US010385716B2

(12) **United States Patent**
Lutjen et al.

(10) **Patent No.: US 10,385,716 B2**
(45) **Date of Patent: Aug. 20, 2019**

(54) **SEAL FOR A GAS TURBINE ENGINE**

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(*) Notice: Subject to any disclaimer, the term of this patent is extended or adjusted under 35 U.S.C. 154(b) by 656 days.

(21) Appl. No.: **14/790,076**

(22) Filed: **Jul. 2, 2015**

(65) **Prior Publication Data**

US 2017/0002675 A1 Jan. 5, 2017

(51) **Int. Cl.**

F01D 11/08	(2006.01)
F01D 11/00	(2006.01)
F01D 9/04	(2006.01)
F01D 25/24	(2006.01)
F02C 7/28	(2006.01)
F01D 11/12	(2006.01)
F01D 9/02	(2006.01)

(52) **U.S. Cl.**

CPC ***F01D 11/08*** (2013.01); ***F01D 9/02*** (2013.01); ***F01D 9/042*** (2013.01); ***F01D 11/001*** (2013.01); ***F01D 11/005*** (2013.01); ***F01D 11/122*** (2013.01); ***F01D 25/246*** (2013.01); ***F02C 7/28*** (2013.01)

(58) **Field of Classification Search**

CPC F01D 9/023; F01D 11/005; F01D 11/025; F01D 25/243; F01D 25/246; F05D 2240/11; F05D 2240/57; F05D 2260/30

See application file for complete search history.

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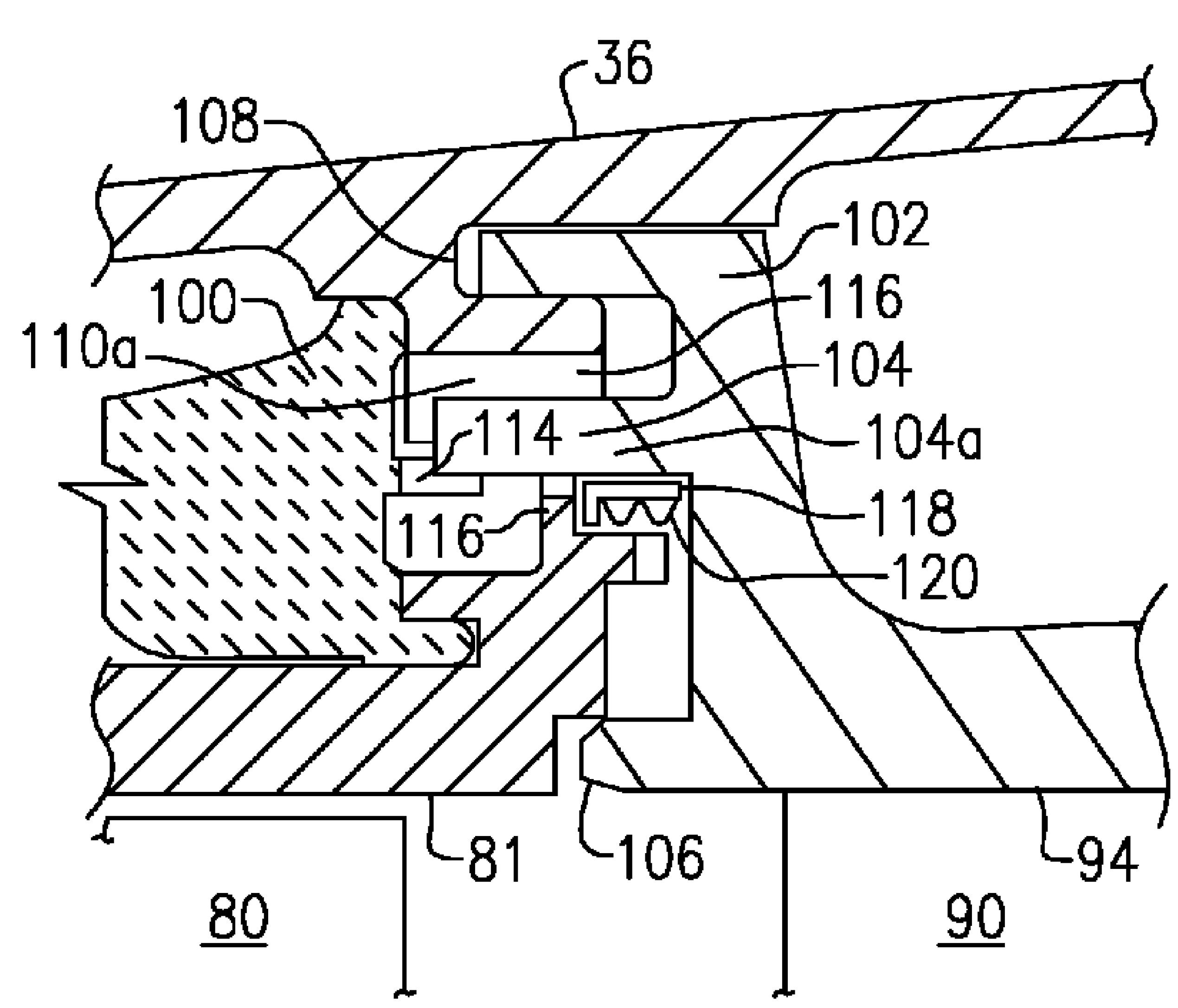
(57) **ABSTRACT**

A gas turbine engine assembly includes a shield that has a first portion and a second portion. The first portion extends radially from an axial end portion of the shield and includes a blade outer air seal contact surface. The second portion extends axially from a radially outer end of the first portion and includes a vane contact surface.

