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(54) **Shrouded turbine blade with contoured platform and axial dovetail**

(57) A turbine blade includes an airfoil having a root, a tip, a concave pressure side, and a laterally opposite convex suction side, the pressure and suction sides extending in chord between opposite leading and trailing edges; an outer platform disposed at the tip of the airfoil, the outer platform having spaced-apart lateral edges which each defme an interlocking element; an inner platform with two spaced-apart curved lateral edges disposed at the root of the airfoil, the inner platform having a hot side facing the airfoil which is contoured in a non-axisymmetric shape; and a dovetail extending radially inward from the opposite side of the inner platform, wherein the dovetail is axially straight.

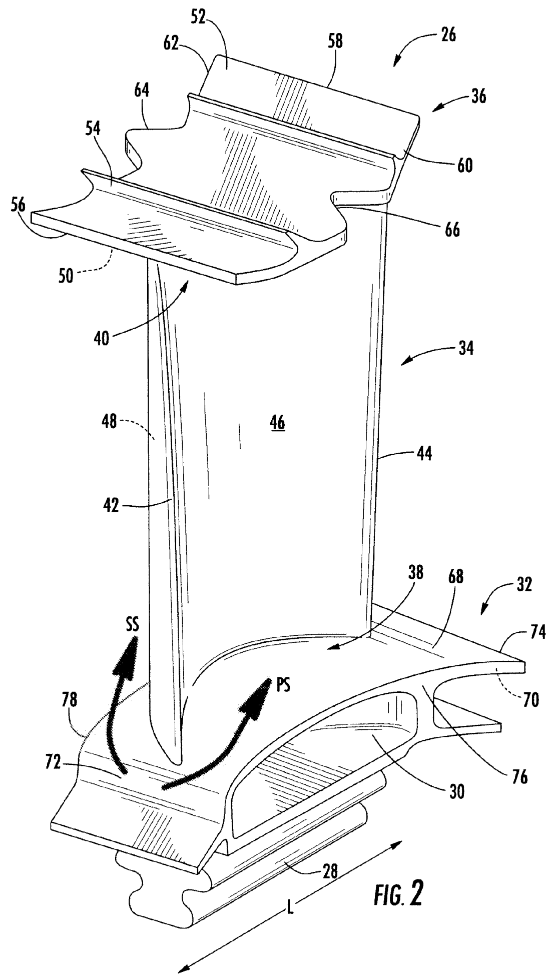

FIG. 2

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Description

[0001] The present invention relates generally to gas turbine engines, and more specifically, to turbines therein.

[0002] In a gas turbine engine, air is pressurized in a compressor and subsequently mixed with fuel and burned in a combustor to generate combustion gases. One or more turbines downstream of the combustor extract energy from the combustion gases to drive the compressor, as well as a fan, shaft, propeller, or other mechanical load. Each turbine comprises one or more rotors each comprising a disk carrying an array of turbine blades or buckets. A stationary nozzle comprising an array of stator vanes having radially outer and inner endwalls in the form of annular bands is disposed upstream of each rotor, and serves to optimally direct the flow of combustion gases into the rotor. Collectively each nozzle and the downstream rotor is referred to as a "stage" of the turbine.

[0003] The complex three-dimensional (3D) configuration of the vane and blade airfoils is tailored for maximizing efficiency of operation, and varies radially in span along the airfoils as well as axially along the chords of the airfoils between the leading and trailing edges. Accordingly, the velocity and pressure distributions of the combustion gases over the airfoil surfaces as well as within the corresponding flow passages also vary.

[0004] Undesirable pressure losses in the combustion gas flowpaths therefore correspond with undesirable reduction in overall turbine efficiency. For example, the combustion gases enter the corresponding rows of vanes and blades in the flow passages therebetween and are necessarily split at the respective leading edges of the airfoils.

[0005] The locus of stagnation points of the incident combustion gases extends along the leading edge of each airfoil. Corresponding boundary layers are formed along the pressure and suction sides of each airfoil, as well as along each radially outer and inner endwall which collectively bound the four sides of each flow passage. In the boundary layers, the local velocity of the combustion gases varies from zero along the endwalls and airfoil surfaces to the unrestrained velocity in the combustion gases where the boundary layers terminate.

[0006] One common source of turbine pressure losses is the formation of horseshoe and passage vortices generated as the combustion gases are split in their travel around the airfoil leading edges. A total pressure gradient is effected in the boundary layer flow at the junction of the leading edge and endwalls of the airfoil. This pressure gradient at the airfoil leading edges forms a pair of counterrotating horseshoe vortices which travel downstream on the opposite sides of each airfoil near the endwall. Turning of the horseshoe vortices introduces streamwise vorticity and thus builds up a passage vortex as well.

[0007] The vortices travel aft along the opposite pressure and suction sides of each airfoil and behave differently due to the different pressure and velocity distributions therealong. For example, computational analysis indicates that the suction side vortex migrates away from the endwall toward the airfoil trailing edge and then interacts following the airfoil trailing edge with the pressure side vortex flowing aft thereto.

[0008] The interaction of the pressure and suction side vortices occurs near the mid-span region of the airfoils and creates total pressure loss and a corresponding reduction in turbine efficiency. These vortices also create turbulence and increase undesirable heating of the endwalls.

[0009] Since the horseshoe and passage vortices are formed at the junctions of turbine rotor blades and their integral root platforms, as well at the junctions of nozzle stator vanes and their outer and inner bands, corresponding losses in turbine efficiency are created, as well as additional heating of the corresponding endwall components.

