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(54) **TURBINE ENGINE WITH A BLADE ASSEMBLY HAVING A PLATFORM PLENUM**

(57) A gas turbine engine having a blade assembly (30) with a platform (50), an airfoil (60), and a shank (40). The airfoil has a plurality of cooling conduits (70), and the shank has a plurality of inlet passages (48) to provide cooling fluid to the cooling conduits in the airfoil. The cooling fluid is vented through a plurality of cooling holes (69) along the trailing edge of the airfoil. The blade assembly has specific geometries that improve durability

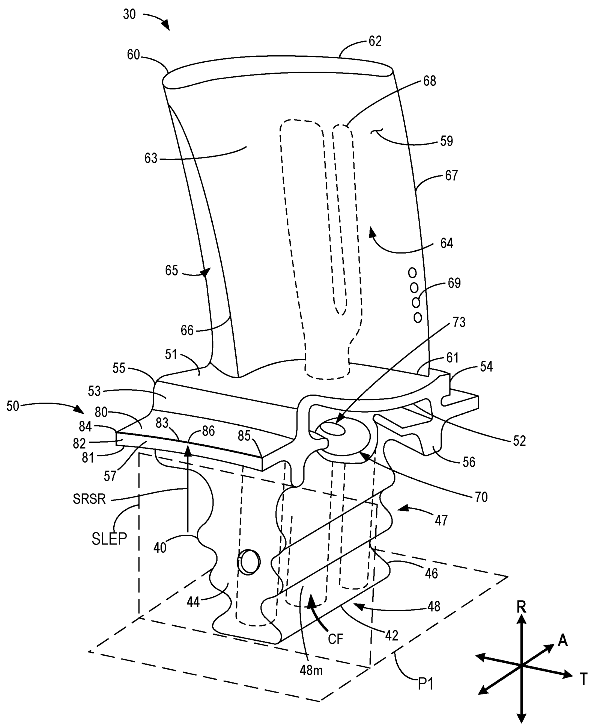


FIG. 3

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## Description

RELATED APPLICATIONS

**[0001]** This patent claims the benefit of U.S. Provisional Patent Application No. 63/597,832, titled "TURBINE ENGINE WITH A BLADE ASSEMBLY HAVING A PLATFORM PLENUM," which was filed on November 10, 2023, and U.S. Provisional Patent Application No. 63/686,055, titled "TURBINE ENGINE WITH A BLADE ASSEMBLY HAVING A PLATFORM PLENUM," which was filed on August 22, 2024. U.S. Provisional Patent Application Nos. 63/597,832 and 63/686,055 are hereby incorporated herein by reference in its entirety. Priority to U.S. Provisional Patent Application Nos. 63/597,832 and 63/686,055 is hereby claimed.

TECHNICAL FIELD

**[0002]** The present subject matter relates generally to a blade assembly for a turbine engine, and more specifically to a blade assembly with a platform plenum.

BACKGROUND

**[0003]** A gas turbine engine typically includes a turbomachine, with a fan in some implementations. The turbomachine generally includes a compressor, combustor, and turbine in serial flow arrangement. The compressor compresses air which is channeled to the combustor where it is mixed with fuel. The mixture is then ignited to generate hot combustion gases. The combustion gases are channeled to the turbine, which extracts energy from the combustion gases for powering the compressor and fan, if used, as well as for producing useful work to propel an aircraft in flight or to power a load, such as an electrical generator.

**[0004]** During operation of the gas turbine engine, various systems generate a relatively large amount of heat and stress. For example, a substantial amount of heat or stress can be generated during operation of the thrust generating systems, lubrication systems, electric motors and/or generators, hydraulic systems or other systems. A design that mitigates heat loads and/or stresses on an engine component is advantageous.

BRIEF DESCRIPTION OF THE DRAWINGS

**[0005]** A full and enabling disclosure of the present disclosure, including the best mode thereof, directed to one of ordinary skill in the art, is set forth in the specification, which makes reference to the appended figures, in which:

FIG. 1 is a schematic cross-sectional view of a gas turbine engine, in accordance with an exemplary embodiment of the present disclosure.

FIG. 2 is a schematic cross-sectional view of the turbine section of the gas turbine engine of FIG. 1, in accordance with an exemplary embodiment of the present disclosure.

FIG. 3 is a perspective view of a blade assembly for use in the gas turbine engine of FIG. 1, in accordance with an exemplary embodiment of the present disclosure.

FIG. 4 is a perspective front view of the blade assembly from FIG. 3 illustrating a platform plenum in dashed line and showing multiple planes, in accordance with an exemplary embodiment of the present disclosure.

FIG. 5 is an enlarged cross-sectional view of a feed conduit fluidly coupled to the platform plenum from FIG. 4, in accordance with an exemplary embodiment of the present disclosure.

FIG. 6 is a schematic used to calculate a stator rotor seal radius of the blade assembly of FIG. 3.

DETAILED DESCRIPTION

**[0006]** Reference will now be made in detail to present embodiments of the disclosure, one or more examples of which are illustrated in the accompanying drawings. The detailed description uses numerical and letter designations to refer to features in the drawings. Like or similar designations in the drawings and description have been used to refer to like or similar parts of the disclosure.

**[0007]** Aspects of the disclosure generally relate to a blade assembly having conduits located within the blade assembly. Specifically, the blade assembly includes an airfoil with a plurality of cooling conduits. The airfoil also includes cooling holes fluidly coupled to the plurality of cooling conduits within the airfoil.

**[0008]** The blade assembly may be a blade assembly in a turbine section of a gas turbine engine. For example, the blade assembly may be a stage one blade assembly of a high pressure turbine, which typically experiences the highest thermal and mechanical stresses.

**[0009]** The blade assembly includes a shank and a platform. The shank is used to attach the blade assembly to a turbine disk. In some implementations the shank is formed as a dovetail received in the turbine disk.

**[0010]** The platform of the blade assembly, together with other circumferentially arranged platforms of other blade assemblies, defines a continuous annular ring that prevents hot gas leakage into the turbine disk cavity and/or a stator ring of the gas turbine engine. The airfoil extends radially from the platform, away from the turbine disk, while the shank extends radially from the platform, toward the turbine disk.

**[0011]** High engine temperatures and operational forces impart relatively large thermal and mechanical stresses on the blade assemblies. In addition, the cooling conduits in the blade assembly create stress concentrations. For example, the size of the cooling conduits affects the thickness of the airfoil wall, which affects stress concentrations in the airfoil. Relatively large stresses can contribute to an unexpected or premature part replacement. Therefore, there is a need for a blade assembly with greater durability to increase time on wing.

**[0012]** Aspects of the disclosure generally relate to a blade assembly having a platform plenum within. Specifically, the blade assembly includes a platform with the platform plenum formed between an airfoil and a shank of the blade assembly. Traditionally, the airfoil also includes cooling holes fluidly coupled to the set of cooling conduits within. During operation, particulate matter can accumulate in the platform plenum and other portions of the blade assembly. Such accumulation can reduce the flow of fluid through the cooling conduits of a blade assembly and increase the thermal stress on the blade assembly. There is a need for blade assemblies that mitigate particulate accumulation without significant sacrifices to other design parameters.

**[0013]** Connection references (e.g., attached, coupled, connected, and joined) are to be construed broadly and can include intermediate structural elements between a collection of elements and relative movement between elements unless otherwise indicated. As such, connection references do not necessarily infer those two elements are directly connected and in fixed relation to one another. The exemplary drawings are for purposes of illustration only and the dimensions, positions, order and relative sizes reflected in the drawings attached hereto can vary.

**[0014]** As used herein, a "stage" of either a compressor or a turbine of a gas turbine engine is a set of blade assemblies and an adjacent set of vane assemblies, with both sets of the blade assemblies and the vane assemblies circumferentially arranged about an engine centerline. A pair of circumferentially-adjacent vanes in the set of vane assemblies are referred to as a nozzle. The blade assemblies rotate relative to the engine centerline and, in one example, are mounted to a rotating structure, such as a disk, to affect the rotation.

**[0015]** As used herein, the word "exemplary" means "serving as an example, instance, or illustration." Any implementation described herein as "exemplary" is not necessarily to be construed as preferred or advantageous over other implementations. Additionally, unless specifically identified otherwise, all embodiments described herein should be considered exemplary.

**[0016]** As used herein, the terms "first", "second", "third", and "fourth" can be used interchangeably to distinguish one component from another and are not intended to signify location or importance of the individual components.

**[0017]** As used herein, a "set" or a set of elements can include any number of said elements, including one.

**[0018]** As used herein, the terms "forward" and "aft" refer to relative positions within a gas turbine engine and refer to the normal operational attitude or direction of travel of the gas turbine engine. For example, with regard to a gas turbine engine, forward refers to a position relatively closer to the nose of an aircraft and aft refers to a position relatively closer to a tail of the aircraft.

