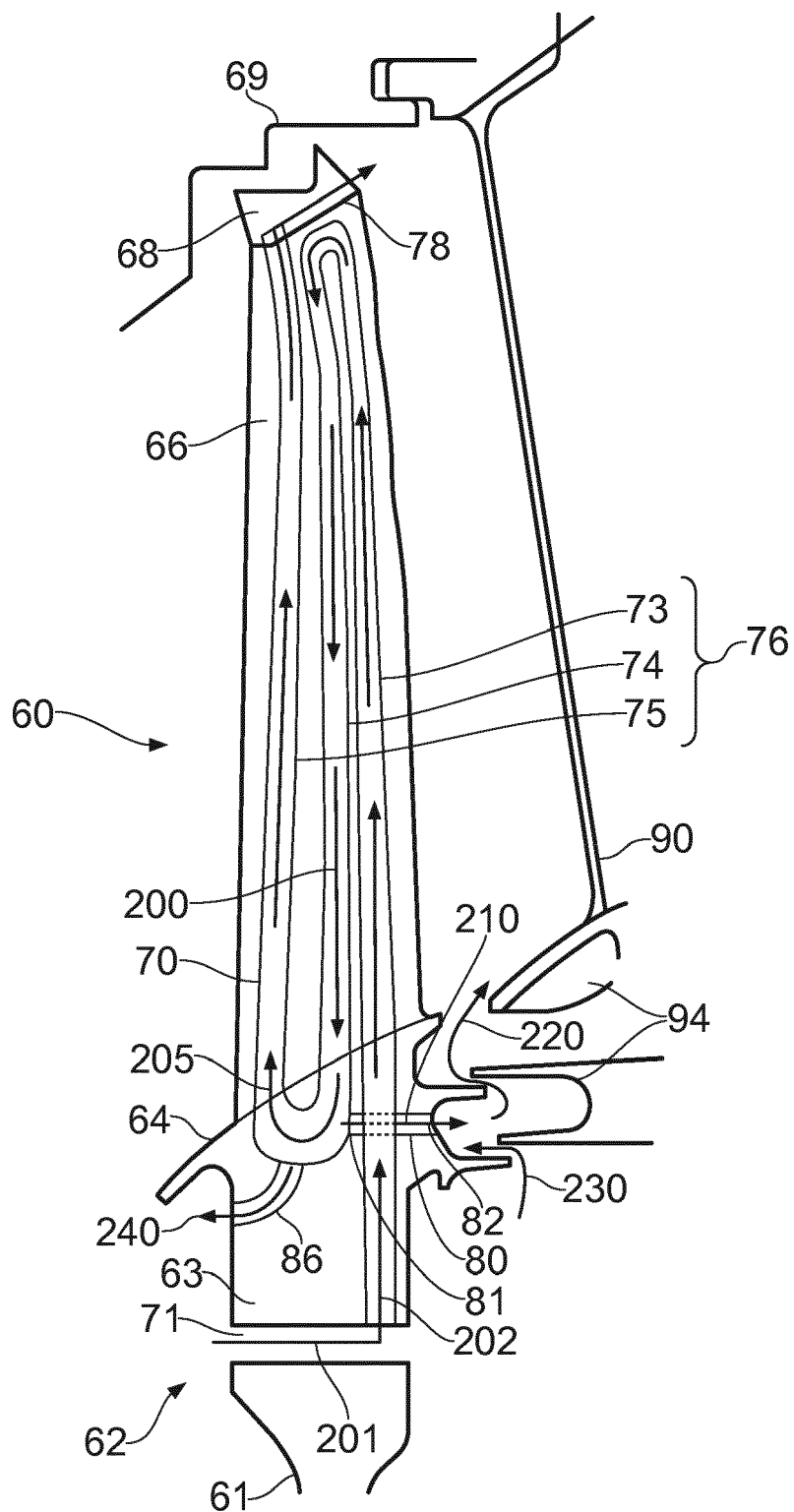


FIG. 6