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(54) **ADDITIVELY MANUFACTURED TURBINE VANE CLUSTER**

(71) Applicant: **RTX Corporation**, Farmington, CT (US)

(72) Inventors: **Lawrence A. Binek**, Glastonbury, CT (US); **Paul M. Lutjen**, Kennebunkport, ME (US); **Jose R. Paulino**, Jupiter, FL (US)

(73) Assignee: **RTX Corporation**, Farmington, CT (US)

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(51) **Int. Cl.**

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B22F 5/04	(2006.01)
B22F 10/28	(2021.01)
B33Y 10/00	(2015.01)
B33Y 80/00	(2015.01)

(52) **U.S. Cl.**
CPC ***F01D 9/041*** (2013.01); ***B22F 5/04*** (2013.01); ***B22F 10/28*** (2021.01); *B33Y 10/00* (2014.12); *B33Y 80/00* (2014.12)

(58) **Field of Classification Search**
CPC F01D 9/041; B33Y 10/00; B33Y 80/00; B22F 5/04; B22F 10/28; Y10T 428/24314; F23R 3/002; F05D 2250/184
See application file for complete search history.

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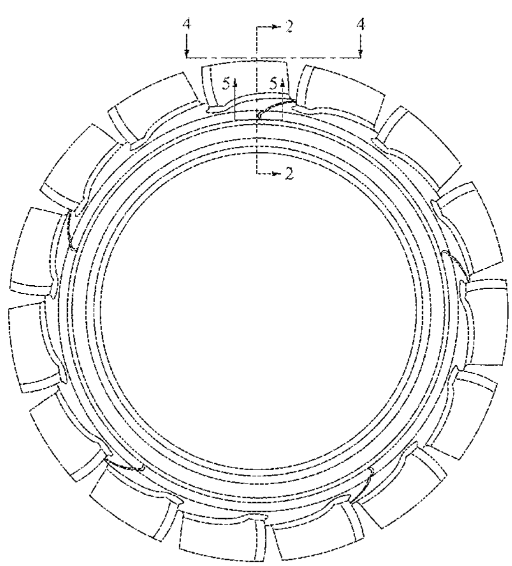
Primary Examiner — J. Todd Newton
(74) *Attorney, Agent, or Firm* — Bachman & LaPointe, P.C.

(57) **ABSTRACT**

An additively manufactured vane cluster has a platform, a shroud, and airfoils joining the platform to the shroud. The platform has openings, each opening between a respective two of the airfoils and forming accommodating differential thermal expansion of a leading end of the platform relative to a trailing end of the platform.

20 Claims, 14 Drawing Sheets



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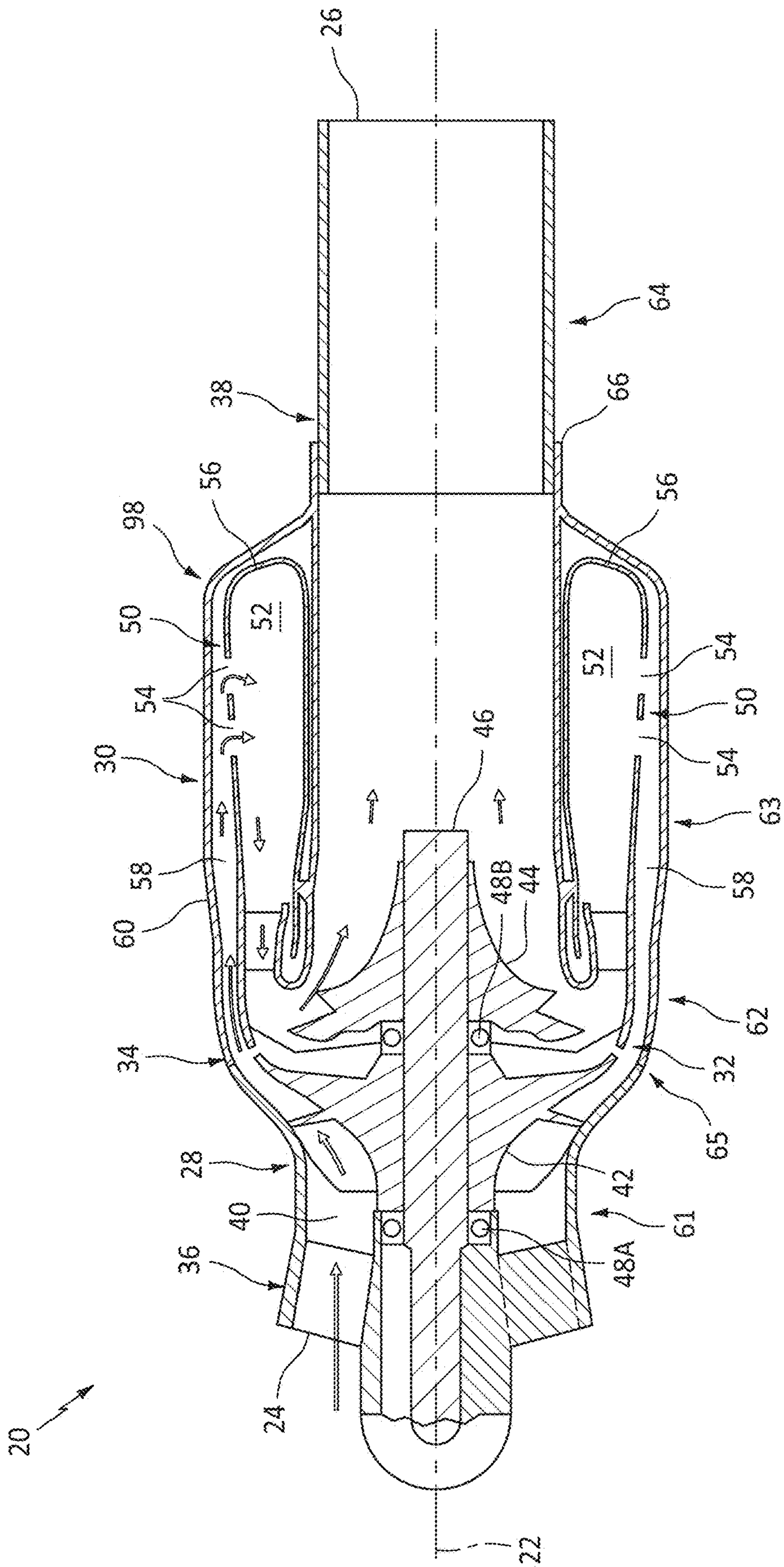
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FIG. 1

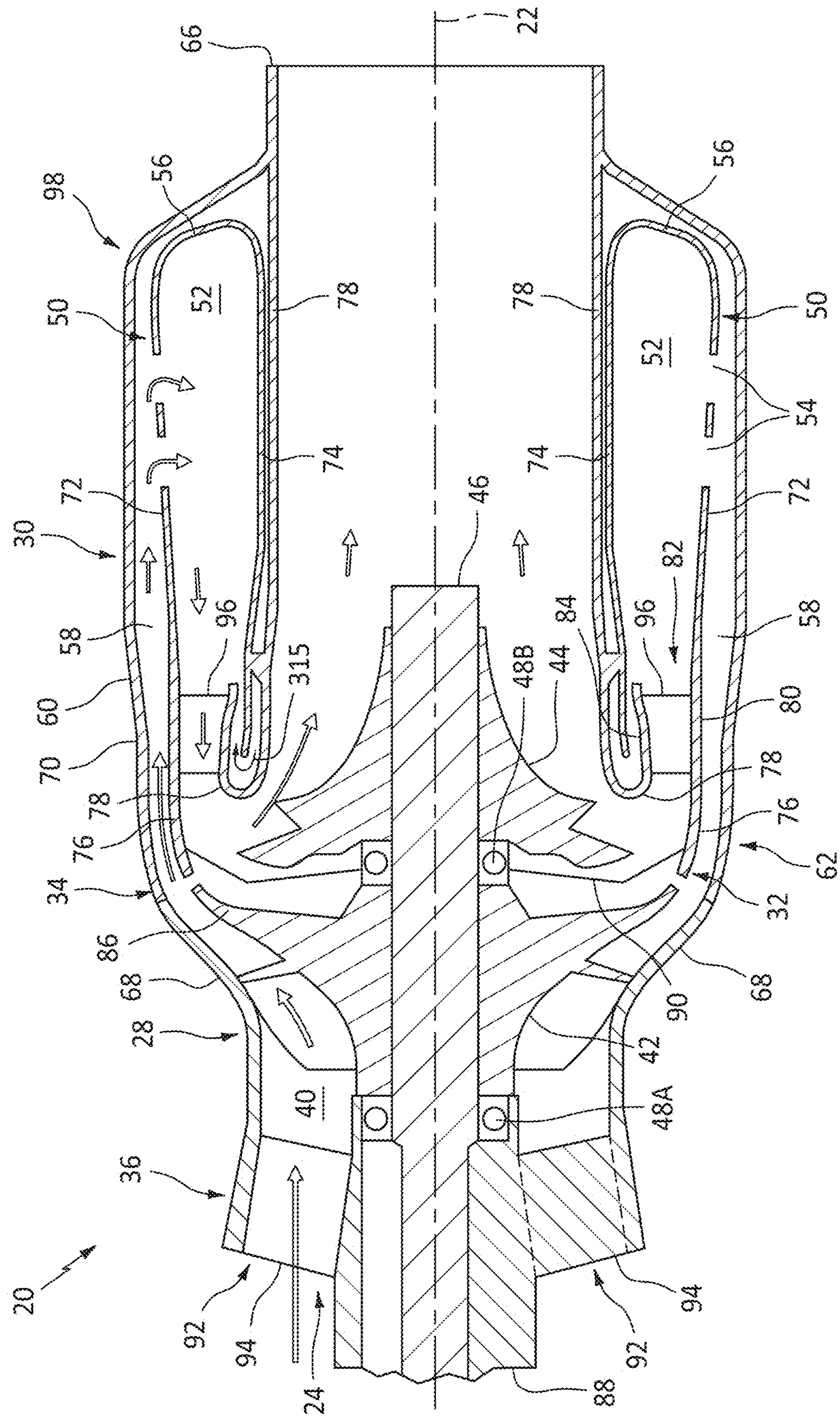
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FIG. 1A