**[0019]** As used herein, the terms "upstream" and "downstream" refer to a direction with respect to a direction of fluid flow along a flowpath.

**[0020]** As used herein, the term "fluid" refers to a gas or a liquid and "fluidly coupled" means a fluid can flow between the coupled regions.

**[0021]** As used herein, forms "a", "an", and "the" include plural references unless the context clearly dictates otherwise.

**[0022]** As used herein, a radial direction (denoted "R") is a direction that is perpendicular to a base plane on a shank of a blade assembly.

**[0023]** As used herein, an axial direction (denoted "A") is a direction that is perpendicular to a shank leading-edge plane on the shank of the blade assembly.

**[0024]** As used herein, a tangential direction (denoted "T") is a direction that is perpendicular to the radial direction and the axial direction.

**[0025]** A stator rotor seal radius (denoted *SRSR*) is a radius of curvature of an upper edge of a stator rotor seal on a blade assembly.

**[0026]** As used herein "cooling conduit" refers to a flow path that conveys a cooling fluid that is formed in a blade assembly.

**[0027]** As used herein "inlet passage" refers to a cooling conduit formed in a shank of the blade assembly.

**[0028]** As used herein "platform plenum" refers to a cavity radially inward of an upper surface of a platform for distribution of a cooling fluid throughout a blade assembly.

**[0029]** As used herein, "feed conduit" refers to a cooling conduit extending between an inlet passage and a platform plenum. As used herein "a feed inlet" refers to the end of the feed conduit at the inlet passage and "a feed outlet" refers to the end of the feed conduit at the platform plenum.

**[0030]** A feed centerline (denoted "FCL") refers to a line extending through a geometric center of a feed inlet and a feed outlet of the feed conduit.

**[0031]** A minimum feed cross-sectional area (denoted *FA*) is the smallest cross-sectional area of a feed conduit taken in a plane perpendicular to a feed centerline of the feed conduit.

**[0032]** A feed angle (denoted $\theta$) is an angle measured from a plane parallel to a base of a shank of a blade assembly and a feed centerline (FCL) of a feed conduit of the blade assembly.

**[0033]** The term redline exhaust gas temperature (referred to herein as "redline EGT") refers to a maximum permitted takeoff temperature documented in a Federal Aviation Administration ("FAA")-type certificate data sheet. For example, in certain exemplary embodiments, the term redline EGT may refer to a maximum permitted takeoff temperature of an airflow after a first stage stator downstream of an HP turbine of an engine that the engine is rated to withstand. The term redline EGT is sometimes also referred to as an indicated turbine exhaust gas temperature or indicated turbine temperature.

**[0034]** All measurements referred to herein are taken of the blade assembly prior to use or as a cold component.

**[0035]** Referring now to the drawings, FIG. 1 is a schematic view of a gas turbine engine 10. As a non-limiting example, the gas turbine engine 10 can be used on an aircraft. The gas turbine engine 10 includes an engine core extending along an engine centerline 20 and including, at least, a compressor section 12, a combustor 14, and a turbine section 16 in serial flow arrangement. In some examples, the gas turbine engine 10 includes a fan (not shown) that is driven by the engine core to produce thrust and provide air to the compressor section 12. The gas turbine engine 10 includes a drive shaft 18 that rotationally couples the fan, compressor section 12, and turbine section 16, such that rotation of one affects the rotation of the others, and defines a rotational axis along the engine centerline 20 of the gas turbine engine 10.

**[0036]** In the illustrated example, the compressor section 12 includes a low-pressure (LP) compressor 22 and a high-pressure (HP) compressor 24 serially fluidly coupled to one another. The turbine section 16 includes an HP turbine 26 and a LP turbine 28 serially fluidly coupled to one another. The drive shaft 18 operatively couples the LP compressor 22, the HP compressor 24, the HP turbine 26 and the LP turbine 28 to one another. In some implementations, the drive shaft 18 includes an LP drive shaft (not illustrated) and an HP drive shaft (not illustrated), where the LP drive shaft couples the LP compressor 22 to the LP turbine 28, and the HP drive shaft couples the HP compressor 24 to the HP turbine 26.

**[0037]** The compressor section 12 includes a plurality of axially spaced stages. Each stage includes a set of circumferentially-spaced rotating blade assemblies and a set of circumferentially-spaced stationary vane assemblies. In one configuration, the compressor blade assemblies for a stage of the compressor section 12 are mounted to a disk, which is mounted to the drive shaft 18. Each set of blade assemblies for a given stage can have its own disk. In one implementation, the vane assemblies of the compressor section 12 are mounted to a casing which extends circumferentially about the gas turbine engine 10. In a counter-rotating turbine engine, the vane assemblies are mounted to a drum, which is similar to the casing, except the drum rotates in a direction opposite the blade assemblies, whereas the casing is stationary. It will be appreciated that the representation of the compressor section 12 is merely schematic. The number of stages can vary.

**[0038]** Similar to the compressor section 12, the turbine section 16 includes a plurality of axially spaced stages, with each stage having a set of circumferentially-spaced, rotating blade assemblies and a set of circumferentially-spaced, stationary vane assemblies. In one configuration, the turbine blade assemblies for a stage of the turbine section 16 are mounted to a disk which is mounted to the drive shaft 18. Each set of blade assemblies for a given stage can have its own disk. In one implementation, the vane assemblies of the turbine section are mounted to the casing in a circumferential manner. In a counter-rotating turbine engine, the vane assemblies can be mounted to a drum, which is similar to the casing, except the drum rotates in a direction opposite the blade assemblies, whereas the casing is stationary. The number of blade assemblies, vane assemblies, and turbine stages can vary.

**[0039]** The combustor 14 is provided serially between the compressor section 12 and the turbine section 16. The combustor 14 is fluidly coupled to at least a portion of the compressor section 12 and the turbine section 16 such that the combustor 14 at least partially fluidly couples the compressor section 12 to the turbine section 16. As a non-limiting example, the combustor 14 is fluidly coupled to the HP compressor 24 at an upstream end of the combustor 14 and to the HP turbine 26 at a downstream end of the combustor 14.

**[0040]** During operation of the gas turbine engine 10, ambient or atmospheric air is drawn into the compressor section 12 via the fan, upstream of the compressor section 12, where the air is compressed defining a pressurized air. The pressurized air then flows into the combustor 14 where the pressurized air is mixed with fuel and ignited, thereby generating hot combustion gases. Some work is extracted from these combustion gases by the HP turbine 26, which drives the HP compressor 24. The combustion gases are discharged into the LP turbine 28, which extracts additional work to drive the LP compressor 22, and the exhaust gas is ultimately discharged from the gas turbine engine 10 via an exhaust section (not illustrated) downstream of the turbine section 16. The driving of the LP turbine 28 drives the LP spool to rotate the fan and the LP compressor 22. The pressurized airflow and the combustion gases together define a working airflow that flows through the fan, compressor section 12, combustor 14, and turbine section 16 of the gas turbine engine 10.

[0041] Turning to FIG. 2, a portion of the turbine section 16 is schematically illustrated. The turbine section 16 includes sets of blade assemblies 30 circumferentially mounted to corresponding disks 32. The number of individual blade assemblies of the set of blade assemblies 30 mounted to each disk 32 may vary. While shown schematically in FIG. 2, it should be understood that the turbine section 16 can be a single stage turbine, or can include additional stages as shown.

[0042] Stationary vane assemblies 34 are mounted to a stator ring 36 located distally exterior of each of the disks 32. A nozzle 38 is defined by the space between circumferentially-adjacent pairs of stationary vane assemblies 34. The number of nozzles 38 provided on the stator ring 36 may vary.

[0043] During operation of the gas turbine engine 10, a flow of hot gas or heated fluid flow (denoted "HF") exits the combustor 14 and enters the turbine section 16. The heated fluid flow HF is directed through the nozzles 38 and impinges on the blade assemblies 30, which rotates the blade assemblies 30 circumferentially around the engine centerline 20 and cause rotation of the drive shaft 18. The engine core is configured to generate a redline exhaust gas temperature (EGT) in a range of 988 degrees Celsius (°C) to 1120°C

[0044] FIG. 3 is a perspective view of a single blade assembly 30 (FIG. 2) for the gas turbine engine 10 (FIG. 1). The blade assembly 30 may correspond to a stage one blade assembly of the HP turbine 26. The blade assembly 30 includes a shank 40, a platform 50, and an airfoil 60 on the platform 50. The blade assembly 30 can be constructed as a single unitary part or component (e.g., a monolithic structure). In other examples, the shank 40, the platform 50, and/or the airfoil 60 can be constructed as separate parts or components that are coupled together to form the blade assembly 30.