20 Claims, 3 Drawing Sheets



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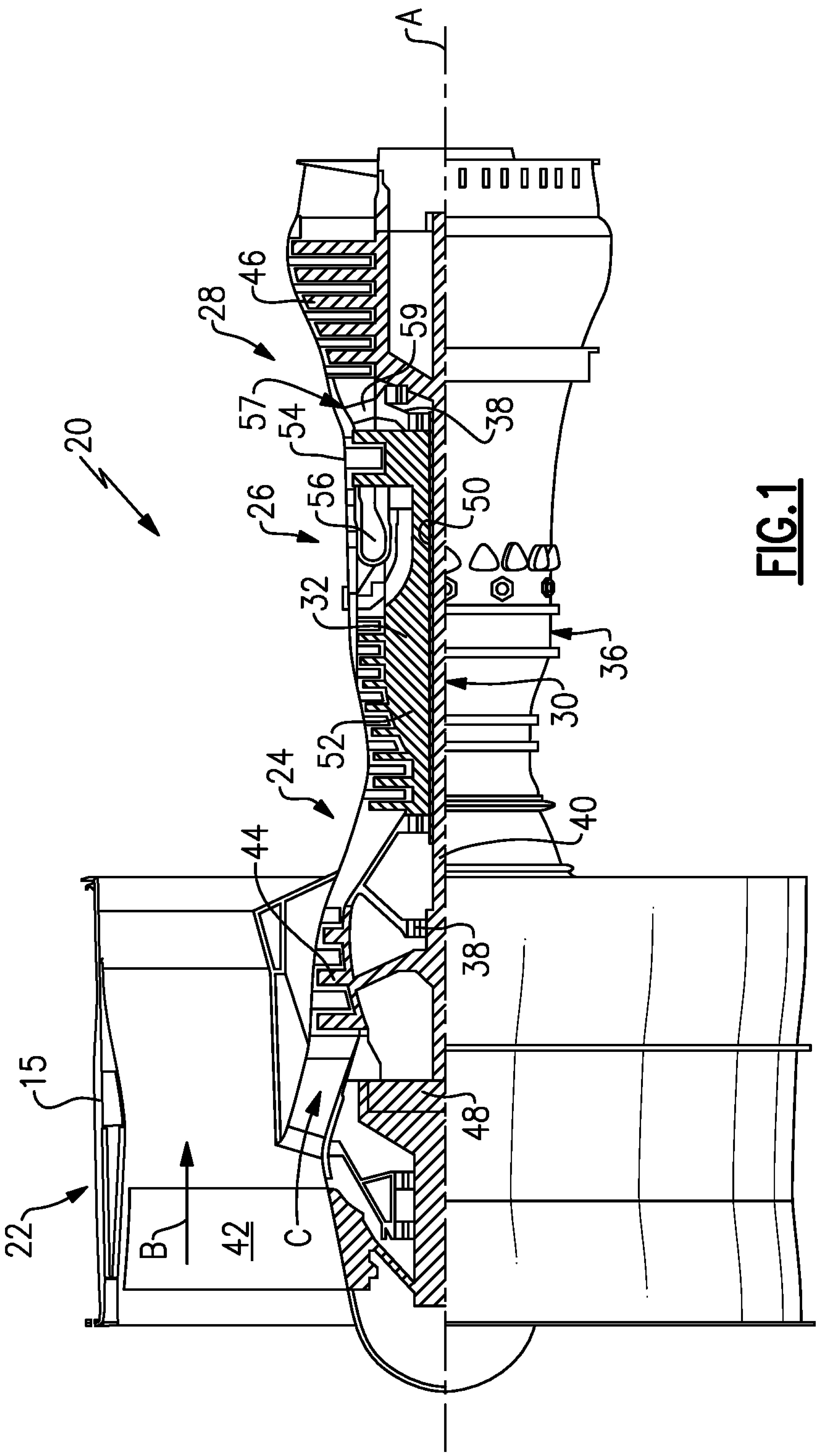