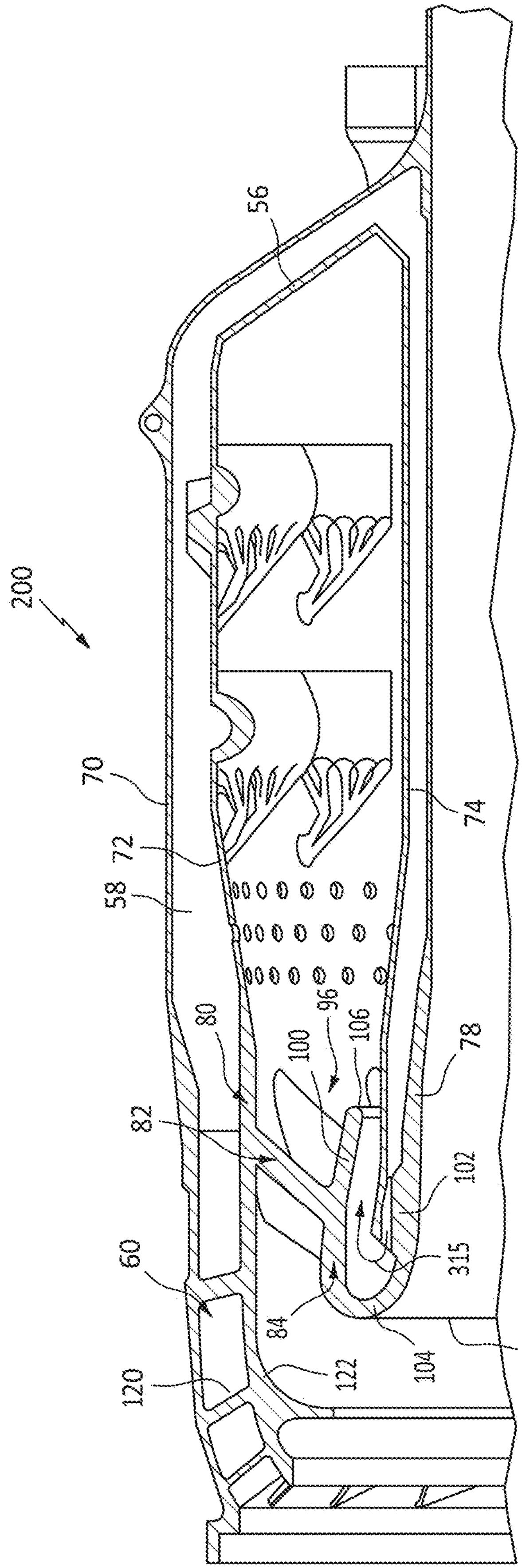

FIG. 2

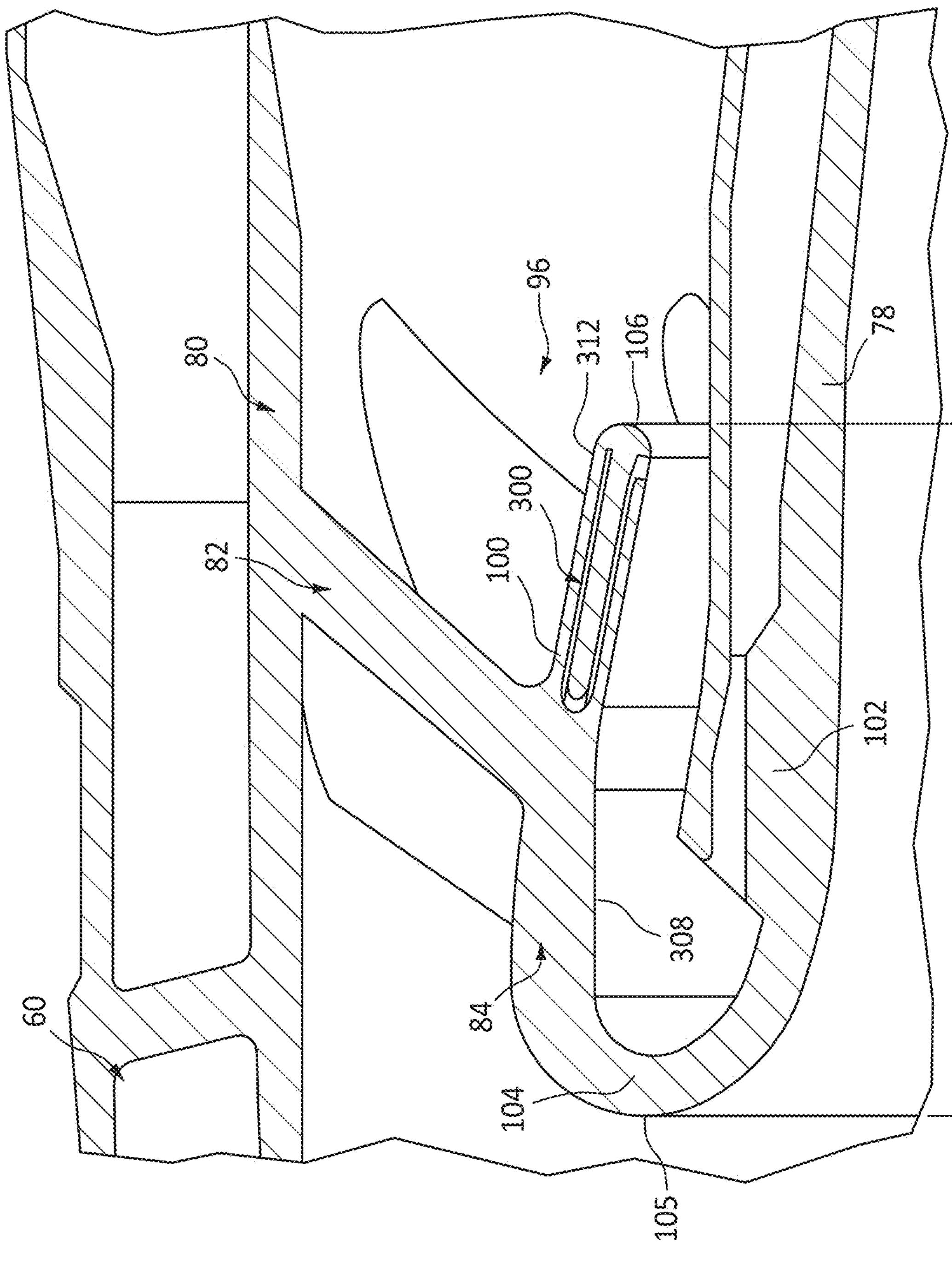

FIG. 2A

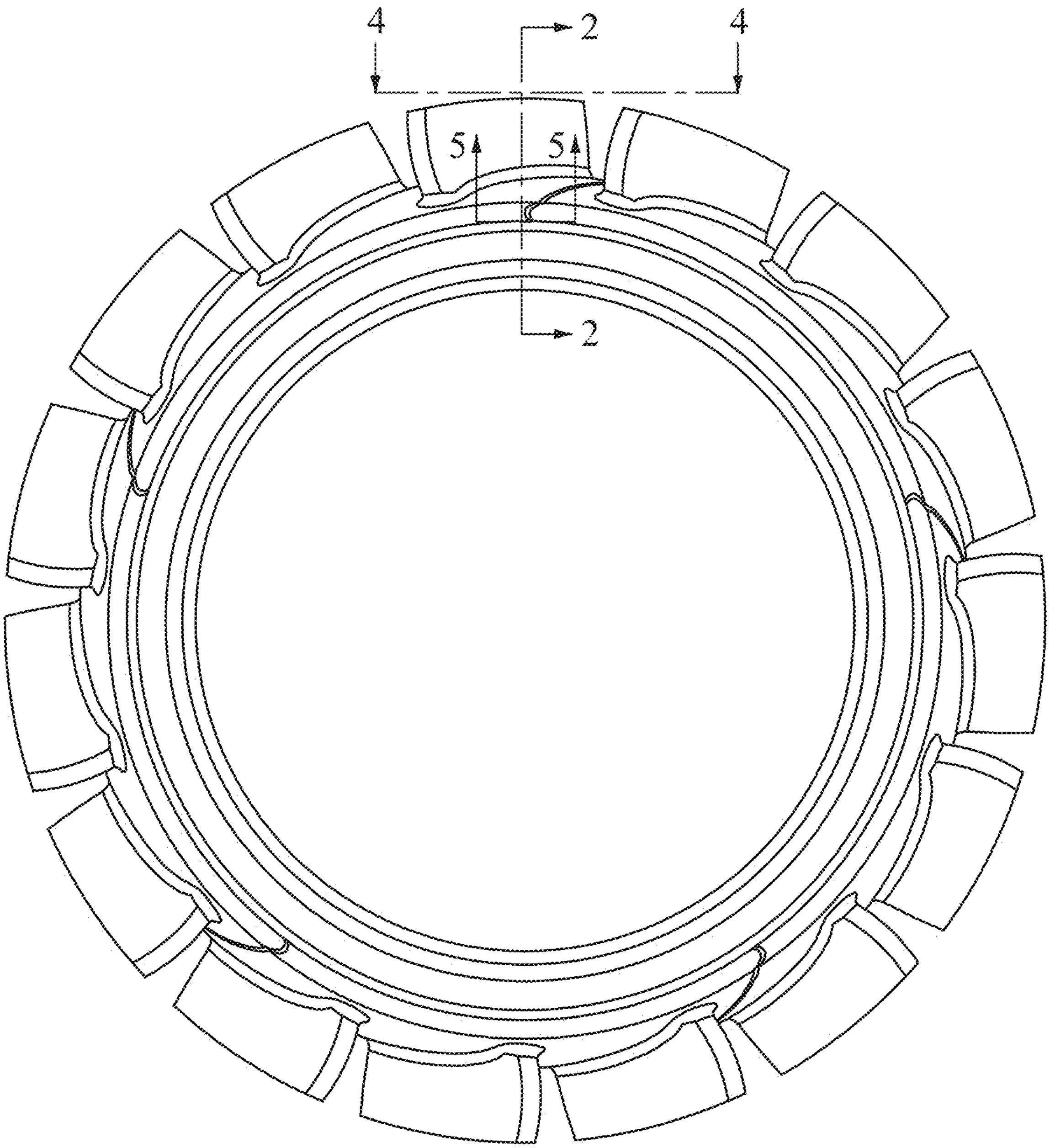

FIG. 3

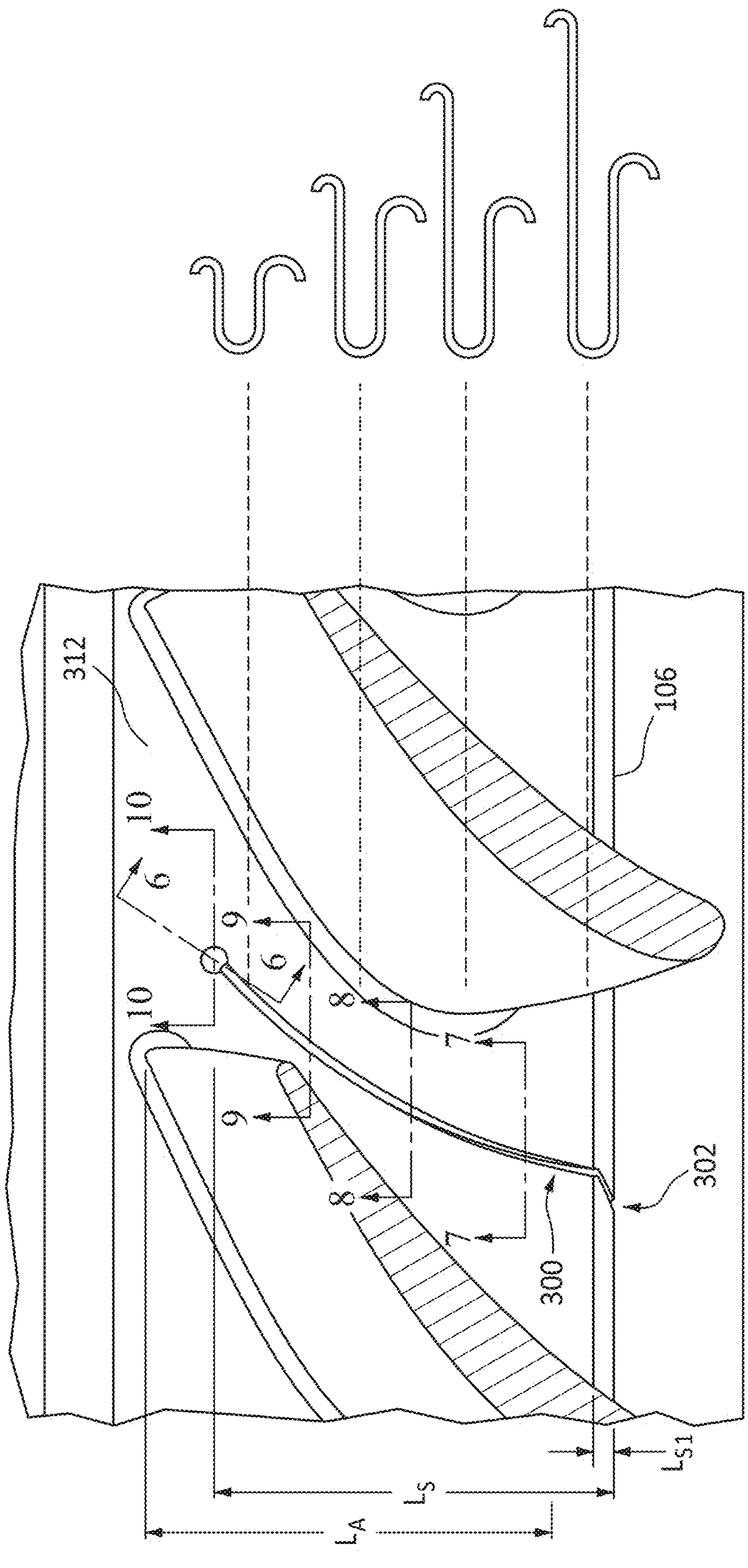

FIG. 4

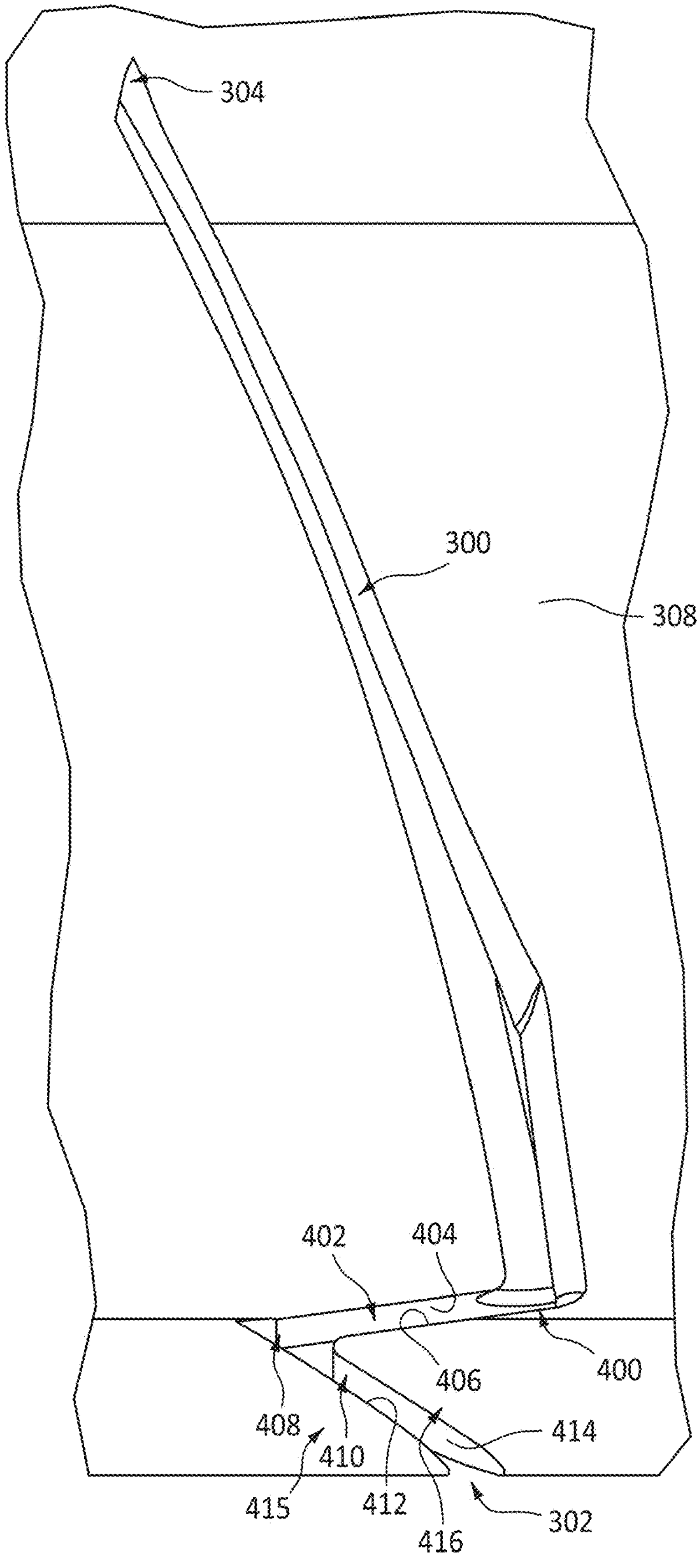

FIG. 5

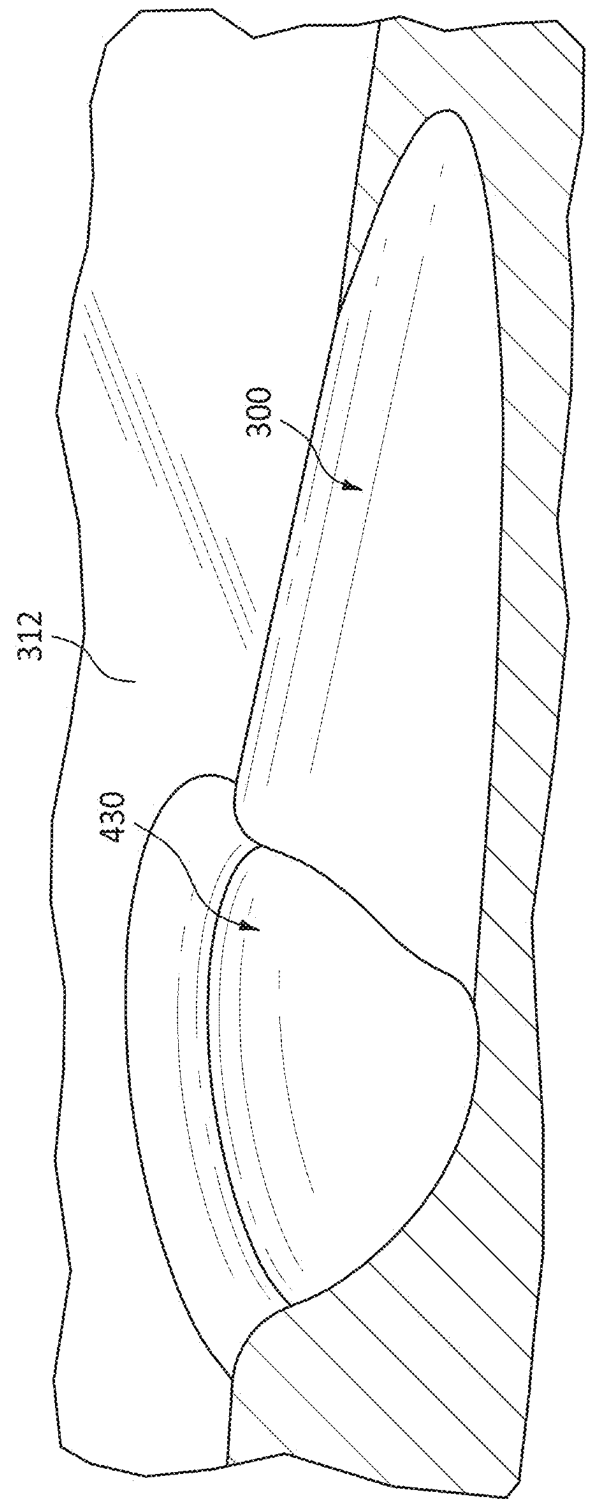

FIG. 6

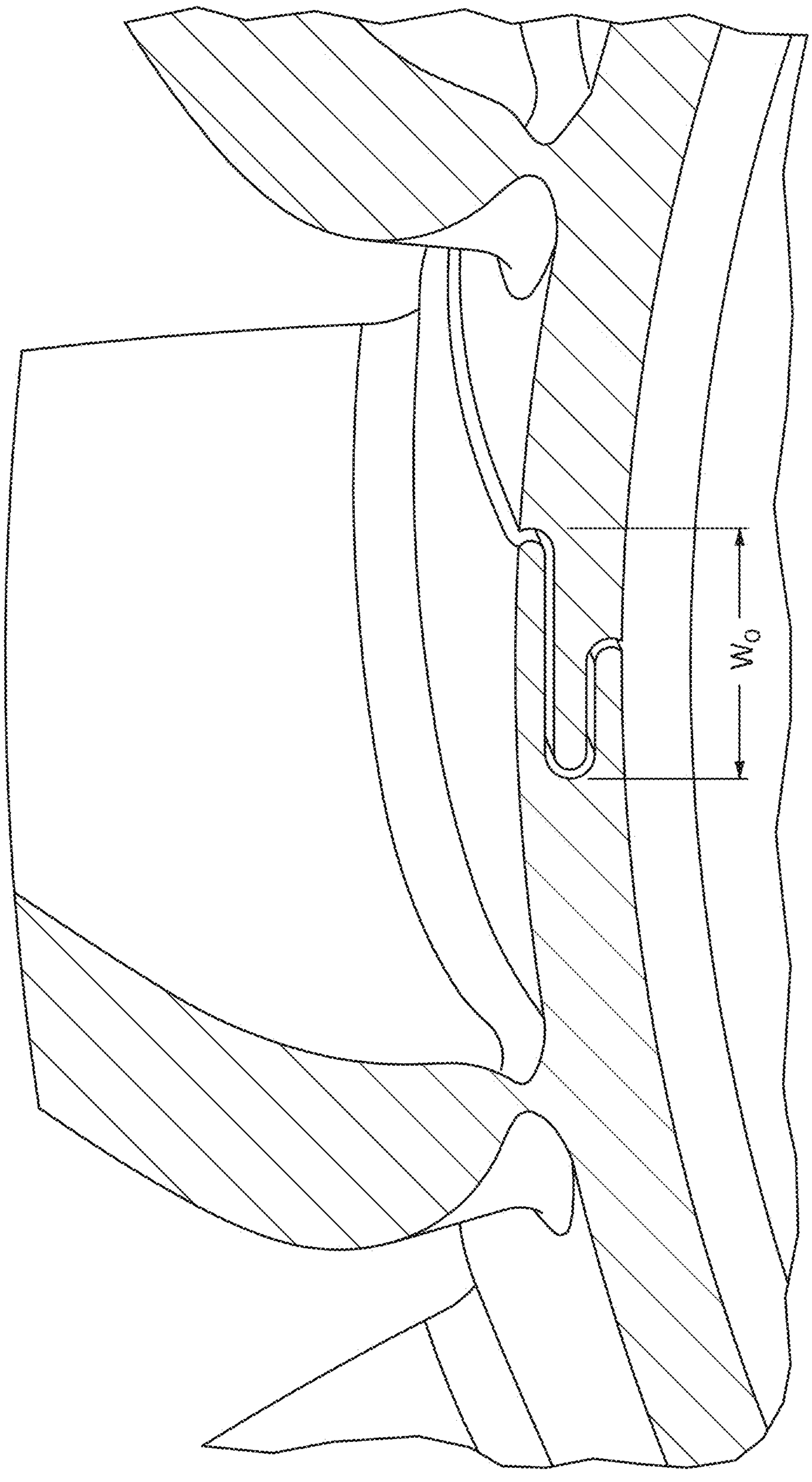

FIG. 7

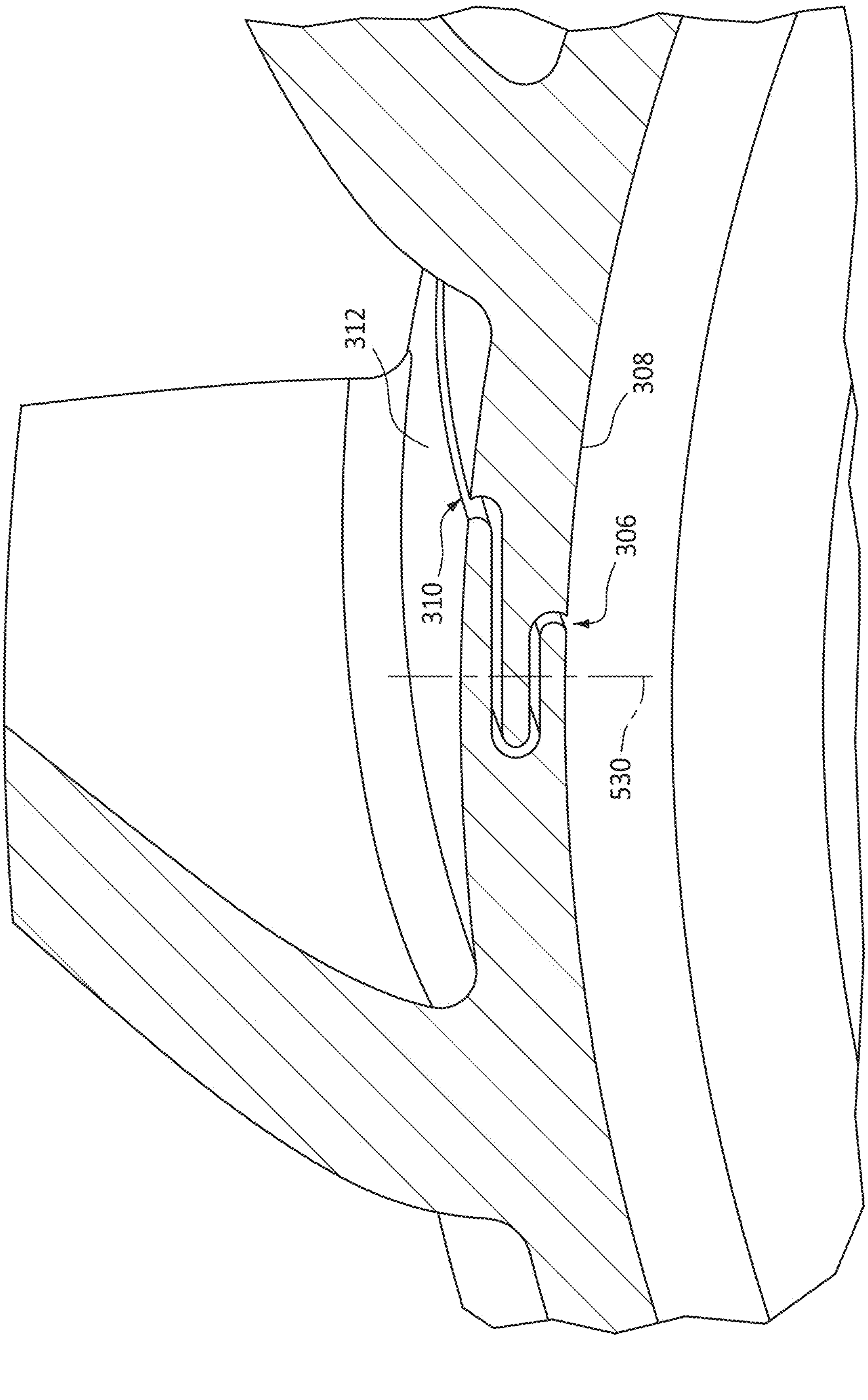

FIG. 8

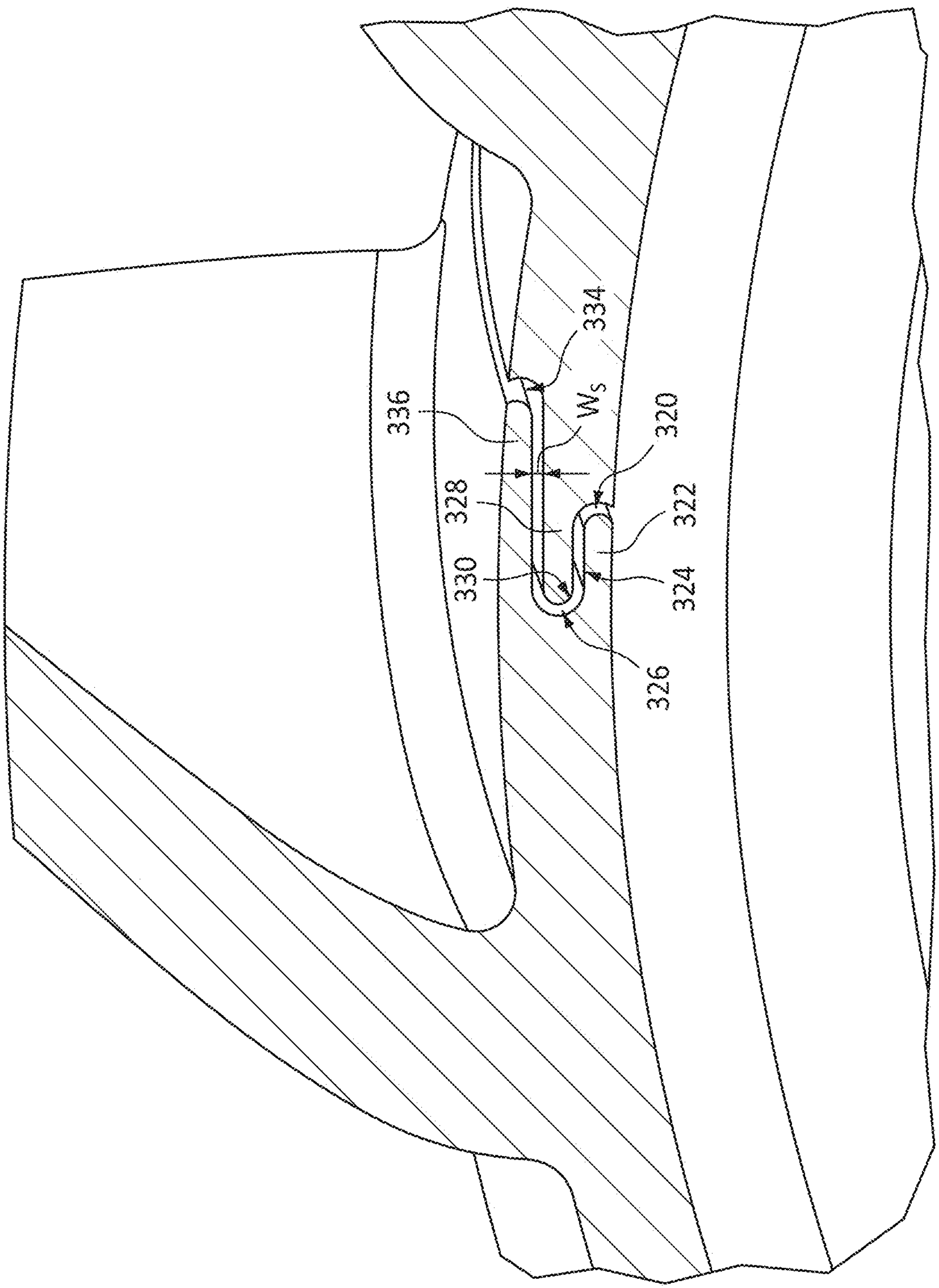

FIG. 9

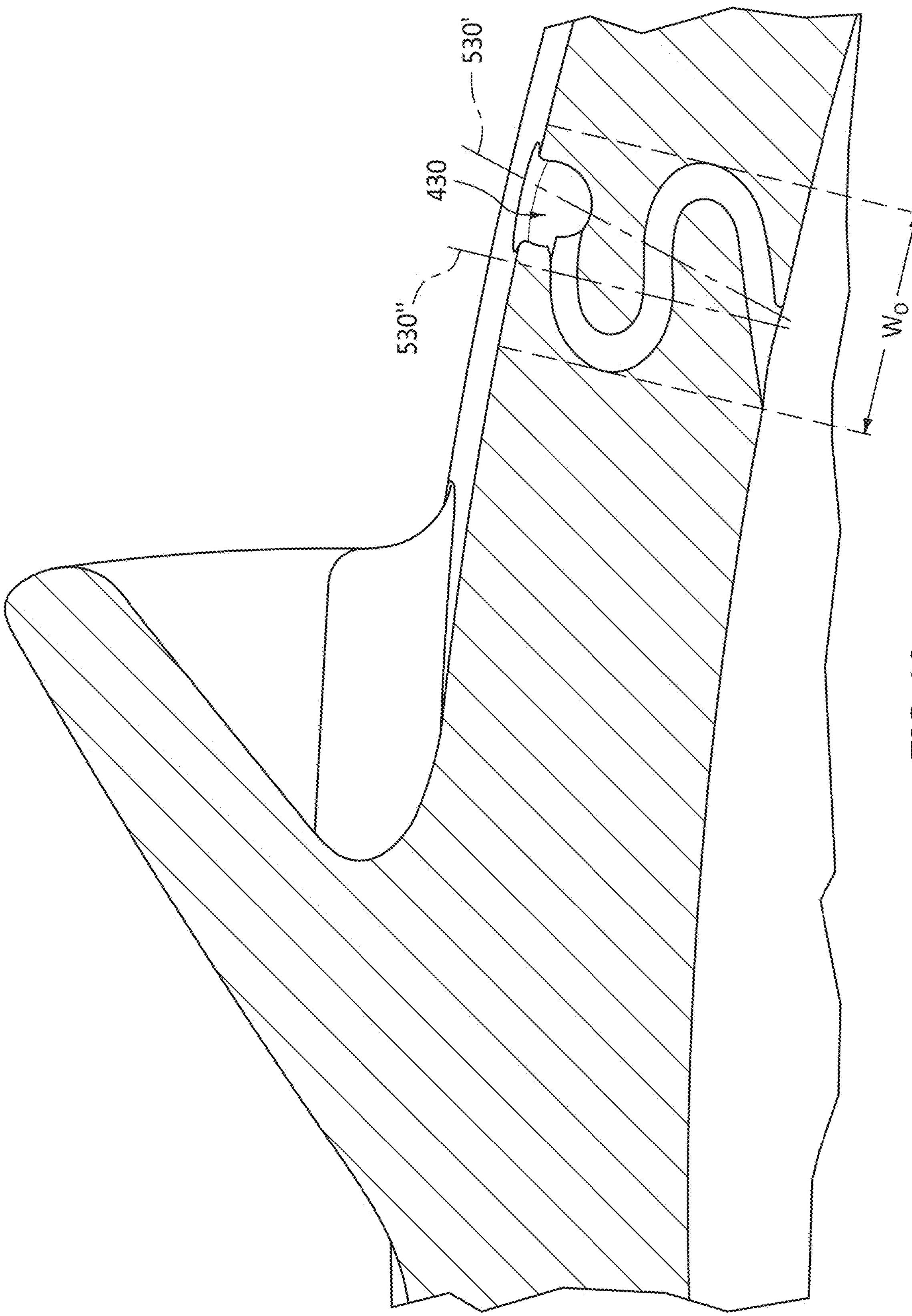

FIG. 10

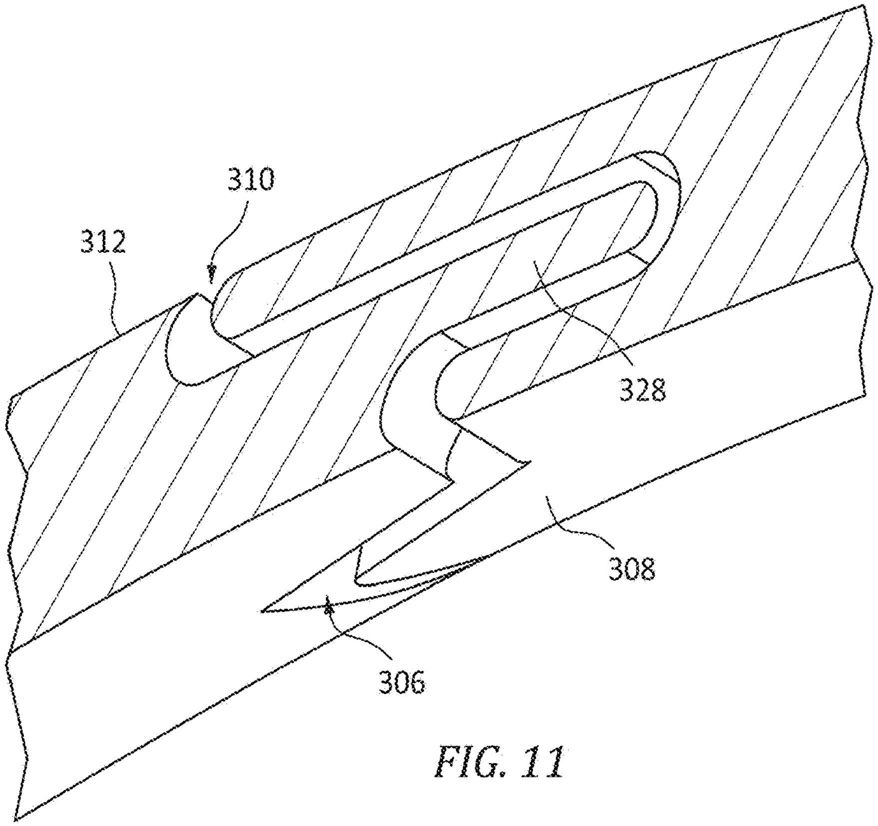

FIG. 11

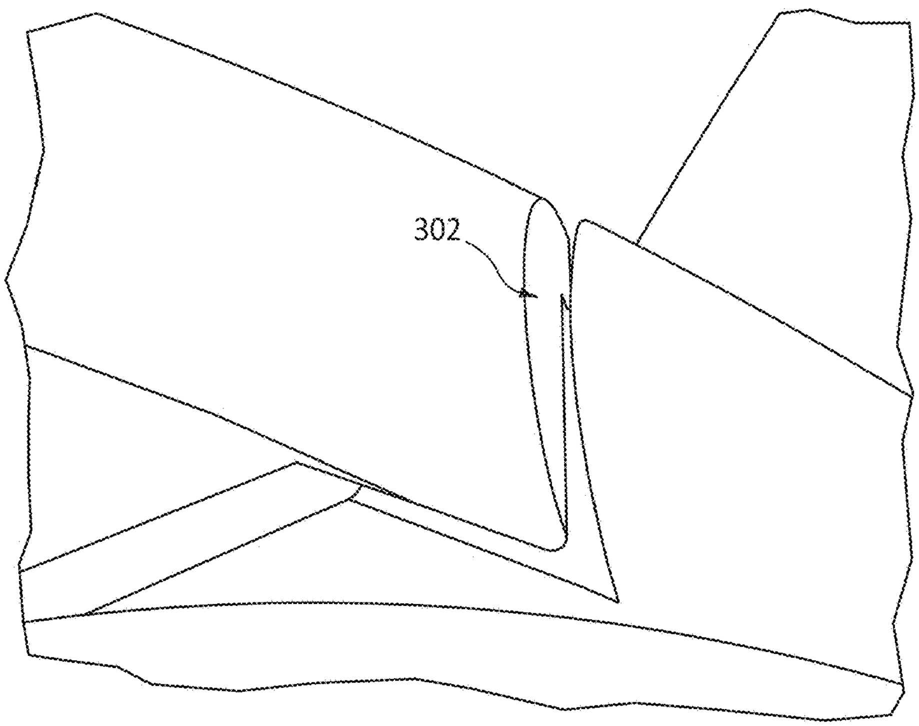

FIG. 12

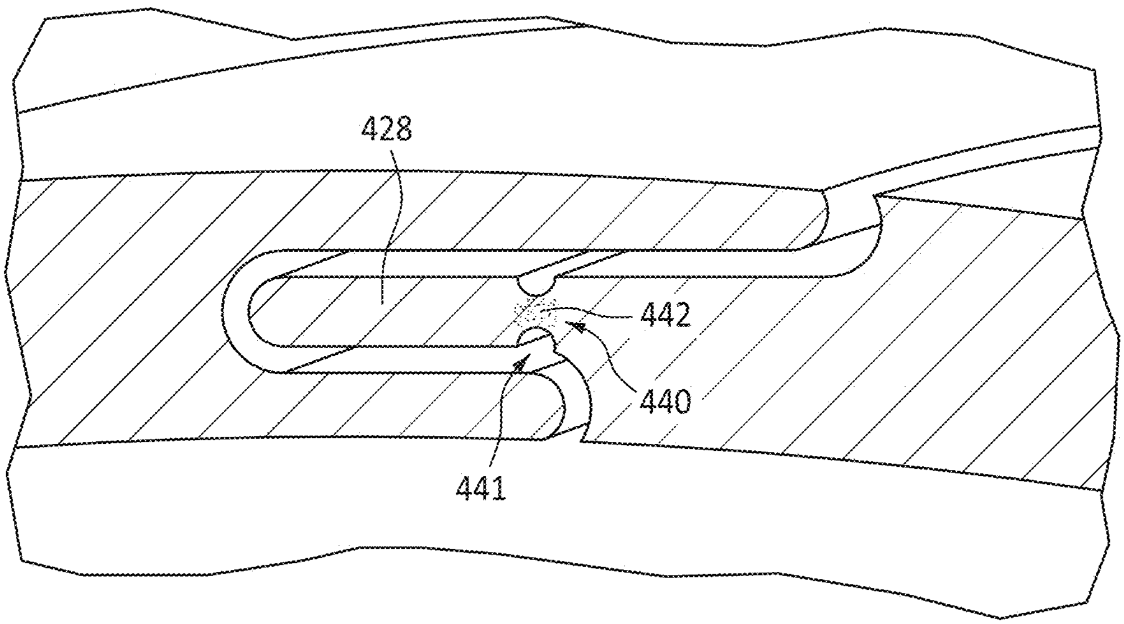

FIG. 13

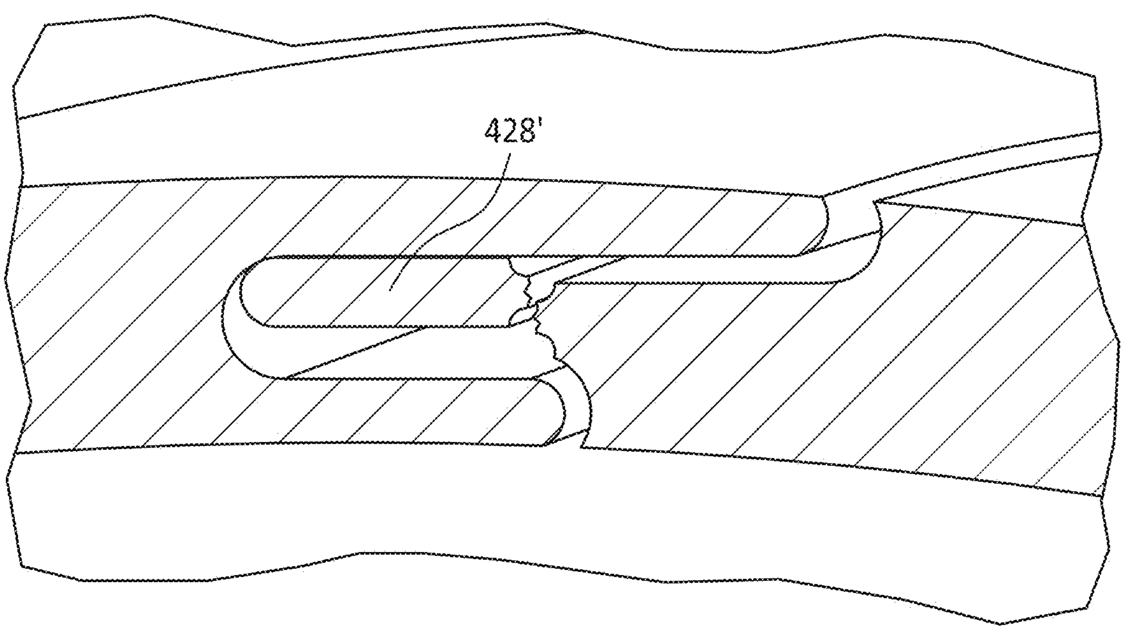

FIG. 14

ADDITIVELY MANUFACTURED TURBINE VANE CLUSTER

BACKGROUND