[0045] A directional reference system is illustrated in FIG. 3. The shank 40 extends between a base 42 and the platform 50. The base 42 of the shank 40 is a flat surface that defines a plane, referred to herein interchangeably as the base plane or the first plane (denoted "P1"). A radial direction (denoted "R") of the blade assembly 30 is a direction that is perpendicular to the base plane BP. Further, the shank 40 extends between a shank leading-edge 44 and a shank trailing-edge 46. The shank leading-edge 44 is a flat surface that defines a plane, referred herein as the shank leading-edge plane (denoted "SLEP"). An axial direction (denoted "A") of the blade assembly 30 is a direction that is perpendicular to the shank leading-edge plane SLEP. A tangential direction (denoted "T") is a direction perpendicular to both the radial direction R and the axial direction A.

[0046] The shank 40 is between a base 42 and the platform 50 in the radial direction. The shank 40 extends between a shank leading-edge 44 and a shank trailing-edge 46 in the axial direction. The shank 40 is configured to mount to the disk 32 (FIG. 2) of the engine 10 in order to rotatably drive the blade assembly 30. In the illustrated example of FIG. 3, the shank 40 is a dovetail. In other examples, the shank 40 can have a different shape, such as a firtree or a bulb. The shank 40 includes a set of inlet passages 48 for receiving a cooling fluid (denoted "CF") for cooling the blade assembly 30. In the illustrated example of FIG. 3, the set of inlet passages 48 can include 3 inlet passages (e.g., a leading-edge inlet passage, a middle inlet passage, and a trailing-edge inlet passage, etc.)

[0047] The airfoil 60 meets the platform 50 to define a root 61 and spans to a tip 62. Additionally, the airfoil 60 includes an outer wall 63 defining an exterior surface 59 including a pressure side 64 and a suction side 65. The airfoil 60 extends between an airfoil leading-edge 66 and an airfoil trailing-edge 67 downstream from the airfoil leading-edge 66. The airfoil leading-edge 66 and the airfoil trailing-edge 67 separate the pressure side 64 from the suction side 65. A set of cooling conduits 68 is formed within the airfoil 60. Any number of cooling holes 69 can be formed in the outer wall 63 to fluidly couple the set of cooling conduits 68 within airfoil 60 of the blade assembly 30 to an exterior of the blade assembly 30.

[0048] The platform 50 has an upper surface 51 (e.g., a first surface, etc.) and a lower surface 52 (e.g., a second surface, etc.) and extends between a platform leading-edge 53 and a platform trailing-edge 54 in the axial direction.

[0049] The platform 50 extends between a platform leading-edge 53 and a platform trailing-edge 54, opposite the platform leading-edge 53, in the axial A direction. The platform 50 further extends between a first slashface 55 and a second slashface 56, opposite the first slashface 55, in the tangential T direction. When assembled, consecutive blade assemblies 30 are arranged in a circumferential direction about the engine centerline 20 (FIG. 1) with sequential slashfaces 55, 56 facing each other.

[0050] A platform plenum 70 is formed below the lower surface 52 and is fluidly coupled to the set of cooling conduits 68 and to the set of inlet passages 48 via a feed outlet 73. The platform plenum 70 is sealed off during manufacturing.

[0051] In operation, a heated fluid flow HF, such as a combustor flow, flows along the blade assembly 30. The airfoil leading-edge 66 is defined by a stagnation point with respect to the heated fluid flow HF. The heated fluid flow HF flows generally in the axial direction, from forward to aft, while the local directionality can vary as the fluid flow HF is driven or turned within the gas turbine engine 10. The cooling fluid flow CF is fed to the set of inlet passages 48 and flows into the set of cooling conduits 68 to cool the airfoil 60. The cooling fluid flow CF is provided throughout the airfoil 60 and exhausted from the set of cooling conduits 68 via the cooling holes as a cooling film. The platform 50 helps to radially contain the gas turbine engine 10 mainstream heated fluid flow HF acting to protect the disk 32. The platform 50 acts to seal the space radially inward of the platform 50 between the flow path of the heated fluid flow HF and the disk 32. The disk 32 requires significant cooling to ensure the durability of the HP turbine 26 components.

[0052] Materials used to form the blade assembly 30 include, but are not limited to, steel, refractory metals such as titanium, or superalloys based on nickel, cobalt, or iron, ceramic matrix composites, or combinations thereof. The structures can be formed by a variety of methods, including additive manufacturing, casting, electroforming, or direct metal laser melting, in non-limiting examples.

**[0053]** Turning to FIG. 4, a front perspective view of the blade assembly 30 is illustrated. A platform plenum 70 is illustrated in phantom and located within the platform 50. A feed conduit 71 extends between a feed inlet 72 and the feed outlet 73. The feed inlet 72 is fluidly coupled to the set of inlet passages 48. In FIG.4, the feed inlet 72 is fluidly coupled to the middle inlet passage 48m of FIG. 3. In other examples, the feed inlet 72 is coupled to a different one of the set of inlet passages 48. The middle inlet passage 48m is located mid-way between the shank leading-edge 44 and the shank trailing-edge 46. The feed centerline FCL extends through a first geometric center 74 (FIG. 5) of the feed inlet 72 and a second geometric center 75 (FIG. 5) of the feed outlet 73.

**[0054]** The first plane (denoted "P1") of FIG. 3 is illustrated in FIG. 3. A second plane (denoted "P2") is defined as the plane that is parallel to the first plane and intersects the feed centerline FCL at the first geometric center 74. The second plane is perpendicular to the radial direction.

**[0055]** Turning to FIG. 5, an enlarged view of the feed conduit 71 is illustrated. It can more clearly be seen that the feed centerline FCL and the second plane P2 form the feed angle $\theta$ therebetween. The minimum feed cross-sectional area *FA* is measured at a location 76 where the cross-sectional area of the feed conduit 71 is smallest. The minimum feed cross-sectional area *FA* is measured in the plane that is perpendicular to the feed centerline FCL.

**[0056]** Returning to FIG. 3, the platform 50 has a stator rotor seal 57 that extends axially forward from the platform leading-edge 53. The stator rotor seal 57 facilitates sealing of a forward wing buffer cavity (not shown) defined within the rotor assembly. The stator rotor seal 57 has an upper surface 80, a lower surface 81 opposite the upper surface 80, and a forward surface 82 between the upper surface 80 and the lower surface 81. The stator rotor seal 57 has an upper edge 83 between the upper surface 80 and the forward surface 82. The upper edge 83 is curved or arc- shaped. In particular, the upper edge 83 is curved between a first end point 84 at the first slashface 55 and a second end point 85 at the second slashface 56. The upper edge 83 of stator rotor seal 57 has a center point 86 that forms the peak of the arc. The upper edge 83 of the stator rotor seal 57 has a radius of curvature, referred to herein as a stator rotor seal radius (denoted "SRSR"). The center of the radius of curvature may be the engine centerline 20 (FIG. 1). As shown in FIG. 4, the SRSR (i.e., the radius of curvature of the upper edge 83 of the stator rotor seal 57) can be calculated using the straight-line distance (S) between the two the end points 84, 85, and the maximum deflection (D), in the radial R direction, between the two end points 84, 85 and the center point 86 of the arc. The SRSR can be calculated using SRSR = (D/2) + ($S^2$ / (8xD)).

**[0057]** The blade assemblies 30 of the HP turbine 26 and, specifically, the stage one blade assemblies 30 are exposed to the highest temperatures in the gas turbine engine 10. These stage one blade assemblies also rotate at extremely high angular velocities. The extreme temperature environment and the high rotational speeds impart large forces on the blade assemblies 30 that can lead to creep and fatigue, especially along the suction side of the airfoil. Creep and fatigue may result in unintended engine removals for inspections and/or serving that limit engine Time on Wing (TOW). Therefore, there is a need for a blade assembly with high durability that can withstand these large centrifugal stresses and reduce (e.g., minimize) creep and fatigue.

**[0058]** To mitigate creep and fatigue, some blade assemblies include cooling networks formed within various parts of the blade assembly to facilitate the flow of cooling fluid throughout the blade assembly. Cooling fluid is introduced to the blade assembly via inlet passages and fed to various locations. The cooling fluid can include particulates, which can accumulate within areas of the blade assembly. The accumulation of such particulates within the blade assembly can prevent the desired cooling by the flow cooling fluid and result in elevated temperatures that drive lower local part durability. While changing the geometry of the cooling conduits can mitigate particulate accumulation, such changes can result in the backflow margin decreasing to inadequate levels. Low backflow margins can result in the ingestion of hot combustion gas into the blade assembly, which can increase the temperature of the blade assembly and reduce part durability. Mitigation of particulate build-up without sacrificing backflow margin is necessary to increase effective cooling and prevent creep and fatigue.