FIG.1

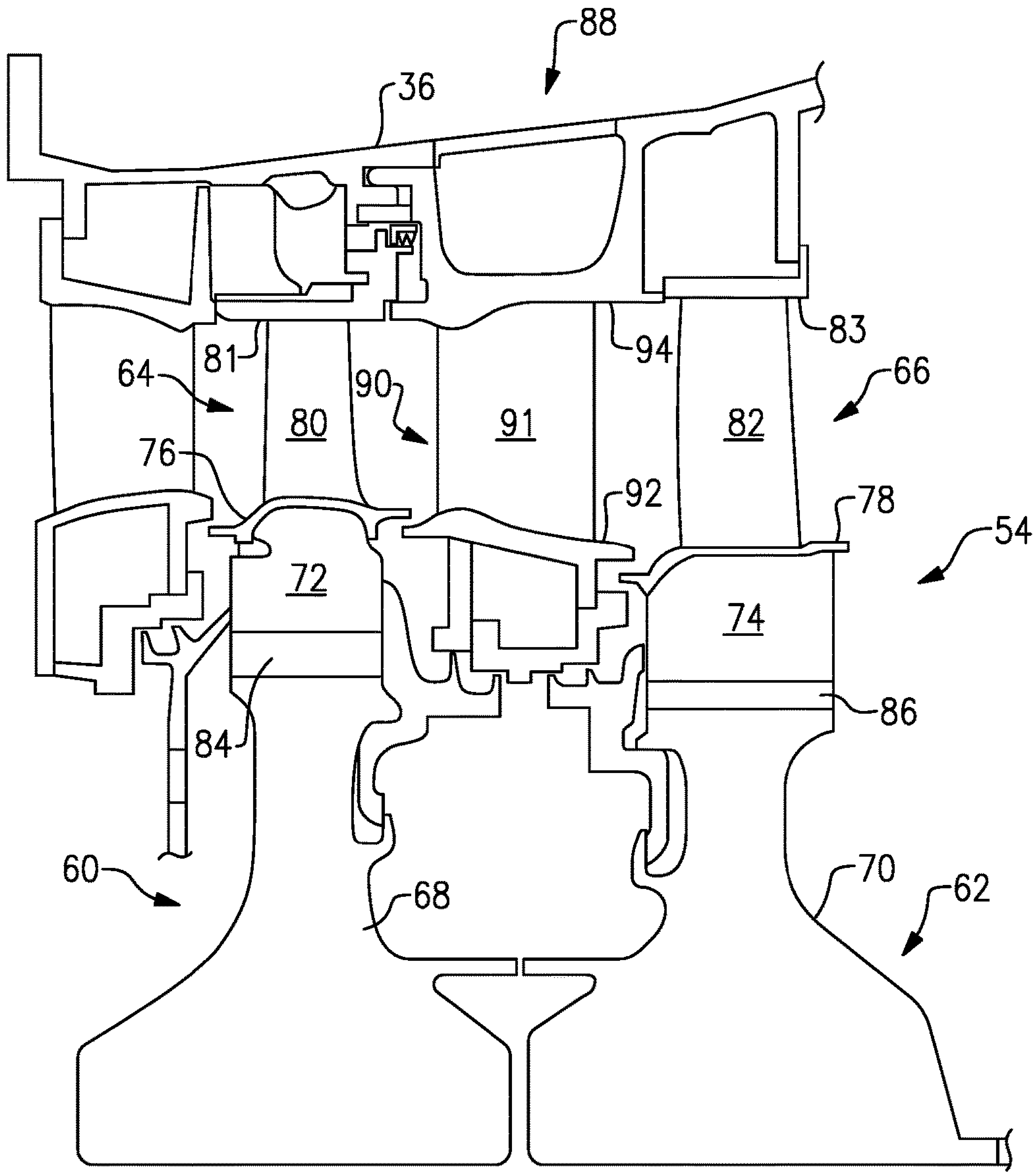

FIG.2

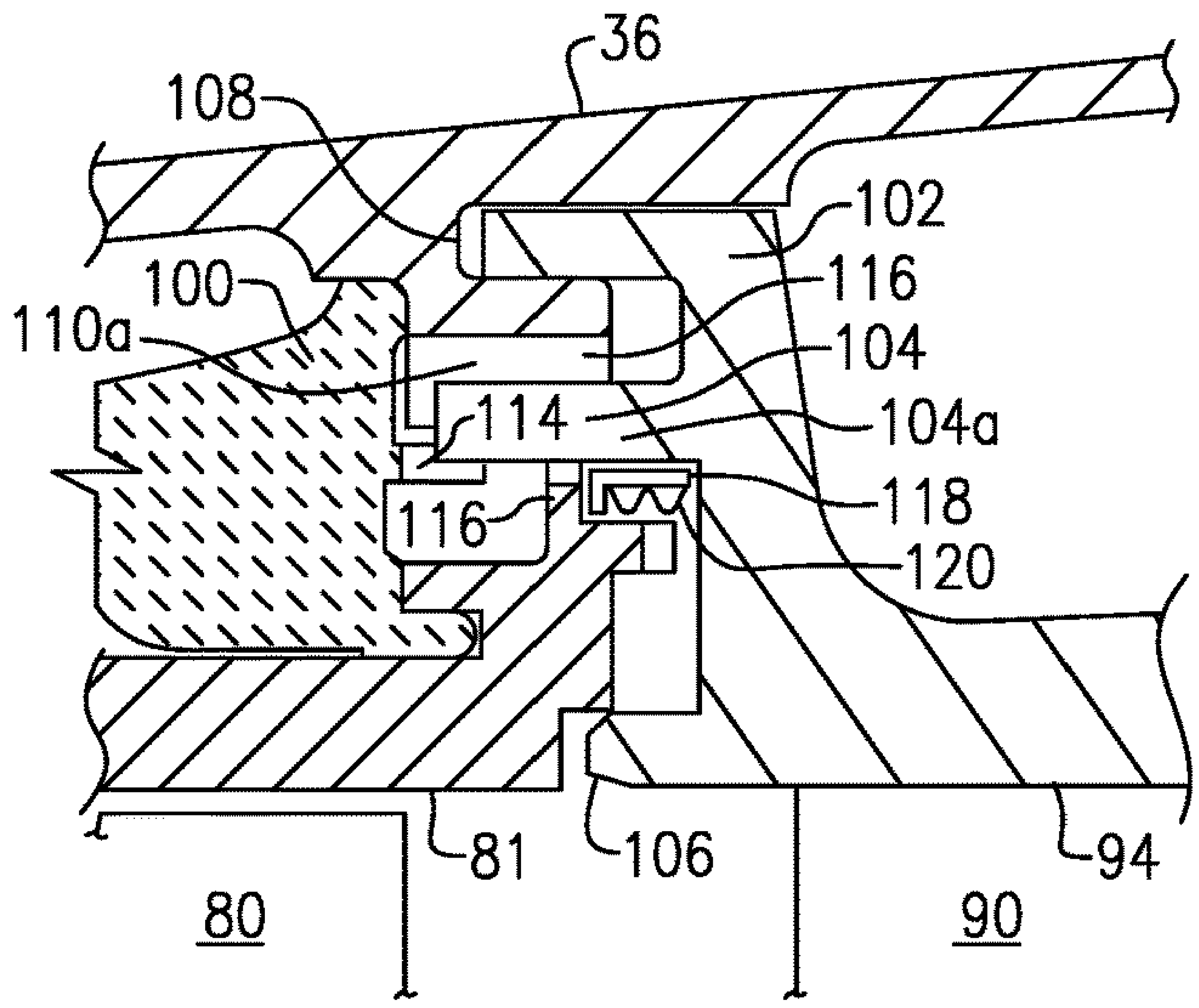

FIG.3

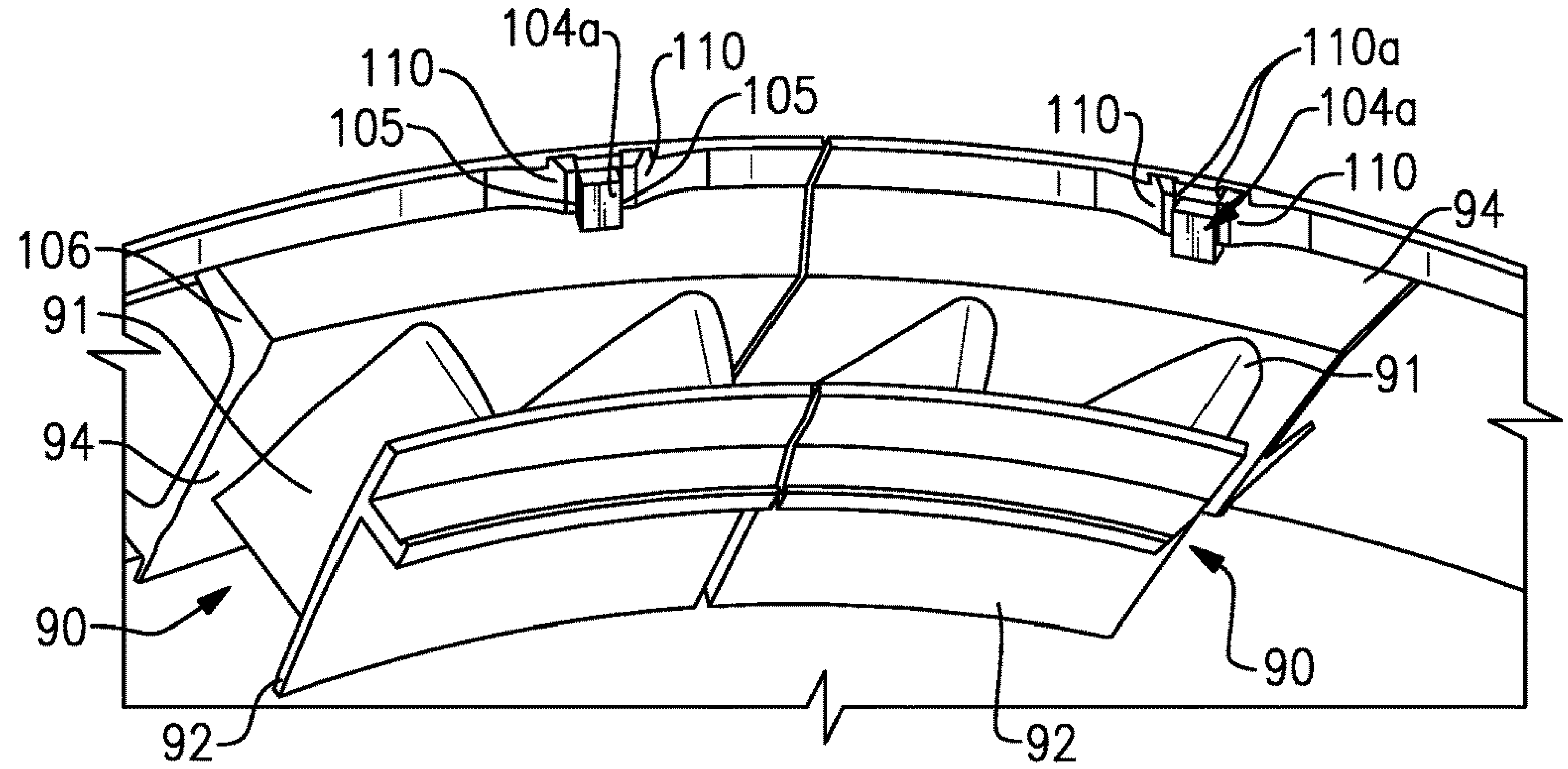

FIG.4

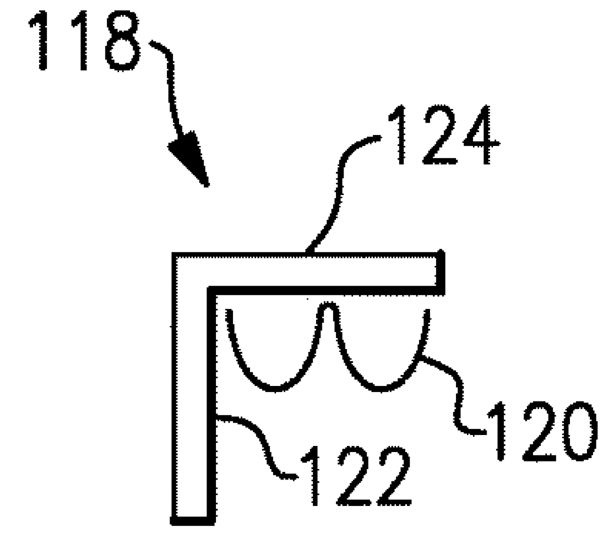

FIG.5

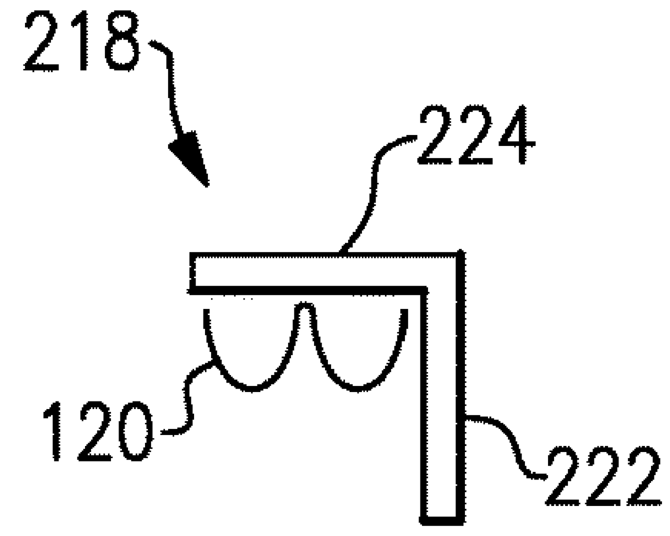

FIG.6

SEAL FOR A GAS TURBINE ENGINE

BACKGROUND

A gas turbine engine typically includes a fan section, a compressor section, a combustor section, and a turbine section. Air entering the compressor section is compressed and delivered into the combustion section where it is mixed with fuel and ignited to generate a high-speed exhaust gas flow. The high-speed exhaust gas flow expands through the turbine section to drive the compressor and the fan section.

Various components are attached to a static structure on the gas turbine engine, such as vanes, that must be prevented from rotating in a circumferential direction relative to the static structure. In order to prevent the circumferential rotation of the components, some form of engagement between the component and the static structure must be formed.

SUMMARY