The disclosure relates to gas turbine engines. More particularly, the disclosure relates to turbine vane clusters for attritable engines.

Gas turbine engines (used in propulsion and power applications and broadly inclusive of turbojets, turboprops, turbofans, turboshafts, prop fans, industrial gas turbines, and the like) have spawned attritable variants particularly for turbojet, turbofan, and turboprop uncrewed aerial vehicles (UAV).

Example attritable engines are shown in U.S. Pat. No. 11,359,543B2 (the '543 patent) of Binek et al., issued Jun. 14, 2022, and entitled "Attritable Engine Additively Manufactured Inlet Cap" and U.S. Pat. No. 11,614,002B2 (the '002 patent) of Binek et al., issued Mar. 28, 2023, and entitled "Split Case Structure for a Gas Turbine Engine".

The disclosures of the '543 patent and '002 patent are incorporated by reference herein in their entireties as if set forth at length.

SUMMARY

One aspect of the disclosure involves a vane cluster comprising: a platform; a shroud; and a plurality of airfoils joining the platform to the shroud. The platform, shroud and airfoils are portions of a single piece. The platform has a plurality of openings, each opening between a respective two of the airfoils; the openings extend from a leading end of the platform toward a trailing end of the platform. The openings have a convoluted shape such that in transverse section a radial line has at least four intersections with the platform.

In a further embodiment of any of the foregoing embodiments, additionally and/or alternatively, the convoluted shape may exist over at least 50% of a total axial length of the opening, preferably, at least 70%.

In a further embodiment of any of the foregoing embodiments, additionally and/or alternatively, the slot may include a leading portion having fewer, if any, such intersections but blocking downstream line of sight into a portion of the slot having the convoluted shape. The leading portion may form a circumferential and upstream-to-downstream (e.g., aft-to-fore) zigzag footprint contrasted with the radial and circumferential convolutions in the convoluted portion. The leading portion may block axial flow, whereas the portion of the slot having the convoluted profile may block radial flow.