[0010] Non-axisymmetric end-wall-contouring (EWC) may be used on turbine airfoils to reduce vortex effects and thereby provide a significant performance improvement. One known design includes a leading edge "bump", a suction side "trough" and a trailing edge "ridge". Typically, the blade dovetail and the edge of the platform are straight. With this straight dovetail/platform design, the trailing edge ridge will cross over from one platform to the adjacent platform. Because of the manufacturing and assembly tolerances, the trailing edge ridge will be interrupted and may see a forward facing step that adversely affects performance. It is desirable to have an improved platform design that can keep the advantage of the EWC without the penalty from the TE ridge interruption. A circular-arc platform may be used to allow the trailing edge ridge to locate within a platform without cross over to the adjacent platform. However, this has only been possible with a blade that does not have a tip shroud and can be individually assembled into a rotor disk. For a shrouded turbine blade, the interlocking tip shroud, the curved platform and a conventional curved dovetail make rotor assembly impossible.

[0011] Accordingly, it is desirable to minimize vortex effects in a shrouded turbine blade while still permitting simple assembly.

BRIEF SUMMARY OF THE INVENTION

[0012] The above-mentioned need is met by the present invention, which provides a shrouded turbine blade having a 3D-countoured inner band surface and an axially straight dovetail.

[0013] According to one aspect of the invention, a turbine blade includes an airfoil having a root, a tip, a concave pressure side, and a laterally opposite convex suction side, the pressure and suction sides extending in chord between opposite leading and trailing edges; an outer platform disposed at the tip of the airfoil, the outer platform having spaced-apart lateral edges which each define an interlocking element; an inner platform with two spaced-apart curved lateral edges disposed at the root of the airfoil, the inner platform having a hot side facing the airfoil which is contoured in a non-axisymmetric shape; and a dovetail extending radially inward from the opposite side of the inner platform, wherein the dovetail is axially straight.

[0014] According to another aspect of the invention, a turbine blade assembly includes a plurality of blades, each blade having: an airfoil having a root, a tip, a concave pressure side, and a laterally opposite convex suction side, the pressure and suction sides extending in chord between opposite leading and trailing edges; and an outer platform disposed at the tip of the airfoil, the outer platform having spaced-apart lateral edges which are configured with an interlocking element; an inner platform with two spaced-apart curved lateral edges disposed at the root of the airfoil, the inner platform having a hot side facing the airfoil which is contoured in a non-axisymmetric shape. The blades are disposed in an annular side-by-side array such so as to define a plurality of flow passages each of which is bounded between two of the inner platforms, two of the outer platforms, and adjacent first and second airfoils. The interlocking elements of adjacent outer platforms are engaged with each other;. The hot sides of the inner platforms in each of the passages are contoured in a non-axisymmetric shape including a peak of relatively higher radial height adjoining the pressure side of the first airfoil adjacent its leading edge, and a trough of relatively lower radial height is disposed parallel to and spaced-away from the suction side of the second airfoil aft of its leading edge; and the peak and trough define cooperatively define an arcuate channel extending axially along the inner platform between the first and second airfoils.

BRIEF DESCRIPTION OF THE DRAWINGS

[0015] The invention may be best understood by reference to the following description taken in conjunction with the accompanying drawing figures in which:

Figure 1 is a schematic view of a gas turbine engine incorporating a turbine nozzle constructed according to an aspect of the present invention;

Figure 2 is a left-side perspective view of a turbine blade of the engine shown in

Figure 1;

Figure 3 is an enlarged view of a portion of the turbine blade shown in Figure 2;

Figure 4 is a cross-sectional view of several turbine blades assembled side-by-side;

Figure 5 is a schematic view taken along lines 5-5 of Figure 4;

Figure 6 is a view taken along lines 6-6 of Figure 4; and

Figure 7 is an exploded perspective view of a portion of a turbine disk along with several turbine blades;

DETAILED DESCRIPTION OF THE INVENTION

[0016] Referring to the drawings wherein identical reference numerals denote the same elements throughout the various views, Figure 1 depicts schematically the elements of an exemplary gas turbine engine 10 having a fan 12, a high pressure compressor 14, a combustor 16, a high pressure turbine ("HPT") 18, and a low pressure turbine 20, all arranged in a serial, axial flow relationship along a central longitudinal axis "A". Collectively the high pressure compressor 14, the combustor 16, and the high pressure turbine 18 are referred to as a "core". The high pressure compressor 14 provides compressed air that passes into the combustor 12 where fuel is introduced and burned, generating hot combustion gases. The hot combustion gases are discharged to the high pressure turbine 18 where they are expanded to extract energy therefrom. The high pressure turbine 18 drives the compressor 10 through an outer shaft 22. Pressurized air exiting from the high pressure turbine 18 is discharged to the low pressure turbine ("LPT") 20 where it is further expanded to extract energy. The low pressure turbine 20 drives the fan 12 through an inner shaft 24. The fan 12 generates a flow of pressurized air, a portion of which supercharges the inlet of the high pressure compressor 14, and the majority of which bypasses the "core" to provide the majority of the thrust developed by the engine 10.

[0017] While the illustrated engine 10 is a high-bypass turbofan engine, the principles described herein are equally applicable to turboprop, turbojet, and turboshaft engines, as well as turbine engines used for other vehicles or in stationary applications. The principles described herein are also applicable to turbines using working fluids other than air, such as steam turbines. Furthermore, while and LPT blade is used as an example, it will be understood that the principles of the present invention may be applied to any turbine blade having inner and outer shrouds or platforms, including without limitation HPT and intermediate-pressure turbine ("IPT") blades.

[0018] In accordance with conventional practice, the LPT 20 includes a series of stages each having a nozzle comprising an array of stationary airfoil-shaped vanes and a downstream rotating disk carrying an array of turbine blades. Figures 2 and 3 illustrate the construction of the turbine blades, labeled 26, in more detail. The blade 26 is a unitary component including a dovetail 28, a shank 30, an inner platform 32, an airfoil 34, and an outer platform 36. The airfoil includes a root 38, a tip 40, a leading edge 42, trailing edge 44, and a concave pressure side 46 opposed to a convex suction side 48. The inner and outer platforms 32 and 36 define the inner and outer radial boundaries, respectively, of the gas flow past the airfoil 34.