In one exemplary embodiment, a gas turbine engine assembly includes a shield that has a first portion and a second portion. The first portion extends radially from an axial end portion of the shield and includes a blade outer air seal contact surface. The second portion extends axially from a radially outer end of the first portion and includes a vane contact surface.

In a further embodiment of the above, the shield forms a complete unitary circumferential hoop.

In a further embodiment of any of the above, the shield forms a circumferential hoop with a single discontinuity.

In a further embodiment of any of the above, a seal is in contact with the shield.

In a further embodiment of any of the above, the second portion of the shield is located radially outward from the seal. The first portion of the shield is located axially upstream from the seal.

In a further embodiment of any of the above, the seal includes a "W" shaped cross section pointing radially outward.

In a further embodiment of any of the above, the second portion of the shield is located radially outward from the seal and the first portion of the shield is located axially downstream from the seal.

In another exemplary embodiment, a gas turbine engine includes at least one vane. At least one blade outer air seal is adjacent at least one vane. A shield is located axially between the at least one vane and the at least one blade outer air seal. A seal is located radially inward from the shield.

In a further embodiment of any of the above, the shield comprises a first portion that extends radially on an axially upstream end of the shield.

In a further embodiment of any of the above, the shield comprises a first portion that extends radially on an axially downstream end of the shield.

In a further embodiment of any of the above, the shield comprises a second portion that extends axially from a radially outer end of the first portion.

In a further embodiment of any of the above, the second portion of the shield is located radially outward from the seal. The first portion of the shield is located axially upstream from the seal.

In a further embodiment of any of the above, at least one vane includes at least one anti-rotation tab

In a further embodiment of any of the above, at least one blade outer air seal includes a feature for engaging at least one anti-rotation tab.