In a further embodiment of any of the foregoing embodiments, additionally and/or alternatively, the vane cluster is a full annulus.

In a further embodiment of any of the foregoing embodiments, additionally and/or alternatively, exactly every third inter-airfoil space has a said opening.

In a further embodiment of any of the foregoing embodiments, additionally and/or alternatively, the platform comprises an outer wall having the openings; an inner wall spaced radially inward of the outer wall; and a turn joining the inner wall and outer wall.

A further embodiment of any of the foregoing embodiments may additionally and/or alternatively include: a diffuser including diffuser vanes extending radially outward from the shroud; a case wall at outer diameter ends of the diffuser vanes; and a combustor body having an inner wall extending forward to merge with the platform along an outer diameter of the platform inner wall.

In a further embodiment of any of the foregoing embodiments, additionally and/or alternatively, said radial line has six said intersections.

In a further embodiment of any of the foregoing embodiments, additionally and/or alternatively, the opening shape defines: at a first circumferential side, a channel opening toward an opposite second circumferential side; and at the second circumferential side, a projection into the channel.

In a further embodiment of any of the foregoing embodiments, additionally and/or alternatively, the projection has one or more of: a necked area of reduced radial span; and a porous zone, optionally being said necked area if present, of at greater porosity than an adjacent portion of the platform, with a porosity difference of at least 50%.

In a further embodiment of any of the foregoing embodiments, additionally and/or alternatively, one or more of: the openings have an axial span of 50% to 130% of an axial span of the airfoils at the platform; no more than half of inter-airfoil spaces have said openings; and the openings have an axial span of 30% to 90% of an axial span of the platform.

In a further embodiment of any of the foregoing embodiments, additionally and/or alternatively, a gas turbine engine includes the vane cluster as a turbine section vane cluster and further comprising: a compressor section a combustor; and a gaspath defining a downstream direction sequentially through the compressor section, combustor, and turbine section.

In a further embodiment of any of the foregoing embodiments, additionally and/or alternatively, the engine is a single-spool engine, the compressor section is a centrifugal compressor, and the combustor is a reverse flow combustor.

In a further embodiment of any of the foregoing embodiments, additionally and/or alternatively, the vane cluster further comprises: a diffuser including diffuser vanes extending radially outward from the shroud; a case wall at outer diameter ends of the diffuser vanes; and a combustor body.

A further embodiment of any of the foregoing embodiments may additionally and/or alternatively include the method comprising additive manufacture forming the openings.

In a further embodiment of any of the foregoing embodiments, additionally and/or alternatively, the additive manufacturing comprises powder bed fusion.

A further embodiment of any of the foregoing embodiments may additionally and/or alternatively include the method comprising: running the vane cluster as a turbine vane cluster in a gas turbine engine; and the running causing thermal expansion of the leading end relative to the trailing end and circumferentially closing the openings.

In a further embodiment of any of the foregoing embodiments, additionally and/or alternatively, the running causes contact of a projection at one circumferential side of the opening with a channel at the other circumferential side of the opening.

In a further embodiment of any of the foregoing embodiments, additionally and/or alternatively, the contact includes rupturing a root section of the projection.

A further aspect of the disclosure involves vane cluster comprising: a platform; a shroud; and a plurality of airfoils joining the platform to the shroud. The platform has a plurality of openings, each opening between a respective two of the airfoils. The openings extend from a leading end of the platform. The openings define: at a first circumferential side, a channel opening toward an opposite second circumferential side; and at the second circumferential side a projection into the channel.

A further aspect of the disclosure involves a vane cluster comprising: a platform; a shroud; and a plurality of airfoils joining the platform to the shroud. The platform has a plurality of openings, each opening between a respective two of the airfoils and forming means for accommodating differential thermal expansion of a leading end of the platform relative to a trailing end of the platform.

In a further embodiment of any of the foregoing embodiments, additionally and/or alternatively, the vane cluster is a single-piece full annulus.

A further aspect of the disclosure involves a vane cluster comprising a platform having: an outer diameter wall; an inner diameter wall; and a turn joining the outer diameter wall and inner diameter wall. A plurality of airfoils join the platform outer diameter wall to a shroud. The platform outer diameter wall has a plurality of openings, each opening between a respective two of the airfoils and forming means for accommodating differential thermal expansion of the outer diameter wall and inner diameter wall.

A further aspect of the disclosure involves an additively manufactured vane cluster having a platform, a shroud, and airfoils joining the platform to the shroud. The platform has openings, each opening between a respective two of the airfoils and forming means for accommodating differential thermal expansion of a leading end of the platform relative to a trailing end of the platform.

The details of one or more embodiments are set forth in the accompanying drawings and the description below. Other features, objects, and advantages will be apparent from the description and drawings, and from the claims.

BRIEF DESCRIPTION OF THE DRAWINGS

FIG. **1** is a schematic central longitudinal sectional view of a gas turbine engine.

FIG. **1**A is an enlarged view of a portion of the engine of FIG. **1**.

FIG. **2** is a view of a single-piece diffuser/combustor/turbine inlet nozzle body.

FIG. **2**A is an enlarged view of the turbine inlet nozzle section of the body of FIG. **2**

FIG. **3** is a transverse sectional view of the nozzle section of FIG. **2**A.

FIG. **4** is a view of an inner platform OD surface in the nozzle section of FIG. **3** with schematic projections of slot geometry.

FIG. **5** is a view of an inner platform OD surface in the nozzle section of FIG. **3**.

FIG. **6** is a sectional view of a slot terminus.

FIG. **7** is a transverse sectional view of inner platform outer wall of the nozzle of FIG. **4**, taken along line **7-7**.

FIG. **8** is a transverse sectional view of inner platform outer wall of the nozzle of FIG. **4**, taken along line **8-8**.

FIG. **9** is a transverse sectional view of inner platform outer wall of the nozzle of FIG. **4**, taken along line **9-9**.

FIG. **10** is a transverse sectional view of inner platform outer wall of the nozzle of FIG. **4**, taken along line **10-10**.

FIG. **11** is a rearward cutaway view of a leading edge region of the inner platform outer wall.

FIG. **12** is a view of the leading edge region of the inner platform outer wall.

FIG. **13** is a sectional view showing a severable projection for forming a seal.

FIG. **14** is a sectional view showing the projection severed.

Like reference numbers and designations in the various drawings indicate like elements.

DETAILED DESCRIPTION

In a modification of the structure shown in the '002 patent above, one or more longitudinal sections of the case or static structure may be unitarily formed as a full annulus rather than circumferentially split structure. Thus, in one example discussed, a forward structure forming a compressor case **61** (FIG. **1**) may largely be one single full annulus piece and an aft structure **63** forming a diffuser, combustor body, and turbine case may largely be a second piece **200** (FIG. **2**) joined at a joint **65** such as a bolt circle at mating flanges. Such a single piece **200** may be additively manufactured such as via powder bed fusion-laser beam (PBF-LB), selective laser sintering (SLS), or directed energy deposition (DED). Example material is a nickel-based superalloy such as the Inconel family (e.g., Inconel 625). Nevertheless, features discussed below may be applied to split cases and may be applied to axially and/or radially less extensive pieces (e.g., wherein the combustor body (walls) and/or the diffuser are not part of the single piece).

For use in a such a reverse flow combustor gas turbine engine, FIG. **2** shows such a second piece **200** including: the diffuser **60**; combustor wall structures **56**,**72**,**74**; combustor exit nozzle or turbine inlet vane ring **82**; and turbine section wall structures **78**. The nozzle **82** has a circumferential array of airfoils or vanes **96** extending radially from inboard ends at an inner platform **84** to outboard ends at an outer shroud or platform **80**. In the reverse flow combustor situation, the airfoils **96** have upstream leading edges aft of forward trailing edges. The inner platform **84** is configured to provide the inside of a turn which turns the gaspath radially inward and back aft/rearward form the reverse flow combustor. Thus, the example inner platform **84** has a generally C-shaped central longitudinal section with a radially outer wall section **100** and, as portions of the turbine wall structure **78**, a radially inner wall section **102** and a turn **104** at a forward end of the inner platform **84**. The outer wall thus extends forward/downstream from an aft/upstream leading edge or rim **106** to a forward junction with the turn **104**. Similarly, the inner wall extends aft/downstream from a junction with the turn **104**.

FIG. **1** is a schematic central longitudinal sectional illustration of a gas turbine engine **20**. The gas turbine engine **20** of FIG. **1** is configured as a single spool, radial-flow turbojet turbine engine. This gas turbine engine **20** is configured for propelling an aircraft such as, but not limited to, an unmanned aerial vehicle (UAV), a drone or any other manned or unmanned aircraft or self-propelled projectile. The present disclosure, however, is not limited to such an example turbojet turbine engine configuration nor to an aircraft propulsion system application. For example, the gas turbine engine **20** may alternatively be configured as a turboshaft, a turboprop, an auxiliary power unit (APU), and/or an industrial gas turbine.

The gas turbine engine **20** of FIG. **1** extends axially along an axial centerline **22** between a forward, upstream airflow inlet **24** and an aft, downstream exhaust **26**. This axial centerline **22** may also be a rotational axis for various components within the gas turbine engine **20**.

The gas turbine engine **20** includes a compressor section **28**, a combustor section **30**, and a turbine section **32**. The gas turbine engine **20** also includes a static engine structure **34**.

This static engine structure **34** houses the compressor section **28**, the combustor section **30**, and the turbine section **32**. The static engine structure **34** of FIG. **1** also forms an inlet section **36** and an exhaust section **38** for the gas turbine engine **20**, where the inlet section **36** forms the airflow inlet **24** and the exhaust section **38** forms the exhaust **26**.

The engine sections **36**, **28**, **30**, **32** and **38** are arranged sequentially from upstream to downstream along a gaspath or core flowpath **40** that extends through the gas turbine engine **20** from the airflow inlet **24** to the exhaust **26**. Each of the engine sections **28** and **32** includes a respective rotor **42** and **44**. The example rotors are co-spooled to rotate as a unit. Each of these rotors **42**, **44** includes a plurality of rotor blades arranged circumferentially around and connected to at least one respective rotor hub (for centrifugal or disk for axial). The rotor blades, for example, may be formed integral with or mechanically fastened, welded, brazed, adhered and/or otherwise attached to the respective rotor disk(s).

The compressor rotor **42** may be configured as a centrifugal/radial flow rotor. The turbine rotor **44** may also be configured as a radial flow rotor. The compressor rotor **42** is connected to the turbine rotor **44** through a shaft **46**. This shaft **46** is rotatably supported by the static engine structure **34** through a plurality of bearings **48**A and **48**B (generally referred to as **48**); e.g., rolling element bearings, journal bearings, etc.

The combustor section **30** includes an example annular combustor **50** with an annular combustion chamber **52**. The combustor **50** of FIG. **2** is configured as a reverse flow combustor. Inlets ports/flow tubes **54** into the combustion chamber **52**, for example, may be arranged at (e.g., on, adjacent or proximate) and/or towards an aft bulkhead wall **56** of the combustor **50**. An outlet from the combustor **50** may be arranged axially aft of an inlet to the turbine section **32**. The combustor **50** may also be arranged radially outboard of and/or axially overlap at least a (e.g., aft) portion of the turbine section **32**. With this arrangement, the core flowpath **40** of FIG. **1** reverses direction (e.g., from a forward-to-aft direction to an aft-to-forward direction) a first time as the flowpath **40** extends from a diffuser plenum **58** surrounding the combustor **50** into the combustion chamber **52**. The core flowpath **40** of FIG. **1** then reverses direction (e.g., from the aft-to-forward direction to the forward-to-aft direction) a second time as the flowpath **40** extends from the combustion chamber **52** into the turbine section **32**.