[0019] The dovetail 28 has a cross-sectional profile having lands and grooves constructed in accordance with conventional practice. The dovetail 28 is axially aligned relative to the engine centerline and its shape is "axially straight". In other words, its shape is equivalent to that generated by translating the dovetail profile along a line "L" parallel to a longitudinal centerline axis of the engine, and is not curved or cambered.

[0020] The outer platform 36 has a "hot side" 50 facing the hot gas flowpath and a "cold side" 52 facing away from the hot gas flowpath. One or more annular seal teeth 54 extend radially outwards from the cold side 52 of the outer platform. The outer platform 36 is bounded by opposed leading and trailing edges 56 and 58, and by lateral edges 60 and 62 that extend between the leading and trailing edges 56 and 58. the lateral edges 60 and 62 of the outer platform 36 have a shape which is nonlinear. Each lateral edge 60 and 62 incorporates an interlocking element, so as to provide an interlocking function in the axial direction when two outer platforms 36 are assembled together. In the illustrated example, the lateral edges 60 and 62 have identical shapes in plan view, with the result that the right-side lateral edge 62 (as viewed aft looking forward) effectively defines a laterally-extending tab 64 while the left-side lateral edge 60 defines a complementary recess 66.

[0021] The inner platform also has a "hot side" 68 facing the hot gas flowpath and a "cold side" 70 facing away from the hot gas flowpath. The inner platform 32 is bounded by opposed leading and trailing edges 72 and 74, and by lateral edges 76 and 78 that extend between the leading and trailing edges 72 and 74. The lateral edges 76 and 78 of the inner platform 32 are curved (the arc may be circular or some other shape depending upon the specific application). In the illustrated example, the lateral edges 76 and 78 have identical shapes in plan view. As a result, one lateral side of the inner platform 32 is convex in plan view, and the other lateral side is concave in plan view. These curvatures correspond to the direction that the airfoil 34 is cambered. As will be described in more detail below, the arcuate shape of the lateral edges 76 and 78 permit 3D contouring features of the inner platform 32 to be implemented without the need to cross over to the inner platform 32 of an adjacent turbine blade 26.

[0022] In operation, the gas pressure gradient at the airfoil leading edges causes the formation of a pair of counterrotating horseshoe vortices which travel downstream on the opposite sides of each airfoil 34 near the inner platform 32. The direction of travel of pressure side and suction side vortices are shown schematically in Figure 2, labeled PS and SS, respectively. Turning of the horseshoe vortices will introduce streamwise vorticity and thus build up a passage vortex, the low momentum fluid in the endwall layer being driven by a transverse pressure gradient to cross the passage between airfoils 34 from pressure to suction side

[0023] As shown in Figures 3-6, the hot side 68 of the inner platform 32 is preferentially contoured in elevation relative to a conventional axisymmetric or circular circumferential profile in order to reduce the adverse effects of the passage and horseshoe vortices. In particular the inner platform contour is non-axisymmetric, but is instead contoured in radial elevation from a wide peak 80 adjacent the pressure side 46 of each blade 26 to a depressed narrow trough 82. This contouring is referred to generally as "3D-contouring". It will be understood that the complete shape defining the aerodynamic "endwall" of the passage between two adjacent airfoils 34 of the assembled rotor is defined cooperatively by portions of the side-by-side inner platforms 32 of the airfoils 34.

[0024] A typical prior art inner band generally has a surface profile which is convexly-curved in a shape similar to the top surface of an airfoil when viewed in longitudinal cross-section (see Figure 5). This profile is a symmetrical surface of revolution about the longitudinal axis of the engine 10. This profile is considered a baseline reference, and is illustrated with a dashed line denoted "B". The 3D-contoured surface profile is shown with a solid line. Points having the same height or radial dimension are interconnected by contour lines in the figures. As seen in Figure 4, each of the airfoils 34 has a chord length "C" measured from its leading edge 42 to its trailing edge 44, and a direction parallel to this dimension denotes a "chordwise" direction. A direction parallel to the leading or trailing edges 72 or 74 of the inner platform 32 is referred to as a tangential direction as illustrated by the arrow marked "T" in Figure 4. As used herein, it will be understood that the terms "positive elevation", "peak" and similar terms refer to surface characteristics located radially outboard or having a greater radius measured from the longitudinal axis A than the local baseline B, and the terms "trough", "negative elevation", and similar terms refer to surface characteristics located radially inboard or having a smaller radius measured from the longitudinal axis A than the local baseline B.

[0025] Referring to Figures 4 and 5, The trough 82 is present in the hot side 68 of the inner platform 32 between each pair of airfoils 34, extending generally from the leading edge 42 to the trailing edge 44 of the airfoil 34. The deepest portion of the trough 82 runs along a line substantially parallel to the suction side 48 of the airfoil 34, coincident with the line 6-6 marked in Figure 4. In the particular example illustrated, the deepest portion of the trough 82 is lower than the baseline profile B by approximately 20% of the total difference in radial height between the lowest and highest locations of the hot side 68, or about 2 units, where the total height difference is about 8.5 units. In the tangential direction, measuring from the suction side 48 of an airfoil 34, the line representing the deepest portion of the trough 82 is positioned about 10% of the distance to the pressure side 46 of the adjacent airfoil 34. In the chordwise direction, the deepest portion of the trough 82 occurs at approximately the location of the maximum section thickness of the airfoil 34 (commonly referred to as a "high-C" location).