In a further embodiment of any of the above, an engine static structure has a plurality anti-rotation protrusions and at least one anti-rotation tab engages one of the plurality anti-rotation protrusions.

In a further embodiment of any of the above, at least one blade outer air seal includes a feature for engaging the at least one of anti-rotation tab.

In a further embodiment of any of the above, the seal directly contacts the shield and includes a "W" shaped cross section.

In another exemplary embodiment, a method assembling a portion of a gas turbine engine includes positioning a shield in abutting contact with a blade outer air seal. A seal is positioned in abutting contact with the shield and the blade outer air seal. A vane is positioned in abutting contact with the shield.

In a further embodiment of any of the above, the shield is located axially between the vane and the blade outer air seal and the seal is located radially inward from the shield.

In a further embodiment of any of the above, the shield comprises a first portion that extends radially on an axially upstream end of the shield. A second portion extends axially from a radially outer end of the first portion.

BRIEF DESCRIPTION OF THE DRAWINGS

FIG. **1** is a schematic view of an example gas turbine engine.

FIG. **2** illustrates a schematic view of an example turbine section of the gas turbine engine.

FIG. **3** illustrates an enlarged view of the turbine section.

FIG. **4** illustrates a perspective view of a pair of vane doublets.

FIG. **5** illustrates a cross-sectional view of an example shield.

FIG. **6** illustrates a cross-sectional view of another example shield.

DETAILED DESCRIPTION

FIG. **1** schematically illustrates a gas turbine engine **20**. The gas turbine engine **20** is disclosed herein as a two-spool turbofan that generally incorporates a fan section **22**, a compressor section **24**, a combustor section **26** and a turbine section **28**. Alternative engines might include an augmentor section (not shown) among other systems or features. The fan section **22** drives air along a bypass flow path B in a bypass duct defined within a nacelle **15**, while the compressor section **24** drives air along a core flow path C for compression and communication into the combustor section **26** then expansion through the turbine section **28**. Although depicted as a two-spool turbofan gas turbine engine in the disclosed non-limiting embodiment, it should be understood that the concepts described herein are not limited to use with two-spool turbofans as the teachings may be applied to other types of turbine engines including three-spool architectures.

The exemplary engine **20** generally includes a low speed spool **30** and a high speed spool **32** mounted for rotation about an engine central longitudinal axis A relative to an engine static structure **36** via several bearing systems **38**. It should be understood that various bearing systems **38** at various locations may alternatively or additionally be provided, and the location of bearing systems **38** may be varied as appropriate to the application.

The low speed spool **30** generally includes an inner shaft **40** that interconnects a fan **42**, a first (or low) pressure compressor **44** and a first (or low) pressure turbine **46**. The inner shaft **40** is connected to the fan **42** through a speed change mechanism, which in exemplary gas turbine engine **20** is illustrated as a geared architecture **48** to drive the fan **42** at a lower speed than the low speed spool **30**. The high speed spool **32** includes an outer shaft **50** that interconnects a second (or high) pressure compressor **52** and a second (or high) pressure turbine **54**. A combustor **56** is arranged in exemplary gas turbine **20** between the high pressure compressor **52** and the high pressure turbine **54**. A mid-turbine frame **57** of the engine static structure **36** is arranged generally between the high pressure turbine **54** and the low pressure turbine **46**. The mid-turbine frame **57** further supports bearing systems **38** in the turbine section **28**. The inner shaft **40** and the outer shaft **50** are concentric and rotate via bearing systems **38** about the engine central longitudinal axis A which is collinear with their longitudinal axes.

The core airflow is compressed by the low pressure compressor **44** then the high pressure compressor **52**, mixed and burned with fuel in the combustor **56**, then expanded over the high pressure turbine **54** and low pressure turbine **46**. The mid-turbine frame **57** includes airfoils **59** which are in the core airflow path C. The turbines **46**, **54** rotationally drive the respective low speed spool **30** and high speed spool **32** in response to the expansion. It will be appreciated that each of the positions of the fan section **22**, compressor section **24**, combustor section **26**, turbine section **28**, and fan drive gear system **48** may be varied. For example, gear system **48** may be located aft of combustor section **26** or even aft of turbine section **28**, and fan section **22** may be positioned forward or aft of the location of gear system **48**.

The engine **20** in one example is a high-bypass geared aircraft engine. In a further example, the engine **20** bypass ratio is greater than about six (6), with an example embodiment being greater than about ten (10), the geared architecture **48** is an epicyclic gear train, such as a planetary gear system or other gear system, with a gear reduction ratio of greater than about 2.3 and the low pressure turbine **46** has a pressure ratio that is greater than about five. In one disclosed embodiment, the engine **20** bypass ratio is greater than about ten (10:1), the fan diameter is significantly larger than that of the low pressure compressor **44**, and the low pressure turbine **46** has a pressure ratio that is greater than about five 5:1. Low pressure turbine **46** pressure ratio is pressure measured prior to inlet of low pressure turbine **46** as related to the pressure at the outlet of the low pressure turbine **46** prior to an exhaust nozzle. The geared architecture **48** may be an epicycle gear train, such as a planetary gear system or other gear system, with a gear reduction ratio of greater than about 2.3:1. It should be understood, however, that the above parameters are only exemplary of one embodiment of a geared architecture engine and that the present invention is applicable to other gas turbine engines including direct drive turbofans.