**[0059]** The inventors have found solutions that decrease the feed angle ($\Theta$) at which the cooling fluid is introduced into the platform plenum 70, which provides an indirect path for particulates to enter the platform plenum 70. The feed angle $\theta$ influences the particulate accumulation with the blade assembly 30, which influences the temperature of the blade assembly and the durability thereof. Greater feed angles $\theta$ are associated with a more direct path for particulates to enter the platform plenum. Accordingly, lowering the feed angle $\theta$ can reduce the accumulation of particles with the blade assembly 30. The inventors have further found that reducing the minimum feed cross-sectional area (FA) of the feed conduit 71 reduces the amount of particulates entering the feed conduit 71. Particularly, the minimum feed cross-sectional area FA influences the particulate accumulation with the blade assembly 30, which influences the temperature of the blade assembly and the durability thereof. Lowering the minimum feed cross-sectional area FA reduces particulate flows, but can reduce the BFM.

**[0060]** The engine exhaust temperature EGT affects the overall temperature on the blade. The higher the temperature for a given stress, the weaker the material from which the blade is formed becomes, making the blade more prone to failure. Further, the inventors determined, through developing multiple blade assembly designs, that the size of the SRSR has a significant effect on the durability of the blade assembly 30. The SRSR is integral to the airfoil 60 external geometry and characterizes the component height in operation. The airfoil 60 is designed for rotational operation and this SRSR relates to the loading characteristics experienced by the airfoil 60. Due to the relationship with airfoil height and rotational operation, the SRSR can be used to characterize the loading and stresses of the airfoil as the primary contributors to airfoil stress are due to rotation, flowpath, and thermal conditions. The stress experienced by the airfoil contributes to component durability.

**[0061]** Therefore, the inventors determined during the course of their blade assembly design that the feed angle and the minimum feed cross-sectional area *FA* of the feed conduit 71 of FIG. 5, the SRSR of FIG. 3 and 6, and the redline EGT have an effect on the particulate accumulation within the blade assembly 30 and durability thereof.

**[0062]** As stated above, the inventors created solutions with relatively high blade durability (e.g., lower particulate accumulation, reduced creep and fatigue, absence of crack formation or propagation after a number of engine cycles) for a defined engine environment. Table 1 below illustrates eighteen examples (denoted Ex. 1-18) of gas turbine engines with blade assemblies developed by the inventors. Table 1 includes feed cross-sectional area values, feed angle values, exhaust gas temperature values, and stator rotor seal radius values for each of the examples.

TABLE 1

| Parameter | *Θ (Feed Angle)* | *FA (Feed Cross-sectional Area)* | *EGT (Exhaust Gas Temperature)* | *SRSR (Stator Rotor Seal Radius)* |
|---|---|---|---|---|
| Parameter Units | degrees (°) | Square Meters ($m^2$ | Degrees Celsius °C | Meters (m) |
| Ex. 1 | 0.01 | 2.00E-06 | 1120 | 0.239 |
| Ex. 2 | 78 | 3.50E-06 | 988 | 0.224 |
| Ex. 3 | 10 | 3.00E-06 | 1074 | 0.236 |
| Ex. 4 | 60 | 2.80E-06 | 991 | 0.226 |
| Ex. 5 | 0.01 | 2.80E-06 | 1100 | 0.235 |
| Ex. 6 | 35 | 2.10E-06 | 990 | 0.236 |
| Ex. 7 | 17 | 3.50E-06 | 1050 | 0.234 |
| Ex. 8 | 24 | 2.00E-06 | 1105 | 0.237 |
| Ex. 9 | 64 | 3.10E-06 | 1043 | 0.236 |
| Ex. 10 | 51 | 3.40E-06 | 1000 | 0.224 |
| Ex. 11 | 55.649 | 2.27E-06 | 995.066 | 0.239 |
| Ex. 12 | 71.448 | 2.17E-06 | 1065.709 | 0.233 |
| Ex. 13 | 0.435 | 2.04E-06 | 999.45 | 0.225 |
| Ex. 14 | 77.562 | 2.02E-06 | 1110.129 | 0.238 |
| | | | | |
| Ex. 15 | 82 | 6.50E-07 | 988 | 0.239 |
| Ex. 16 | 87 | 1.81E-06 | 1105 | 0.231 |
| Ex. 17 | 102 | 8.51E-07 | 1120 | 0.224 |
| Ex. 18 | 99 | 1.28E-06 | 1089 | 0.229 |

**[0063]** The inventors found that blade assembly designs with parameters defined in Examples 1-14 exhibit relatively high structural integrity, low particulate accumulation, and durability while remaining within current engine constraints. Conversely, Examples 15-18 have relatively low durability for the particular engine environment.

**[0064]** The examples developed by the inventors shown in Table 1 can be characterized by an Expression (EQ) that can be used to distinguish those designs in Examples 1-14 that meet the performance (durability) requirements from those designs in Examples 15-18 that do not meet the performance requirements. As such, the Expression (EQ) can be used to identify an improved blade assembly design, better suited for a particular engine operating environment and taking into account the constraints imposed on blade assembly design with cooling holes used in such a system.

**[0065]** The Expression (EQ) is defined as:

$$EQ = \left( \left( \frac{FA(m^2)}{0.01m^2} \right)^{-1} \left( \frac{\theta(°) - 79°}{79°} \right) \right) \times \left( 1.7 \times \frac{Redline\ EGT\,(°C)}{500°C} \times \frac{SRSR(m)}{1m} \right) \quad (1),$$

*FA* represents the minimum feed cross-sectional area of the feed conduit 71 of FIG. 5. θ represents the feed angle of the feed conduit 71 with respect to the second plane P2 of the feed conduit 71 of FIG. 5. SRSR represents the stator rotor seal radius of FIGS. 3 and 6. Redline EGT represents the redline exhaust gas temperature for the gas turbine engine 10.

**[0066]** Values for (EQ) for each of the examples of Table 1 are provided in Table 2 below:

TABLE 2

| Parameter | *Θ (Feed Angle)* | *FA (Feed Cross-sectional Area)* | *EGT (Exhaust Gas Temperature)* | *SRSR (Stator Rotor Seal Radius)* | *Expression 1 (EQ)* |
|---|---|---|---|---|---|
| Parameter Units | degrees (°) | Square Meters ($m^2$ | Degrees Celsius °C | Meters (m) | Dimensionless |
| Ex. 1 | 0.01 | 2.00E-06 | 1120 | 0.239 | -4549.984 |
| Ex. 2 | 78 | 3.50E-06 | 988 | 0.224 | -27.214 |
| Ex. 3 | 10 | 3.00E-06 | 1074 | 0.236 | -2508.973 |
| Ex. 4 | 60 | 2.80E-06 | 991 | 0.226 | -654.078 |
| Ex. 5 | 0.01 | 2.80E-06 | 1100 | 0.235 | -3138.531 |
| Ex. 6 | 35 | 2.10E-06 | 990 | 0.236 | -2106.844 |
| Ex. 7 | 17 | 3.50E-06 | 1050 | 0.234 | -1873.185 |
| Ex. 8 | 24 | 2.00E-06 | 1105 | 0.237 | -3099.525 |
| Ex. 9 | 64 | 3.10E-06 | 1043 | 0.236 | -512.402 |
| Ex. 10 | 51 | 3.40E-06 | 1000 | 0.224 | -793.924 |
| Ex. 11 | 55.649 | 2.27E-06 | 995.066 | 0.239 | -1054.279 |
| Ex. 12 | 71.448 | 2.17E-06 | 1065.709 | 0.233 | -372.091 |
| Ex. 13 | 0.435 | 2.04E-06 | 999.45 | 0.225 | -3721.827 |
| Ex. 14 | 77.562 | 2.02E-06 | 1110.129 | 0.238 | -81.150 |
| | | | | | |
| Ex. 15 | 82 | 6.50E-07 | 988 | 0.239 | 469.045 |
| Ex. 16 | 87 | 1.81E-06 | 1105 | 0.231 | 485.554 |
| Ex. 17 | 102 | 8.51E-07 | 1120 | 0.224 | 2918.207 |
| Ex. 18 | 99 | 1.28E-06 | 1089 | 0.229 | 1677.470 |

**[0067]** Based on the characteristics of Examples 1-14, it was determined that gas turbine engine and blade assembly designs with an EQ value in the range of -4549.984 to -27.214 (i.e., - 4549.984 ≤ EQ ≤ -27.214) advantageously meet the durability constraints while remaining within desired tolerances and being capable of use in existing engine systems.