[0026] As seen in Figures 4 and 5, the peak 80 runs along a line substantially parallel to the pressure side 46 of the adjacent airfoil 34. A ridge 81 extends from the highest portion of the peak 80 and extends in a generally tangential direction away from the pressure side 46 of the adjacent airfoil 34. The radial height of the peak 80 slopes away from this ridge 81 towards both the leading edge 42 and the trailing edge 44 of the airfoil 34. The peak 80 increases in elevation behind the leading edge 42 from the baseline elevation B to the maximum elevation with a large gradient over the first third of the chord length from the leading edge 42, whereas the peak 80 increases in elevation from the trailing edge 44 over the same magnitude over the remaining two-thirds of the chord length from the trailing edge 44 at a substantially shallower gradient or slope.

[0027] In the particular example illustrated, the highest portion of the peak 80 is higher than the baseline profile B by approximately 80% of the total difference in radial height between the lowest and highest locations of the hot side 68, or about 7 units, where the total height difference is about 8.5 units. In the chordwise direction, the highest portion of the peak 80 is located between the mid-chord position and the leading edge 42 of the adjacent airfoil 34.

[0028] A trailing edge ridge 84 is present in the hot side 68 of the inner platform 32 aft of the airfoil 34 (See Figures 3 and 4). It runs aft from the trailing edge 44 of the airfoil 34, along a line which is substantially an extension of the chord line of the airfoil 34. The radial height of the trailing edge ridge 84 slopes away from this line towards both the leading edge 42 and the trailing edge 44 of the airfoil 34. In the particular example illustrated, the highest portion of the trailing edge ridge 84 is higher than the baseline profile B by approximately 60% of the total difference in radial height between the lowest and highest locations of the hot side 68, or about 5 units, where the total height difference is about 8.5 units. The highest portion of the trailing edge ridge 84 is located immediately adjacent the trailing edge 44 of the airfoil 34 at its root 38.

[0029] It is noted that the specific numerical values described above are merely examples and that they may be varied to provide optimum performance for a specific application. For example, the radial heights noted above could easily be varied by plus or minus 20%, and the tangential locations could be varied by plus or minus 15%.

[0030] Whereas the peak 80 is locally isolated near its maximum height, the trough 82 has a generally uniform and shallow depth over substantially its entire longitudinal or axial length. Collectively, the elevated peak 80, depressed trough 82, and trailing edge ridge 84 provide an aerodynamically smooth chute or curved flute that follows the arcuate contour of the flowpath between the concave pressure side 46 of one airfoil 34 and the convex suction side 48 of the adjacent airfoil 34 to smoothly channel the combustion gases therethrough. In particular the peak 80 and trough 82 cooperating together conform with the incidence angle of the combustion gases for smoothly banking or turning the combustion gases for reducing the adverse effect of the horseshoe and passage vortices. The circular-arc inner platform 32 allows the trailing edge ridge 84 to locate within an inner platform 32 without crossing over to the adjacent inner platform 32. Consequently, the endwall boundary layer flows along the trailing edge ridge 84 will not "see" a radial discontinuity or "step". In particular there is no forward facing step. This feature helps maintain the aerodynamic performance improvement of the 3D contouring.

[0031] In the example shown here, there is not a significant fillet or other similar structure present on the hot side 68 of the inner platform 32 between the trailing edge 44 of the airfoil 34 and the trailing edge ridge 84. In other words, there is a sharply defined intersection present between the trailing edge 44 of the airfoil 34 at it root 38 and the trailing edge ridge 84. For mechanical strength, it may be necessary to include some type of fillet at this location. For aerodynamic purposes any fillet present should be minimized.

[0032] Computer analysis of the airfoil and inner platform configuration described above predicts significant reduction in aerodynamic pressure losses near the inner platform hot side 68 during engine operation. The improved pressure distribution extends from the hot side 68 over a substantial portion of the lower span of the airfoil 34 to significantly reduce vortex strength and cross-passage pressure gradients that drive the horseshoe vortices toward the airfoil suction sides 48. The 3D contoured hot side 68 also decreases vortex migration toward the mid-span of the airfoils 34 while reducing total pressure loss. These benefits increase performance and efficiency of the LPT 20 and engine 10.

[0033] Referring to Figure 7, the blades 26 may be mounted to a turbine disk 86 as follows. First, a set of the blades 86 are assembled into a complete 360 degree array. A holding fixture or jig (not shown) may be used to clamp the blades 26 in position. Thus assembled, the lateral edges 76 and 78 of the inner platforms 32 are touching or closely adjacent, and the lateral edges 60 and 62 of the outer platforms 36 are touching or closely adjacent. The tab 64 of each outer platform 36 is received in the recess 66 of the adjacent outer platform 36. This effectively interlocks the outer platforms 36 to as to resist axial movement of the outer platforms 36. Next, the array of blades 26 can then be slid into the dovetail slots 88 of the disk 86 (only a portion of which is shown in Figure 7). The blades 26 may be then be retained to the disk 86 in the axial direction using known components such as bolted retainers, disk plates, annular seals, or the like (not shown). When assembled, a flow passage "P" is defined in the spaces between the blades 26. Each flow passage P is bounded by two adjacent inner platforms 32, two adjacent outer platforms 36, and two adjacent airfoils 34.

[0034] The foregoing has described a shrouded turbine blade having a 3D-contoured inner band. While specific embodiments of the present invention have been described, it will be apparent to those skilled in the art that various modifications thereto can be made without departing from the spirit and scope of the invention. Accordingly, the foregoing description of the preferred embodiment of the invention and the best mode for practicing the invention are provided for the purpose of illustration only and not for the purpose of limitation, the invention being defined by the claims.

Claims

1. A turbine blade comprising:

    an airfoil having a root, a tip, a concave pressure side, and a laterally opposite convex suction side, the pressure and suction sides extending in chord between opposite leading and trailing edges;
    an outer platform disposed at the tip of the airfoil, the outer platform having spaced-apart lateral edges, each defining an interlocking element;
    an inner platform with two spaced-apart curved lateral edges disposed at the root of the airfoil, the inner platform having a hot side facing the airfoil which is contoured in a non-axisymmetric shape; and
    a dovetail extending radially inward from the opposite side of the inner platform,
    wherein the dovetail is axially straight.