A significant amount of thrust is provided by the bypass flow B due to the high bypass ratio. The fan section **22** of the engine **20** is designed for a particular flight condition—typically cruise at about 0.8 Mach and about 35,000 feet. The flight condition of 0.8 Mach and 35,000 ft (10,668 meters), with the engine at its best fuel consumption—also known as "bucket cruise Thrust Specific Fuel Consumption ('TSFC')"—is the industry standard parameter of 1 bm of fuel being burned divided by 1 bf of thrust the engine produces at that minimum point. "Low fan pressure ratio" is the pressure ratio across the fan blade alone, without a Fan Exit Guide Vane ("FEGV") system. The low fan pressure ratio as disclosed herein according to one non-limiting embodiment is less than about 1.45. "Low corrected fan tip speed" is the actual fan tip speed in ft/sec divided by an industry standard temperature correction of $[(\text{Tram } ° \text{ R})/(518.7° \text{ R})]^{0.5}$. The "Low corrected fan tip speed" as disclosed herein according to one non-limiting embodiment is less than about 1150 ft/second (350.5 meters/second).

FIG. **2** illustrates an enlarged schematic view of the high pressure turbine **54**, however, other sections of the gas turbine engine **20** could benefit from this disclosure. The high pressure turbine **54** includes a two-stage turbine section with a first rotor assembly **60** and a second rotor assembly **62**.

The first rotor assembly **60** includes a first array of rotor blades **64** circumferentially spaced around a first disk **68** and the second rotor assembly **62** includes a second array of rotor blades **66** circumferentially spaced around a second disk **70**. Each of the first and second array of rotor blades **64**, **66** include a respective first root portion **72** and a second root portion **74**, a first platform **76** and a second platform **78**, and a first airfoil **80** and a second airfoil **82**. Each of the first and second root portions **72**, **74** is received within a respective first rim and a second rim **84**, **86** of the first and second disk **68**, **70**. The first airfoil **80** and the second airfoil **82** extend radially outward toward a first and second blade outer air seal (BOAS) assembly **81**, **83**, respectively.

The first and second array of rotor blades **64**, **66** are disposed in the core flow path that is pressurized in the compressor section **24** then heated to a working temperature in the combustor section **26**. The first and second platforms **76**, **78** separate a gas path side inclusive of the first and second airfoils **80**, **82** and a non-gas path side inclusive of the first and second root portions **72**, **74**.

A shroud assembly **88** within the engine case structure **36** between the first rotor assembly **60** and the second rotor assembly **62** directs the hot gas core airflow in the core flow path from the first array of rotor blades **64** to the second array of rotor blades **66**. The shroud assembly **88** includes an array of vanes **90** that each include at least two airfoils **91** that extend between a respective inner vane platform **92** and an outer vane platform **94**. The outer vane platform **94** of the vane **90** may at least partially engage the first and second BOAS **81**, **83**.

FIG. **3** illustrates an enlarged view of the region surrounding a leading edge **106** of the outer vane platform **94** of the vane **90**. In the illustrated example, the outer vane platform **94** of the vane **90** includes a vane hook **102**, at least one anti-rotation tabs **104**, and the leading edge **106** adjacent the gas path. The vane hook **102** engages a recess **108** formed in the engine static structure **36** to limit radial and axially forward movement of the vane **90** relative to the engine static structure **36**.

The anti-rotation tabs **104** include a primary anti-rotation tab **104***a* (FIG. **4**) on the vane **90**. In the illustrated example, each vane **90** includes a pair of airfoils **91** extending between the outer vane platform **94** and inner vane platform **92**, and a primary anti-rotation tab **104***a*. The primary anti-rotation tab **104***a* includes a pair of circumferential faces **105** (FIG. **4**) that engage corresponding circumferential faces **110***a* on anti-rotation protrusions **110** that extend radially inward from the engine static structure **36** to prevent the vane **90** from rotating circumferentially during operation. The circumferential faces **110***a* on the anti-rotation protrusions **110** extend along the axis A. As shown in FIGS. **3** and **4**, the anti-rotation protrusions **110** engage a forward and radially outer portion of the anti-rotation tab **104**. Although the anti-rotation protrusions **110** are shown in pairs, only one anti-rotation protrusion **110** could be used to engage the primary anti-rotation tab **104***a*.

The engine static structure **36** only includes one of the pairs of anti-rotation protrusions **110** per vane **90** as shown in FIG. **4**. The anti-rotation tabs **104** extend from a radially outer portion of the outer vane platform **94** and are located radially inward from the vane hook **102**. The anti-rotation tabs **104** extend axially forward from the outer vane platform **94** and include a simple cross section, such as a square or rectangle, to simplify the manufacturing process.

A blade outer air seal support structure **100** and the blade outer air seal **81** also engage at least one of the anti-rotation tabs **104** to prevent circumferential movement of the blade outer air seal support structure **100** and the blade outer air seal **81**. As shown in FIG. **3**, the blade outer air seal support structure **100** includes a feature **114**, such as a pair of tabs (only one shown), that engages corresponding circumferential faces on the primary anti-rotation tab **104***a* to prevent rotation of the blade outer air seal support structure **100**. In the illustrated example, the features **114** engage a forward and radially inner portion of the primary anti-rotation tab **104***a*.