During operation, air enters the gas turbine engine **20** through the inlet section **36** and its airflow inlet **24**. The inlet section **36** directs this air from the airflow inlet **24** into the core flowpath **40** and the compressor section **28**. The airflow inlet **24** of FIG. **1** thereby forms a forward, upstream inlet to the core flowpath **40** and the compressor section **28**. The air within the core flowpath **40** may be referred to as core air.

The core air is compressed by the compressor rotor **42** and directed through a diffuser **60** and its plenum **58** into the combustion chamber **52**. Fuel is injected and mixed with the compressed core air to provide a fuel-air mixture. This fuel-air mixture is ignited within the combustion chamber **52**, and combustion products thereof flow through the turbine section **32** and cause the turbine rotor **44** to rotate. This rotation of the turbine rotor **44** drives rotation of the compressor rotor **42** and, thus, compression of the air received from the airflow inlet **24**. The exhaust section **38** receives the combustion products from the turbine section **32**. The exhaust section **38** directs the received combustion products out of the gas turbine engine **20** to provide forward engine thrust.

The static engine structure **34** of FIG. **1** may include some or all static engine components included in the gas turbine engine **20**. Herein, the term "static" may describe a component that does not rotate with the rotating assembly or spool (e.g., an assembly of the rotors **42** and **44** and the shaft **46**) during gas turbine engine operation. A static component, for example, may refer to any component that remains stationary during gas turbine engine operation such as, but not limited to, a wall, a liner, a strut, a fixed vane, a fuel nozzle, a conduit, etc. The static engine structure **34** of FIG. **1**, for example, includes a forward, case structure **62** and an aft, exhaust duct structure **64**.

The example case structure **62** of FIG. **1**A is configured as a generally tubular structure formed in two general sections: an inlet/compressor case **61**; and a diffuser/combustor/turbine case **63**. The case structure **62**, for example, extends axially along the axial centerline **22** from the forward airflow inlet **24** to an outlet **66** from the turbine section **32**. The case structure **62** also extends circumferentially about (e.g., completely around) the axial centerline **22** such that the case structure **62** has, for example, a full hoop geometry. The two sections **61**, **63** may be secured to each other at a bolt flange joint **65**.

The case structure **62** includes one or more case walls. The inlet/compressor case **61** of FIG. **2**A, for example, includes a compressor wall **68**. The diffuser/combustor/turbine case **63** and its FIG. **2** main piece **200** have a diffuser wall **70**, an outer combustor wall **72** of the combustor **50**, an inner combustor wall **74** of the combustor **50**, the bulkhead wall **56** of the combustor **50**, an outer turbine wall **76** and an inner turbine wall **78**. Each of these case walls **56**, **68**, **70**, **72**, **74**, **76** and/or **78** may be generally tubular or generally annular. Each of the case walls **68**, **70**, **72**, **74**, **76**, **78** of FIG. **1**A, for example, is tubular, and the bulkhead wall **56** is annular.

The compressor wall **68** extends axially along the axial centerline **22** between and is connected to the inlet section **36** and the diffuser wall **70**. The compressor wall **68** of FIG. **2**A circumscribes, axially overlaps and thereby houses the compressor rotor **42**.

The diffuser wall **70** extends axially along the axial centerline **22** between and is connected to the compressor wall **68** and an aft end portion of the inner turbine wall **78**. The diffuser wall **70** is spaced/displaced radially outboard from and axially overlaps the combustor **50**. The diffuser wall **70** of FIG. **1**A thereby forms an outer peripheral boundary of the diffuser plenum **58** that surrounds the combustor **50** and the combustor wall **72** locally forms an inner boundary. FIG. **2** also shows diffuser vanes **120** radially between a forward portion of the diffuser wall **70** and an inner wall **122** that merges with the outer platform **80** which, in turn, merges with the combustor outer wall **72**.

The outer combustor wall **72** extends axially along the axial centerline **22** between and may be connected to the bulkhead wall **56** and an outer platform **80** of an exit nozzle or turbine inlet vane ring **82** from the combustion chamber **52**. The inner combustor wall **74** is circumscribed and axially overlapped by the outer combustor wall **72**. The inner combustor wall **74** extends axially along the axial centerline **22** between and may be connected to the bulkhead wall **56** and an inner platform **84** of the exit nozzle **82**. The bulkhead wall **56** extends radially between and is connected to aft end portions of the outer combustor wall **72** and the inner combustor wall **74**. The case walls **56**, **72** and **74** may thereby collectively form peripheral boundaries of the combustion chamber **52** therebetween.

The outer turbine wall **76** may be connected to the exit nozzle outer platform **80**. The outer turbine wall **76** projects axially out from the exit nozzle outer platform **80** and extends axially towards/to an aft, downstream end of an inner platform or hub **86** of the compressor rotor **42**. This outer turbine wall **76** is circumscribed and axially overlapped by the diffuser wall **70**. The outer turbine wall **76** of FIG. **1**A may thereby form an inner peripheral boundary of the core flowpath **40** within the diffuser **60**, and may form an outer peripheral boundary of the core flowpath **40** within a (e.g., upstream) portion of the turbine section **32**. The outer turbine wall **76** of FIG. **1**A also circumscribes, axially overlaps and thereby houses a (e.g., upstream) portion of the turbine rotor **44**.

The inner turbine wall **78** may be connected to the exit nozzle inner platform **84**. An upstream portion of the inner turbine wall **78** projects axially (in the aft-to-forward direction) out from the exit nozzle inner platform **84** to a turning portion of the inner turbine wall **78**. A downstream portion of the inner turbine wall **78** projects axially (in the forward-to-aft direction) away from the inner turbine wall turning portion to the turbine section outlet **66**. The inner turbine wall **78** is circumscribed and axially overlapped by the combustor **50**. The inner turbine wall **78** is also spaced/displaced radially inboard from the combustor **50**. The inner turbine wall **78** of FIG. **1**A thereby forms an inner peripheral boundary of the diffuser plenum **58** that surrounds the combustor **50**. The inner turbine wall **78** forms an outer peripheral boundary of the core flowpath **40** within a (e.g., downstream) portion of the turbine section **32**. The inner turbine wall **78** also circumscribes, axially overlaps and thereby houses a (e.g., downstream) portion of the turbine rotor **44**.

The static engine structure **34** may also include one or more internal support structures with one or more support members. Examples of support members include, but are not limited to, struts, structural guide vanes, bearing supports, bearing compartment walls, etc. The static engine structure **34** of FIG. **1**A, for example, includes a forward support structure **88**, an aft support structure **90**, an inlet nozzle **92** and the exit nozzle **82**. The forward support structure **88** and the inlet nozzle **92** may be configured together. The forward support structure **88** may be configured to support the forward bearing **48**A. The aft support structure **90** may be configured to support the aft bearing **48**B. The inlet nozzle **92** may be configured to condition the core air entering the compressor section **28**. The inlet nozzle **92**, for example, may include one or more guide vanes **94** which impart swirl to the core air. The exit nozzle **82** may similarly be configured to condition the combustion products exiting the combustor section **30**. The exit nozzle **82**, for example, may include one or more guide vanes **96** which import swirl to the combustion products, where these guide vanes **96** are connected to and extend radially between the exit nozzle inner and outer platforms **84** and **80**. The static engine structure **34**, of course, may also or alternative include various other static/stationary gas turbine engine components.

As discussed above, in an example engine having a reverse flow combustor, an example HPT vane has an outer diameter shroud and an inner diameter platform. The example platform is of generally c-shaped central longitudinal section, having: an outer diameter wall at inner diameter ends of the airfoils; an inner diameter wall spaced radially inward thereof; and forward turn joining those walls. The outer diameter wall generally forms an inner diameter boundary of the gaspath exiting the combustor; the platform turn then forms the inside/aft boundary of a turn of the gaspath radially inward toward the turbine inlet; and the inner diameter wall then forms the outer diameter boundary of the gaspath at or near the turbine section outlet.

In service, the outer diameter surface of the inner platform outer diameter (OD) wall (**100** in the FIG. **2** example) may be exposed to some of the hottest temperatures in the engine due to immediate exposure to gas leaving the combustor. Gas expansion in the turbine may cause a reduction in temperature from upstream to downstream over the gaspath-facing surface of the platform (the outer/forward surface of the platform turn **104** and then the ID surface of the ID wall **102**). Such temperature gradient can result in differential thermal expansion of the ID wall **102** and OD wall **100**. Such differential thermal expansion may cause problems which may include some combination of: performance degradation due to deformation; failure due to fracture; and penalties in weight and performance associated with restraining such deformation. Accordingly, a stress relief feature is added to the platform OD wall such as described further below.

The example stress relief feature is a circumferential array of radial through-openings (gaps or slots) **300** (FIG. **2**A) in the platform OD wall **100** extending downstream/forward from its leading end **106** (aft end for the reverse flow combustor). The example slots extend from a respective upstream open end **302** (FIG. **4** OD view and FIG. **5** ID view) and terminate at a terminus **304** (FIG. **5**) before reaching the opposite axial extreme **105** of the platform. Thus, they may terminate slightly before reaching the turn or slightly after in various examples. The slots have ID openings **306** (FIG. **8**) at/to the ID surface **308** of the OD wall **100** and OD openings **310** at/to the OD surface **312** of the OD wall **100**.

However, the slots represent a potential air/gas leakage path thus compromising efficiency. Accordingly, the number of slots may be limited to provide needed accommodation while limiting performance loss. In the example, the slots are not between every adjacent pair of airfoils. Instead, they are shown in FIG. **3** at an example between every third pair (thus at only one third of the available inter-airfoil gaps). A broader example is every second to fifth gap.

Additionally, the slots may be formed with a convoluted profile such that portions of the OD wall at either circumferential side of the slot interfit with each other. This may, effectively, form a labyrinth seal. Further variations on this are discussed below. In this particular example of a convoluted profile, the convolution is such that a radial line **530** (FIG. **8**) will have multiple intersections with the platform in addition to the inherent two intersections. In the particular configuration of FIG. **8**, over portions of the slot, there are six intersections.

The example slot thus has an ID opening **306** and an OD opening **310**. The convolution is such that there is, in transverse section, no straight line path between the ID opening and the OD opening. FIG. **8** shows a radial line with six intersections with the material. FIG. **10** shows a slightly off-radial line **530'** with four such intersections and a nearby radial line **530"** with six. Along most of the length of the slot, from ID to OD, the slot first proceeds radially outward. In this example, an initial outward radial pass **320** (FIG. **9**) is along a turn forming the convex end of an ID circumferentially projecting plate like projection **322** to one side (the counterclockwise side looking forward in the example). In the example, this finger protrudes in a circumferential direction (clockwise) from the pressure side of one adjacent airfoil toward the suction side of the other. The slot then turns in a circumferential direction (counterclockwise in the example) having an inner circumferential leg **324** extending back toward the pressure side. The slot then again turns radially outward in an arcuate turn **326** bounding the convex end of an intermediate circumferential projection **328** (on the clockwise side) and concave trough/base of a channel **330** (on the counterclockwise side). The slot then again turns circumferential in an outer circumferential leg **332**, finally turning back radially outward at turn **334** to form the convex end of an outer/OD circumferential projection **336** at the OD opening **310**.