**[0068]** Benefits are realized when the manufactured component including the blade assembly 30 have a geometry where Expression (EQ) falls within the range -4549.984 to -27.214 (i.e., - 4549.984 ≤ EQ ≤ -27.214). In particular, such blade assemblies have low particulate accumulation in the platform 50, which increases effectiveness of the cooling of the blade assembly, which diminishes the propensity for creep and fatigue in the blade assembly 30, which increases the durability of the blade assembly. The improved durability increases the life of the blade assembly 30, which decreases required maintenance and costs, while increasing overall engine reliability and time one wing (TOW).

**[0069]** Further still, the benefits included herein provide for a blade assembly 30 that fits within existing engines. For example, the values for Expression (EQ) as provided herein take existing engines into consideration, permitting replacement of current blade assemblies with replacement blade assemblies (or new blade assemblies) having the parameters of the blade assembly 30 described herein. Such consideration provides for replacing and improving current engine systems without requiring the creation of new engine parts capable of holding the blade assembly 30. This provides for improving current engine durability without increasing costs to prepare new engines or further adapt existing engines.

**[0070]** Table 3 below illustrates minimum and maximum value ranges for the feed cross-sectional area *FA,* the feed angle $\Theta$, the stator rotor seal radius *SRSR,* and the redline exhaust gas temperature *EGT* along with a range of values for Expression (EQ) suited for a blade assembly 30 that meets durability constraints.

TABLE 3

| Parameter: | Engine Element: | Minimum: | Maximum: | Units: |
|---|---|---|---|---|
| *FA* | Feed Cross-Sectional Area | 2.00E-6 | 3.50E-6 | Meters squared ($m^2$) |
| $\theta$ | Feed Angle | 0.0100 | 78 | degrees (°) |
| *EGT* | Exhaust Gas Temperature | 988 | 1120 | Degrees Celsius (°C) |
| *SRSR* | Stator rotor seal radius | 0.224 | 0.2383 | Meters (m) |
| *EQ* | Expression 1 | -4549.984 | -27.214 | n/a |

**[0071]** Additional benefits associated with the blade assembly 30, with the feed conduit 71, and with the stator rotor seal 57 described herein include a quick assessment of design parameters in terms of blade assembly size and cooling conduit geometry, engine operational conditions, and blade and vane assembly numbers for engine design and particular blade design. Narrowing these multiple factors to a region of possibilities saves time, money, and resources. The blade assembly 30 with the feed conduit 71 and the stator rotor seal 57 described herein enables the development and production of high-performance turbine engines and blade assemblies across multiple performance metrics within a given set of constraints.

**[0072]** As noted above, designs, such as Examples 15-18 of Tables 1 and 2, were found to have greater particulate accumulation relatively low durability for the particular engine environment. This is reflected in associated Expression (EQ) values outside the range of - 4549.984 to -27.214 (i.e., $-4549.984 \leq EQ \leq -27.214$). Lower durability results in less time on wing (TOW) and greater maintenance costs.

**[0073]** Additionally or alternatively, designs outside the ranges of EQ may attempt to increase durability by making sacrifices in terms of weight, aerodynamic performance, and efficiency. For example, the standard practice for solving the problem of improving blade assembly durability has been to utilize stronger material. However, such materials lead to increased costs, system weight, and overall space occupied by the blade assembly. Using a cost-benefit analysis, the overall engine efficiency may be reduced and related components may have to be redesigned to compensate for the stronger materials. In some cases, this result of such a cost-benefit analysis is impractical or impossible. Therefore, a solution for reducing stresses located in airfoils presently used in existing engines is needed, without requiring redesign of related components or without sacrificing overall engine efficiency.

**[0074]** In other examples, increasing size of the airfoil or related components, utilizing stronger material, and/or providing additional cooling features can combat centrifugal and thermal stresses. However, such increased size, stronger materials, and additional cooling features can lead to increased costs, system weight, overall space occupied by the blade assembly, and performance loss, as well as increased local stresses at the cooling conduits due to increased weight and size relating to the centrifugal forces. Increased cooling features results in a relatively less amount of material utilized, which can result in an increase in local stresses at the cooling conduits. Therefore, a solution for reducing stresses at the cooling conduits is needed without otherwise increasing stresses, weight, size, or decreasing engine efficiency.

**[0075]** As disclosed above, the inventors have found that the Examples 1-14 of Tables 1 and 2 provide successful solutions without the need to increase thickness, weight, strength, or the number of cooling features. The Example 1-14 of Tables 1-2 illustrate that designs having an Expression (EQ) value from -4549.984 to -27.214 (i.e., $-4549.984 \leq EQ \leq -27.214$) achieve increased durability without penalties to size, weight, strength, or stress through the use of additional cooling features.

**[0076]** In other words, rather than making areas of the airfoil thicker, or using heavier, stronger materials, or adding additional cooling features, effective particulate accumulation and stress reduction can be achieved by the Examples 1-14 of Tables 1 and 2.

**[0077]** To the extent one or more structures provided herein can be known in the art, it should be appreciated that the present disclosure can include combinations of structures not previously known to combine, at least for reasons based in part on conflicting benefits versus losses, desired modes of operation, or other forms of teaching away in the art.

**[0078]** This written description uses examples to disclose the present disclosure, including the best mode, and also to enable any person skilled in the art to practice the disclosure, including making and using any devices or systems and performing any incorporated methods. The patentable scope of the disclosure is defined by the claims, and can include other examples that occur to those skilled in the art. Such other examples are intended to be within the scope of the claims if they include structural elements that do not differ from the literal language of the claims, or if they include equivalent structural elements with insubstantial differences from the literal languages of the claims.

**[0079]** Further aspects of the disclosure are provided by the subject matter of the following clauses:

Example 1 includes a gas turbine engine, comprising an engine core configured to generate a redline exhaust gas temperature (EGT) in a range of 988 degrees Celsius (°C) to 1120°C, the engine core extending along an engine centerline and including a compressor section, a combustor, and a turbine section, the turbine section including a blade assembly rotatable about the engine centerline, the blade assembly including a platform having a first surface and a second surface, the platform having a stator rotor seal with a upper edge having a radius of curvature defined as a stator rotor seal radius (SRSR), wherein the SRSR is 0.224 to 0.239 meters, an airfoil extending radially outward from the first surface, the airfoil having an outer wall defining an exterior surface, the exterior surface defining a pressure side and a suction side, the outer wall extending between a leading-edge and a trailing-edge, and also extending between a root and a tip, a shank extending radially inward from the second surface to a base, the shank including a set of inlet passages, the base defining a first plane, a feed conduit comprising a passage extending from a feed inlet to a feed outlet, the feed inlet and the feed outlet defining a feed centerline therebetween, the feed inlet fluidly coupled to the set of the inlet passages, the feed inlet having a geometric center defining a second plane parallel to the first plane and intersecting the feed centerline to define a feed angle (Θ) measured between the feed centerline and the second plane, the feed angle (Θ) ranging from example 0 includes 01° to 78°, the feed conduit defines a minimum feed cross-sectional area (FA) determined from a cross-section of the passage perpendicular to the feed centerline, the minimum feed cross-sectional area (A) ranging from 0.00000200 meters square (m2) to 0.00000350 m2, a platform plenum fluidly coupled to the feed outlet, and wherein

$$-4549.984 \leq \left(\left(\frac{FA(m^2)}{0.01m^2}\right)^{-1}\left(\frac{\theta(°)-79°}{79°}\right)\right) \times \left(1.7 \times \frac{EGT(°C)}{500°C} \times \frac{SRSR(m)}{1m}\right) \leq -27.214 .$$

Example 2 includes the gas turbine engine of any preceding example, wherein the set of the inlet passages includes a leading-edge inlet passage, a middle inlet passage, and a trailing-edge inlet passage.

Example 3 includes the gas turbine engine of any preceding example, wherein the feed inlet is fluidly coupled to the middle inlet passage.

Example 4 includes the gas turbine engine of any preceding example, wherein the shank includes a shank leading-edge and a shank trailing-edge, the middle inlet passage is mid-way between the shank leading-edge and the shank trailing-edge.

Example 5 includes the gas turbine engine of any preceding example, wherein the geometric center is a first geometric center and the feed centerline extends through the first geometric center and a second geometric enter of the feed outlet.

Example 6 includes the gas turbine engine of any preceding example, wherein the shank has a dovetail.

Example 7 includes the gas turbine engine of any preceding example, wherein the platform plenum is fluidly coupled to a cooling conduit of the airfoil.

Example 8 includes the gas turbine engine of any preceding example, wherein the blade assembly includes a plurality of cooling holes, the cooling holes coupling the cooling conduit and an exterior of the blade assembly.