2. The turbine blade of claim 1 wherein one of the lateral edges of the outer platform defines a protruding tab, and the other lateral edge defines a recess which a shape complementary to the tab.

3. The turbine blade of claim 1 or 2 wherein the hot side of the inner platform is contoured in a non-axisymmetric shape including a peak of relatively higher radial height adjoining the pressure side of the airfoil adjacent its leading edge, and a trough of relatively lower radial height disposed parallel to and spaced-away from the suction side of the airfoil aft of its leading edge.

4. The turbine blade of any of the preceding claims wherein the hot side of the inner platform includes a trailing edge ridge of relatively higher radial height extending aft of the trailing edge of the airfoil.

5. A turbine blade according to any of the preceding claims wherein the peak is centered at the pressure side of the airfoil between the leading edge and a mid-chord position, and decreases in height forward, aft, and laterally therefrom; and the trough is centered at the suction side near the maximum thickness of the airfoil, and decreases in depth forward, aft, and laterally therefrom.

6. A turbine blade according to any of the preceding claims wherein one of the lateral edges of the inner platform is convex and the other lateral edge is concave.

7. A turbine blade assembly comprising:

    a plurality of blades, each blade including:

        an airfoil having a root, a tip, a concave pressure side, and a laterally opposite convex suction side, the pressure and suction sides extending in chord between opposite leading and trailing edges;
        an outer platform disposed at the tip of the airfoil, the outer platform having spaced-apart lateral edges which are configured with an interlocking element;
        an inner platform with two spaced-apart curved lateral edges disposed at the root of the airfoil, the inner platform having a hot side facing the airfoil which is contoured in a non-axisymmetric shape;
        the blades disposed in an annular side-by-side array such so as to define a plurality of flow passages each of which is bounded between two of the inner platforms, two of the outer platforms, and adjacent first and second airfoils;
        wherein the interlocking elements of adjacent outer platforms are engaged with each other;
        wherein the hot sides of the inner platforms in each of the passages are contoured in a non-axisymmetric shape including a peak of relatively higher radial height adjoining the pressure side of the first airfoil adjacent its leading edge, and a trough of relatively lower radial height is disposed parallel to and spaced-away from the suction side of the second airfoil aft of its leading edge; and
        wherein the peak and trough defme cooperatively define an arcuate channel extending axially along the inner platform between the first and second airfoils.

8. A turbine blade assembly according to claim 7 wherein the peak decreases in height around each the leading edge of the first airfoil to join the trough along the suction side of the second airfoil; and the trough extends along the suction sides of the second airfoil to its trailing edge.

9. A turbine blade assembly according to claim 7

wherein a line defining the deepest portion of the trough is positioned about 10% of the tangential distance from the suction side of the second airfoil to the pressure side of the first airfoil.

10. The turbine blade assembly of claim 7 wherein the hot side of each inner platform includes a trailing edge ridge of relatively higher radial height extending aft of the trailing edge of the airfoil.

11. A turbine blade assembly according to claim 7 wherein the peak is centered at the pressure side of each airfoil between the leading edge and a mid-chord position, and decreases in height forward, aft, and laterally therefrom; and the trough is centered at the suction side near the maximum thickness of the airfoil, and decreases in depth forward, aft, and laterally therefrom.

12. A turbine blade assembly according to claim 7 wherein one of the lateral edges of each inner platform is convex and the other lateral edge is concave.