This general cross-sectional shape is along a majority of the axial length L_S (FIG. **4**) of the slot. There are transitions near the leading end/rim of the platform outer wall to help block combustion gas from axially infiltrating the slot. This is achieved by having the slot not simply extend in the aforementioned cross-sectional shape all the way to the leading end/rim **106** of the platform. Instead, the slot extends in said interlocking projections to a location short of the leading end/rim and then takes a sharp circumferential turn **400** (FIG. **5**) so as to have a segment/leg **402** largely defined between radially and circumferentially extending faces **404**, **406** of the shroud at opposite sides. The slot then sharply turns **408** extends diagonally (aft and with an opposite circumferential component) along a leg **410** between radially and circumferentially extending faces **412**, **414** to the leading end/rim. This creates nesting sections **415**,**416** of the leading rim so that any combustion gas flow driven will have to make multiple turns to then enter the main portion of the slot. As is discussed below, such a labyrinthine cross-section may start after a leading region of span L_{S1} (FIG. **4**). Example L_{S1} is a relatively short span (e.g., 3% to 10% or 3% to 15% of L_S).

As shown in FIG. **2**A, as a characteristic platform length for the example reverse flow C-sectioned platform, a length L_P is defined as from the leading edge **106** of the platform to the forward extreme **105** of the turn **104**. In an alternative platform length lacking the C-shaped section, an alternative length may be simply between leading and trailing ends of a non-arcuate cross-section. Additionally, an airfoil length at the platform is shown as L_A (FIG. **4**).

Example L_S is about 100% of L_A, more broadly, 50% to 130% or 90% to 110%. Example L_S is about 70% of L_P, more broadly, 30% to 90% or 40% to 80% or 50% to 75%. The convoluted/labyrinthine profile blocking radial line of sight and, more generally, all ID to OD line of sight extends over a length of at least 50% of L_S, more particularly, at least 70% or 70% to 97% in embodiments that have a leading line of sight region of length L_{S1} as discussed above.

Additionally, as shown in FIGS. **7**-**10**, the slot cross-sectional overall dimension (width W_O which may be measured as a linear dimension or an angle) of the slot may decrease from the beginning of the convolution toward the terminus. This accommodates progressive thermal expansion from upstream to downstream (aft to fore in the reverse flow combustor). Additionally, the terminus includes a concave stress relief feature **430** (FIG. **6**) at the OD surface. The ID surface **308** is cooler than the OD surface **312** because it is exposed to pre-combustion dilution air **315** (FIG. **1**A).

Axially (longitudinally) and streamwise, the slots start at the leading/upstream end/rim of the platform and can extend up to the trailing edge of the vane airfoils. An example longitudinal span is at least half the longitudinal span of the airfoils at the platform and terminating at least 75% of the chord or the longitudinal length. As the engine runs, temperatures increase, resulting in thermal expansion of material. The slots accommodate differential thermal expansion (due to streamwise gradient along the platform outer wall, due to gradient radially across the platform outer wall, and/or due to radial gradient causing differential expansion between the platform outer wall on the one hand and the platform inner wall and/or shroud on the other hand). This accommodation may improve durability, allowing increased lifespan and/or greater operational parameter domain.

Radially, the slots act as a tortuous path, so combustor cooling air is less likely to bleed through the platform and instead enter the combustion chamber. The path is a series of interlocking concentric (arcuate with center of curvature at or near the engine centerline/rotation axis) features which enables material expansion whilst limiting the amount of bleed air.

An example slot width W_S (face to face) is 0.010 inch (0.25 millimeters, more broadly 0.20 millimeter to 1.0 millimeter with alternative upper ends of 0.40, 0.50, and 0.75 millimeter). The slot begins with a narrow opening then expands into a more labyrinthine profile with respect to travel along the platform. In the illustrated example, a portion of the slot at leading edge **106** and extending from the opening **302** extends generally radially from ID surface **308** to OD surface **312** and zigzags circumferentially and forward until the transition to the labyrinthine profile. Circumferentially projecting axial overlap of sections **415**, **416** (FIG. **5**) of the leading portion thus block axial access (along a majority of a radial span) but may have a radial or near radial line of sight openness/clearance. The labyrinthine profile lacks radial or near radial line of sight openness but is more axially open between the transition from the leading portion to the terminus. Thus, from the open end **302**. the transverse section of the slot appears close to a radially extending rectangle, abruptly transitioning to the convoluted shape at the end of the leg **402**.

In this instance, the main section of the slot is formed of a passage with bends, the lengths of which are concentric to the platform. This ensures that as the material grows, it doesn't impinge on itself generating stress concentrations.

In use, differential heating will cause the leading end of the platform to heat more and thus expand more than the trailing (forward) end. This causes the slot to circumferentially constrict, further improving sealing and reducing any leakage flow. The circumferential width of the slot will decrease, locally substantially reducing W_S at the circumferential end/tips of the projections **322**,**328**, **336**.

The sectional views of FIGS. **7**-**10** show how the passage profile changes. This tortuous passage restricts flow from bleeding through the bottom of the platform instead of passing around and entering through the turbine vane inlet.

The slot terminates with a small roughly hemispherical feature on the OD face which helps reduce effects of stress concentration. The feature does not perforate both sides of the platform and thereby does not cause additional gas bypass through the platform. Without this feature, the slot would act as a crack initiation site. The illustrated example includes this feature at the OD. In other examples, there may be a similar such feature at the ID.

Additionally, in some examples, the slot may include an integral feather seal which may be configured to deform during engine operation and help close the bypass passage. In the specific example, the intermediate projection **328** may be allowed to deform or fracture. For example, a weakened region **440** (FIG. **13**) of projection **428** may be formed near a proximal/root end of the projection allowing the portion distally thereof to deflect radially outward and seal against the ID face/side of the OD/outer projection. For example, the projection can be additively defined in a way that the weakened region is necked from ID and/or OD by channels **441**.

Such region where the projection/seal connects to the remainder of the platform may additionally or alternatively contain induced porosity **442** allowing the seal to deflect easier (e.g., hinging at the region **440**) or even detach. The necked region **440** (or unnecked) could be parametrically defined in the PBF process to contain porosity **442** to aid in deflection or liberation. If the seal disconnects (FIG. **14**) it will allow the platform to flex and block gas bypass. The example PBF process may achieve such porosity via control of speed (laser relative to bed) and feed (laser power) in what would be an otherwise undesirable combination such as increased/excess speed and/or decreased/insufficient power. The result is macroporous so-called "key hole" defects. An example porosity of the region is at least 10% or at least 50%. Away from the region, example material is near fully dense (e.g., 99% or more). An example porosity difference or delta is at least 10% or at least 50% or an example 50% to 80% or 50% to 75% between porous material of the region relative and remaining material of the platform.

Component materials and manufacture techniques and assembly techniques may be otherwise conventional.

The use of "first", "second", and the like in the following claims is for differentiation within the claim only and does not necessarily indicate relative or absolute importance or temporal order. Similarly, the identification in a claim of one element as "first" (or the like) does not preclude such "first" element from identifying an element that is referred to as "second" (or the like) in another claim or in the description.

A difference in porosity is measured in absolute terms. Thus, a region of 10% porosity has a difference of 5% relative to a region of 5% porosity (not a relative difference of 100%).

One or more embodiments have been described. Nevertheless, it will be understood that various modifications may be made. For example, when applied to an existing baseline cluster configuration, details of such baseline may influence details of particular implementations. Accordingly, other embodiments are within the scope of the following claims.

What is claimed is:

1. A vane cluster comprising:
a platform;
a shroud; and
a plurality of airfoils joining the platform to the shroud, wherein:
the platform, shroud and airfoils are portions of a single piece;
the platform has a plurality of openings, each opening between a respective two of the airfoils;
the openings extend from a leading end of the platform toward a trailing end of the platform; and
the openings have a convoluted shape such that in transverse section a radial line has at least four intersections with the platform.

2. The vane cluster of claim **1** being a full annulus.

3. The vane cluster of claim **2** wherein:
every third inter-airfoil space has a said opening.

4. The vane cluster of claim **1** wherein the platform comprises:
an outer wall having the openings;
an inner wall spaced radially inward of the outer wall; and
a turn joining the inner wall and outer wall.

5. The vane cluster of claim **4**, further comprising:
a diffuser including diffuser vanes extending radially outward from the shroud;
a case wall at outer diameter ends of the diffuser vanes; and
a combustor body having an inner wall extending forward to merge with the platform along an outer diameter of the platform inner wall.

6. The vane cluster of claim **1**, wherein: said radial line has six said intersections.

7. The vane cluster of claim **1** wherein the opening shape defines:
at a first circumferential side, a channel opening toward an opposite second circumferential side; and
at the second circumferential side, a projection into the channel.

8. The vane cluster of claim **7**, wherein the projection has one or more of:
a necked area of reduced radial span; and
a porous zone, optionally being said necked area if present, of at greater porosity than an adjacent portion of the platform, with a porosity difference of at least 50%.

9. The vane cluster of claim **1** wherein one or more of:
the openings have an axial span of 50% to 130% of an axial span of the airfoils at the platform;
no more than half of inter-airfoil spaces have said openings; and
the openings have an axial span of 30% to 90% of an axial span of the platform.

10. A gas turbine engine including the vane cluster of claim **1** as a turbine section vane cluster and further comprising:
a compressor section;
a combustor; and
a gaspath defining a downstream direction sequentially through the compressor section, combustor, and turbine section.

11. The gas turbine engine of claim **1** wherein:
the engine is a single-spool engine;
the compressor section is a centrifugal compressor; and
the combustor is a reverse flow combustor.

12. The gas turbine engine of claim **11** wherein the vane cluster further comprises:
a diffuser including diffuser vanes extending radially outward from the shroud;
a case wall at outer diameter ends of the diffuser vanes; and
a combustor body.

13. A method for manufacturing the vane cluster of claim **1**, the method comprising:
additive manufacture forming the openings.

14. The method of claim **13** wherein the additive manufacturing comprises:
powder bed fusion.

15. A method for using the vane cluster of claim **1**, the method comprising:
running the vane cluster as a turbine vane cluster in a gas turbine engine; and
the running causing thermal expansion of the leading end relative to the trailing end and circumferentially closing the openings.

16. The method of claim **15** wherein:
the running causes contact of a projection at one circumferential side of the opening with a channel at the other circumferential side of the opening.

17. The method of claim **16** wherein:

the contact includes rupturing a root section of the projection.

18. A vane cluster comprising:

a platform;

a shroud; and

a plurality of airfoils joining the platform to the shroud, wherein:

the platform has a plurality of openings, each opening between a respective two of the airfoils;

the openings extend from a leading end of the platform; and

the openings define:

at a first circumferential side, a channel opening toward an opposite second circumferential side; and

at the second circumferential side a projection into the channel.

19. A vane cluster comprising:

a platform;

a shroud; and

a plurality of airfoils joining the platform to the shroud, wherein:

the platform has a plurality of openings, each opening between a respective two of the airfoils and forming means for accommodating differential thermal expansion of a leading end of the platform relative to a trailing end of the platform.

20. The vane cluster of claim **19** wherein:

the vane cluster is a single-piece full annulus.

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