Example 9 includes a blade assembly for a gas turbine engine having an engine core configured to generate a redline exhaust gas temperature (EGT) in a range of 988 degrees Celsius (°C) to 1120°C, the blade assembly to be connected to the engine core and rotatable about an engine centerline of the engine core, the blade assembly comprising a platform having a first surface and a second surface, the platform having a stator rotor seal with an upper edge having a radius of curvature defined as a stator rotor seal radius (SRSR), wherein the stator rotor seal radius (SRSR) is 0.224 to 0.239 meters, an airfoil extending radially outward from the first surface, the airfoil having an outer wall defining an exterior surface, the exterior surface defining a pressure side and a suction side, the outer wall extending between a leading-edge and a trailing-edge, and also extending between a root and a tip, a shank extending radially inward from the second surface to a base, the shank including a set of inlet passages, the base defining a first plane, a feed conduit comprising a passage extending from a feed inlet to a feed outlet, the feed inlet and the feed outlet defining a feed centerline therebetween, the feed inlet fluidly coupled to the set of the inlet passages, the feed inlet having a geometric center defining a second plane parallel to the first plane and intersecting the feed centerline to define a feed angle (Θ) measured between the feed centerline and the second plane, the feed angle (Θ) ranging from example 0 includes 01° to 78°, the feed conduit defines a minimum feed cross-sectional area (FA) determined from a cross-section of the passage perpendicular to the feed centerline, the minimum feed cross-sectional area (A) ranging from 0.00000200 meters square (m2) to 0.00000350 m2, a platform plenum fluidly coupled to the feed outlet, and wherein

$$-4549.984 \leq \left(\left(\frac{FA(m^2)}{0.01m^2}\right)^{-1}\left(\frac{\theta(°)-79°}{79°}\right)\right) \times \left(1.7 \times \frac{EGT(℃)}{500℃} \times \frac{SRSR(m)}{1m}\right) \leq -27.214$$

Example 10 includes the blade assembly of any preceding example, wherein the set of the inlet passages includes a leading-edge inlet passage, a middle inlet passage, and a trailing-edge inlet passage.

Example 11 includes the blade assembly of any preceding example, wherein the feed inlet is fluidly coupled to the middle inlet passage.

Example 12 includes the blade assembly of any preceding example, wherein the shank includes a shank leading-edge and a shank trailing-edge, the middle inlet passage is mid-way between the shank leading-edge and the shank trailing-edge.

Example 13 includes the blade assembly of any preceding example, wherein the geometric center is a first geometric center and the feed centerline extends through the first geometric center and a second geometric enter of the feed outlet.

Example 14 includes the blade assembly of any preceding example, wherein the shank has a dovetail.

Example 15 includes the blade assembly of any preceding example, wherein the platform plenum is fluidly coupled to a cooling conduit of the airfoil.

Example 16 includes the blade assembly of any preceding example, wherein the blade assembly includes a plurality of cooling holes, the cooling holes coupling the cooling conduit and an exterior of the blade assembly.

## Claims

**1.** A blade assembly for a gas turbine engine having an engine core configured to generate a redline exhaust gas temperature (EGT) in a range of 988 degrees Celsius (°C) to 1120°C, the blade assembly to be connected to the engine core and rotatable about an engine centerline of the engine core, the blade assembly comprising:

a platform having a first surface and a second surface, the platform having a stator rotor seal with an upper edge having a radius of curvature defined as a stator rotor seal radius (SRSR), wherein the stator rotor seal radius (SRSR) is 0.224 to 0.239 meters;

an airfoil extending radially outward from the first surface, the airfoil having an outer wall defining an exterior surface, the exterior surface defining a pressure side and a suction side, the outer wall extending between a leading-edge and a trailing-edge, and also extending between a root and a tip;

a shank extending radially inward from the second surface to a base, the shank including a set of inlet passages, the base defining a first plane;

a feed conduit comprising a passage extending from a feed inlet to a feed outlet, the feed inlet and the feed outlet defining a feed centerline therebetween, the feed inlet fluidly coupled to the set of the inlet passages, the feed inlet having a geometric center defining a second plane parallel to the first plane and intersecting the feed centerline to define a feed angle (Θ) measured between the feed centerline and the second plane, the feed angle (Θ) ranging from 0.01° to 78°, the feed conduit defines a minimum feed cross-sectional area (*FA*) determined from a cross-section of the passage perpendicular to the feed centerline, the minimum feed cross-sectional area (A) ranging from 0.00000200 meters square ($m^2$) to 0.00000350 $m^2$;

a platform plenum fluidly coupled to the feed outlet; and

wherein:

$$-4549.984 \leq \left(\left(\frac{FA(m^2)}{0.01m^2}\right)^{-1}\left(\frac{\theta(°)-79°}{79°}\right)\right) \times \left(1.7 \times \frac{EGT(℃)}{500℃} \times \frac{SRSR(m)}{1m}\right) \leq -27.214$$

**2.** The blade assembly of claim 1, wherein the set of the inlet passages includes a leading-edge inlet passage, a middle inlet passage, and a trailing-edge inlet passage.

**3.** The blade assembly of claim 2, wherein the feed inlet is fluidly coupled to the middle inlet passage.

**4.** The blade assembly of claim 2 or 3, wherein the shank includes a shank leading-edge and a shank trailing-edge, the middle inlet passage is mid-way between the shank leading-edge and the shank trailing-edge.

**5.** The blade assembly of any preceding claim, wherein the geometric center is a first geometric center and the feed centerline extends through the first geometric center and a second geometric enter of the feed outlet.

**6.** The blade assembly of any preceding claim, wherein the platform plenum is fluidly coupled to a cooling conduit of the airfoil.

**7.** The blade assembly of claim 6, wherein the blade assembly includes a plurality of cooling holes, the cooling holes coupling the cooling conduit and an exterior of the blade assembly.

**8.** The blade assembly of any preceding claim, wherein the shank is dovetail.

**9.** A gas turbine engine including:
the engine core, the engine core including:

a compressor section;
a combustor; and
a turbine section, the turbine section including the blade assembly of any of claims 1-8.