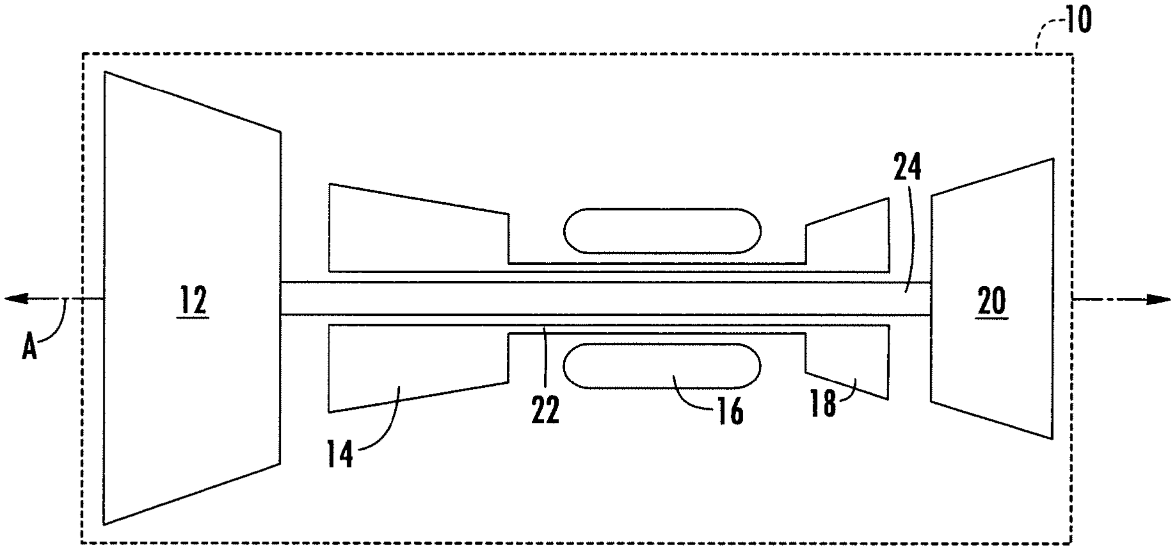

FIG. 1

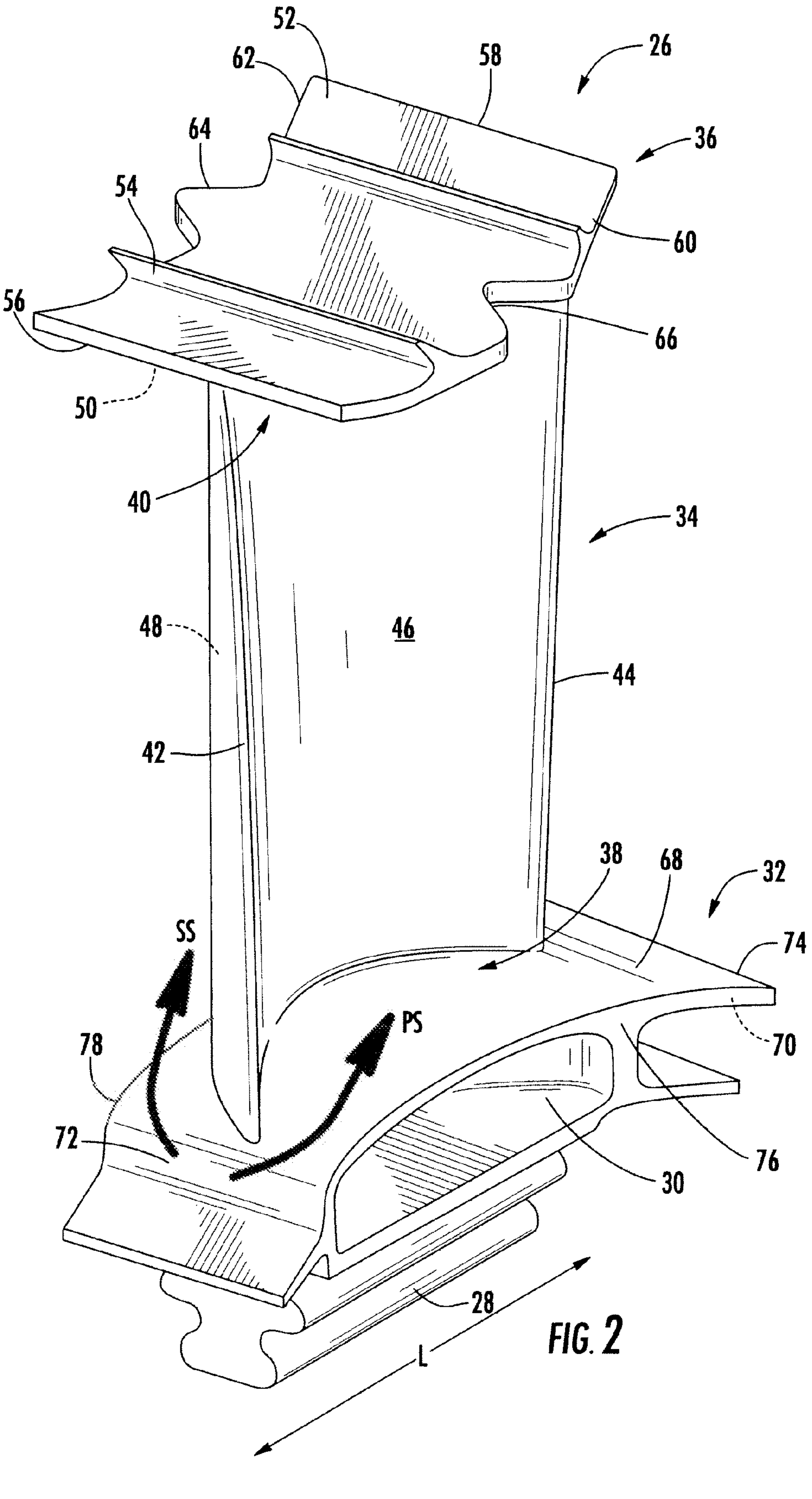
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FIG. 2