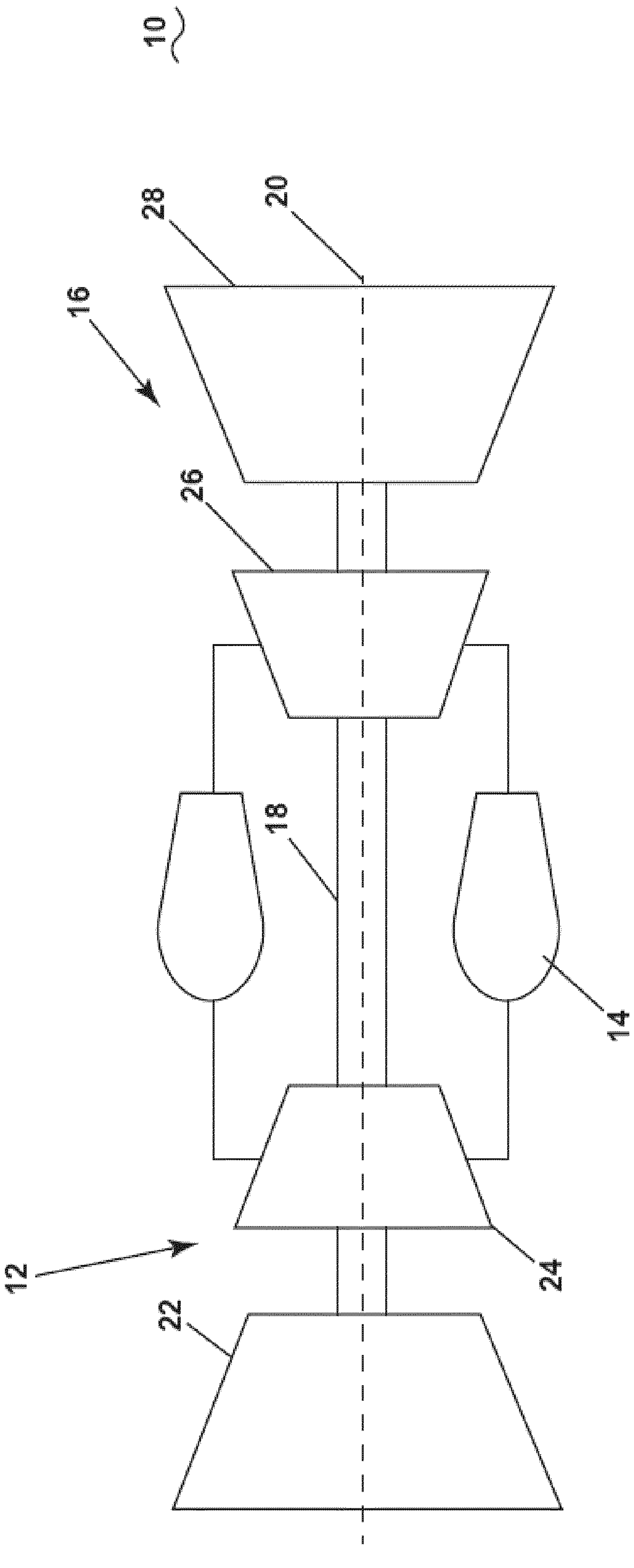


FIG. 1

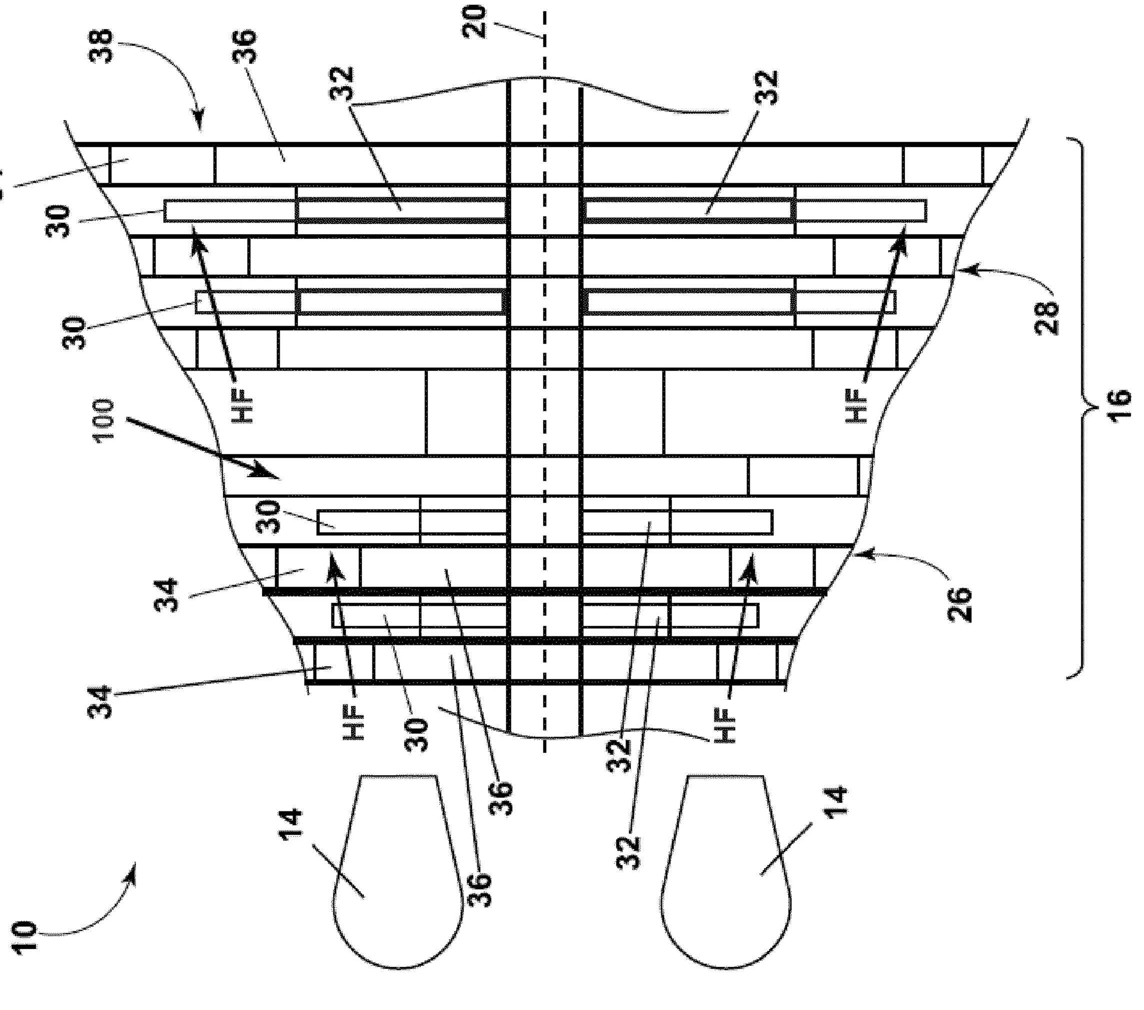
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FIG. 2

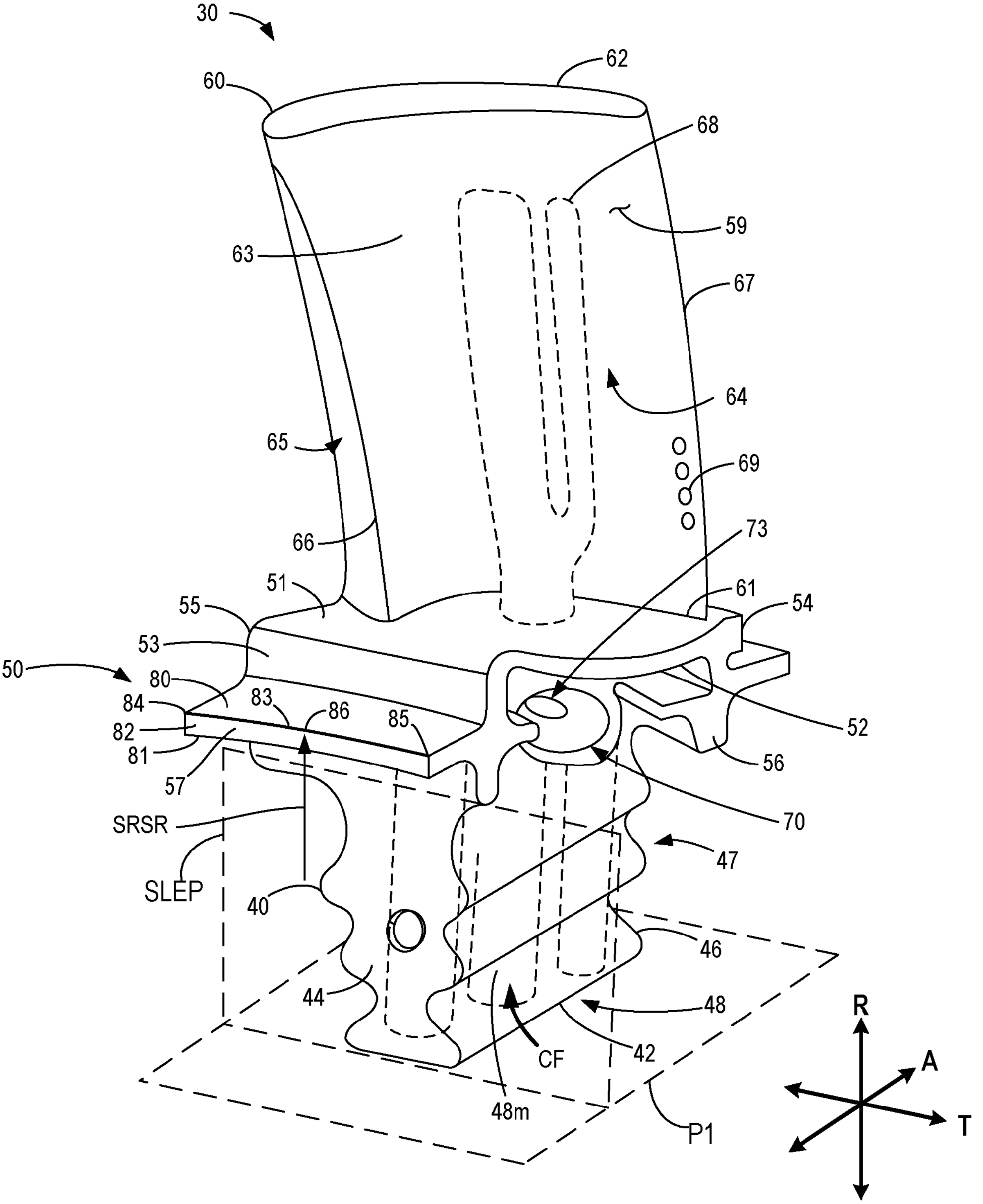
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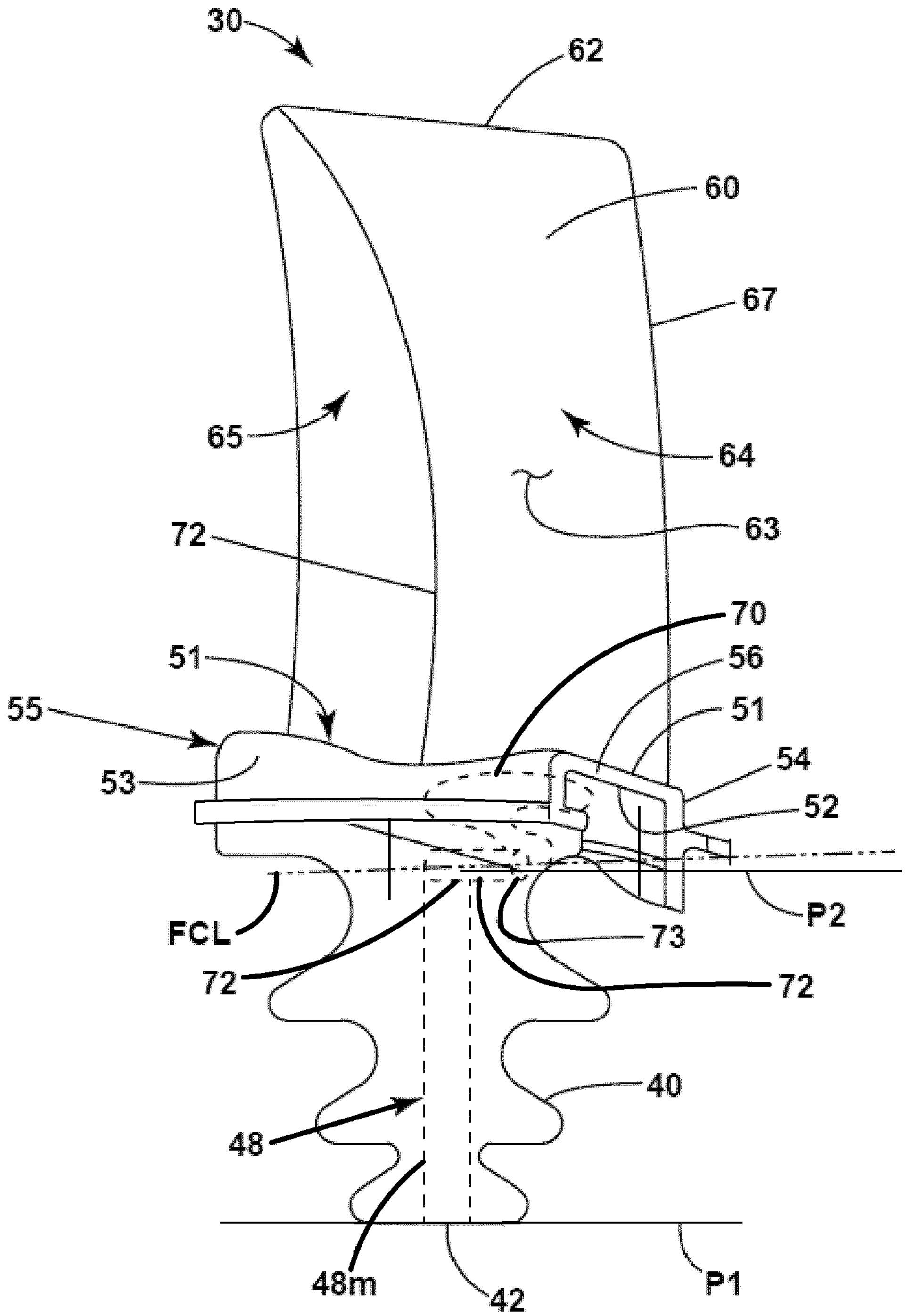
FIG. 3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P1

FIG. 4

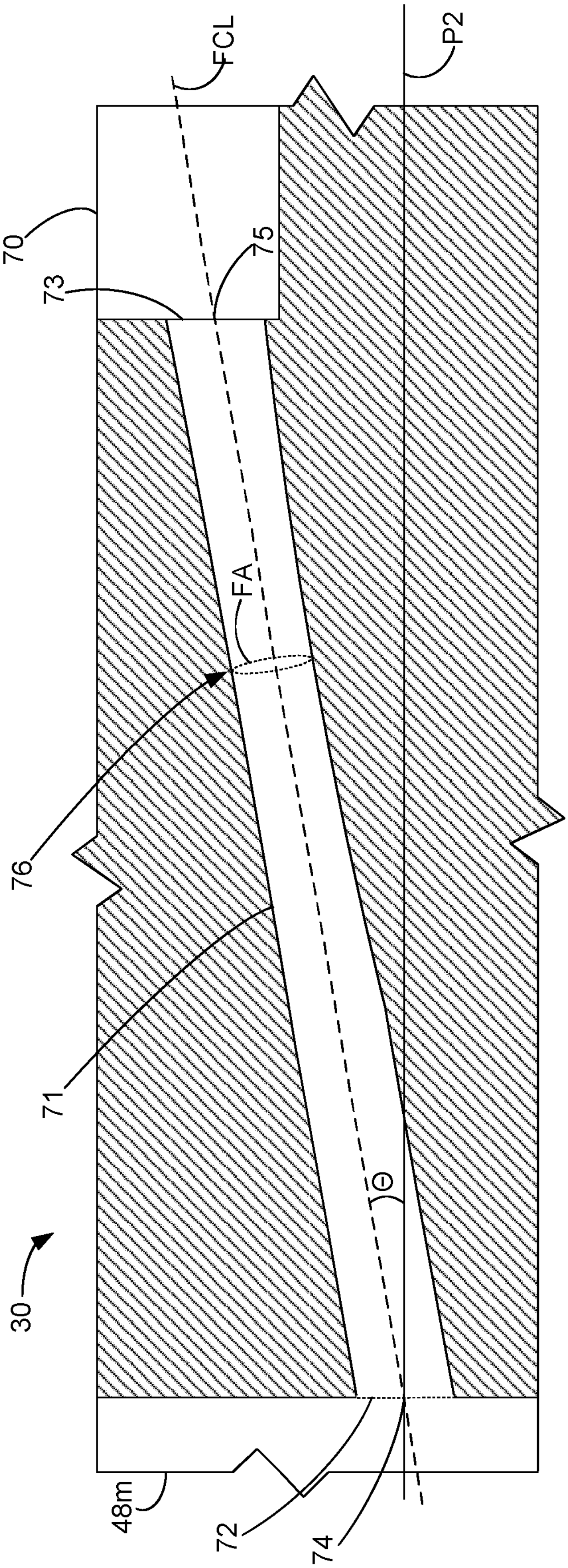


FIG. 5

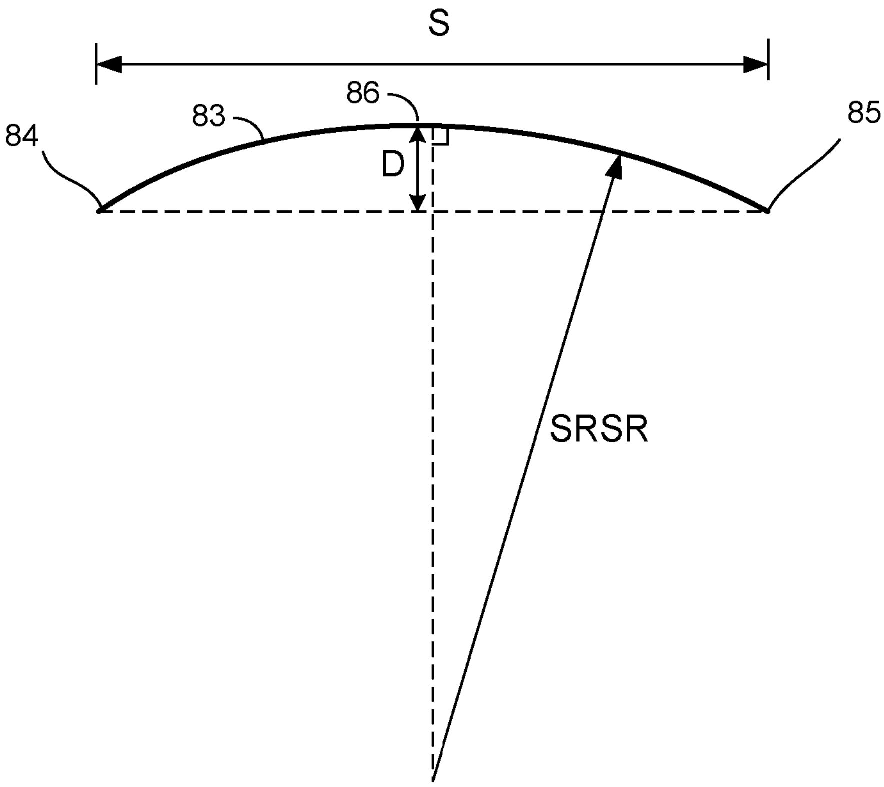


FIG. 6

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# EUROPEAN SEARCH REPORT

Application Number
EP 24 21 1811

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| Category | Citation of document with indication, where appropriate, of relevant passages | Relevant to claim | CLASSIFICATION OF THE APPLICATION (IPC) |
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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 | 5 March 2025 | Rau, Guido |

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