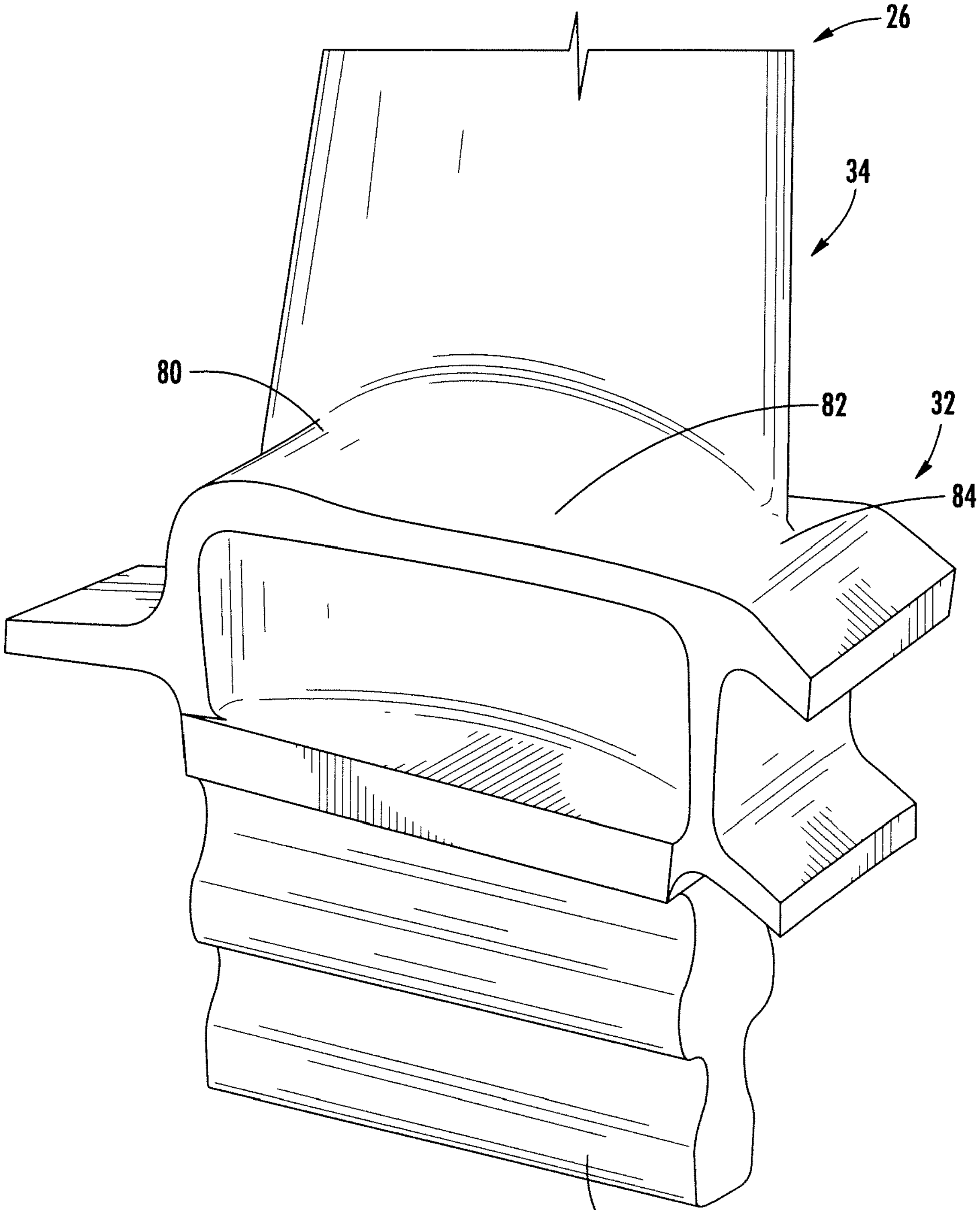

FIG. 3

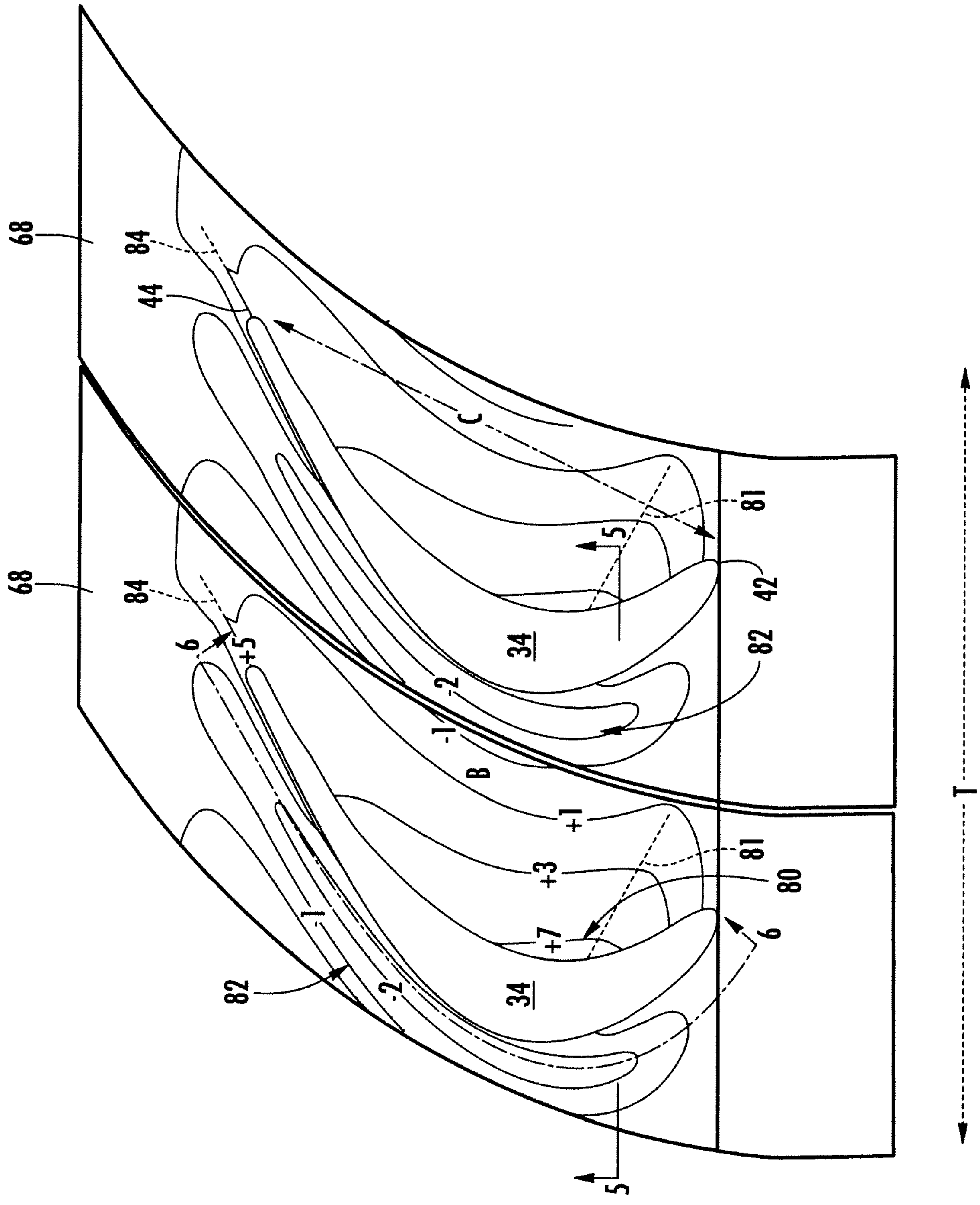

FIG. 4

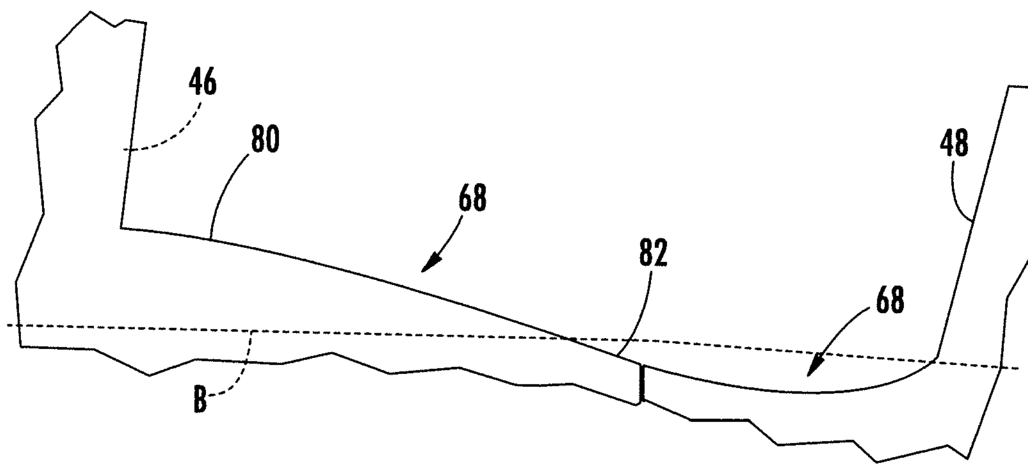

FIG. 5

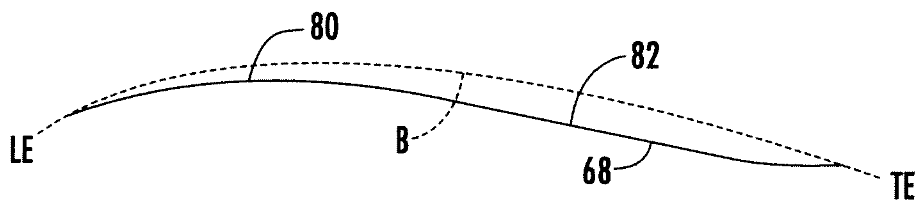

FIG. 6

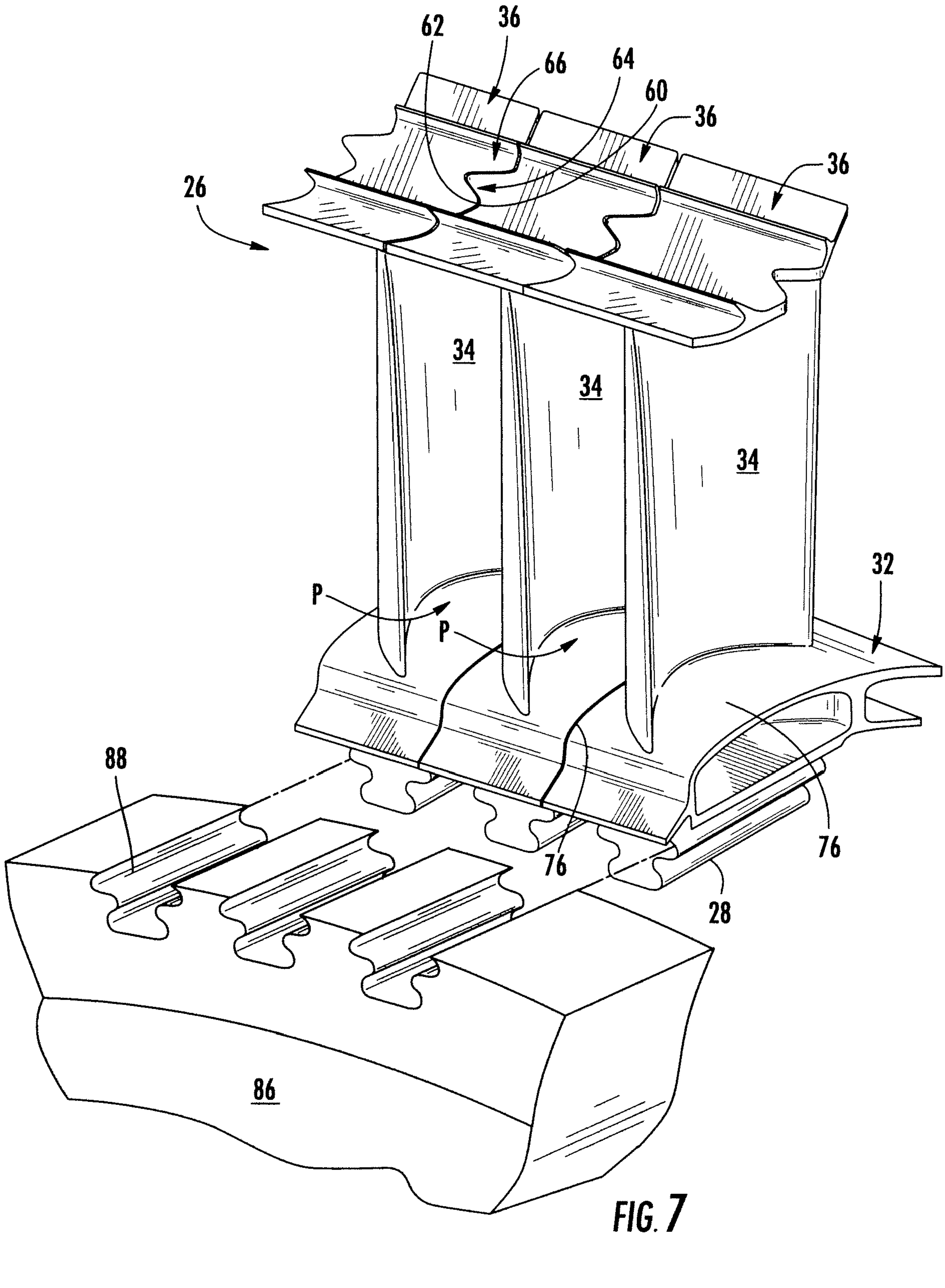

FIG. 7