As shown in FIG. **3**, the BOAS **81** includes a feature **116**, such as a pair of tabs (only one shown), that engages corresponding circumferential faces on the primary anti-rotation tab **104***a* to prevent rotation of the BOAS **81**. In the illustrated example, the feature **116** engages a middle and a radially inner portion of the primary anti-rotation tab **104***a*.

A shield **118** is located adjacent a radially inner surface of the anti-rotation feature **104** and a downstream surface of the feature **116** on the BOAS **81**. The shield **118** forms a circumferential hoop that surrounds the axis A of the gas turbine engine **20**. In one example, the shield **118** forms a continuous hoop without any discontinuities and in another example the shield **118** includes a discontinuity forming a split in the hoop.

The shield **118** forms a continuous surface that engages a seal **120** located between the vane **90** and the BOAS **81**. In the illustrated example, the seal **120** is a "W" shaped shield pointing radially outward. The seal **120** reduces discrete contact points on the shield **118** where the shield **118** contacts the anti-rotation tab **104** and the feature **116**.

FIG. **5** illustrates a cross sectional view of the shield **118**. The shield **118** includes a first portion **122** extending radially on an axially upstream end of the shield **118** and a second portion **124** extending axially from a radially outer end of the first portion **122**. In one example, the shield **118** forms a complete unitary circumferential hoop that surrounds the axis A of the gas turbine engine **20**. In another example, the shield **118** forms a circumferential hoop with a single discontinuity. In yet another example, the shield **118** forms a circumferential hoop with at least two discontinuities.

In the illustrated example, the second portion **124** of the shield **118** is located radially outward from the seal **120** and the first portion **122** of the shield **118** is located axially upstream from the seal **120**.

FIG. **6** illustrates a cross-sectional view of another example shield **218**. The shield **218** is similar to the shield **118** except where described below or shown in the drawings. The shield **218** includes a first portion **222** extending radially on an axially downstream end of the shield **218** and a second portion **224** extending axially from a radially outer end of the first portion **222**. In one example, the shield **218** forms a complete unitary circumferential hoop that surrounds the axis A of the gas turbine engine **20**. In another example, the shield **218** forms a circumferential hoop with a single discontinuity. In yet another example, the shield **218** forms a circumferential hoop with at least two discontinuities.

In the illustrated example, the second portion **224** of the shield **218** is located radially outward from the seal **220** and the first portion **222** of the shield **218** is located axially downstream from the seal **120**.

The preceding description is exemplary rather than limiting in nature. Variations and modifications to the disclosed examples may become apparent to those skilled in the art that do not necessarily depart from the essence of this disclosure. The scope of legal protection given to this disclosure can only be determined by studying the following claims.

What is claimed is:

1. A gas turbine engine assembly comprising:

a shield having a first portion and a second portion, the first portion extending radially from an axial end portion of the shield and includes a blade outer air seal contact surface and the second portion extends axially from a radially outer end of the first portion and includes a vane contact surface, wherein the blade outer air seal contact surface faces in an axial direction and the vane contact surfaces faces in a radial direction.

2. The gas turbine engine assembly of claim **1**, wherein the shield forms a complete unitary circumferential hoop.

3. The gas turbine engine assembly of claim **1**, wherein the shield forms a circumferential hoop with a single discontinuity.

4. The gas turbine engine assembly of claim **1**, comprising a seal in contact with the shield.

5. The gas turbine engine assembly of claim **4**, wherein the second portion of the shield is located radially outward from the seal and the first portion of the shield is located axially upstream from the seal.

6. The gas turbine engine assembly of claim **4**, wherein the seal directly contacts a radially inner surface of the first portion of the shield and includes a "W" shaped cross section pointing radially outward.

7. The gas turbine engine assembly of claim **4**, wherein the second portion of the shield is located radially outward from the seal and the first portion of the shield is located axially downstream from the seal.

8. The gas turbine engine assembly of claim **1**, wherein the shield includes an "L" shaped cross section.

9. A gas turbine engine comprising:

at least one vane;

at least one blade outer air seal adjacent the at least one vane;

a shield located axially between the at least one vane and the at least one blade outer air seal and the shield includes a first portion extending axially from an axial end of the shield having a blade outer air seal contact surface facing in an axial direction and a second portion extending axially from a radially outer end of the first portion having a vane contact surface facing in a radial direction; and

a seal located radially inward from the shield.

10. The gas turbine engine of claim **9**, wherein the first portion extends in a radial direction from an axially upstream end of the shield.

11. The gas turbine engine of claim **9**, wherein the first portion extends in a radial direction from an axially downstream end of the shield.

12. The gas turbine engine of claim **9**, wherein the second portion of the shield is located radially outward from the seal and the first portion of the shield is located axially upstream from the seal.

13. The gas turbine engine of claim 9, wherein the at least one vane includes at least one anti-rotation tab.

14. The gas turbine engine of claim 13, wherein the at least one blade outer air seal includes a pair of tabs for engaging the at least one anti-rotation tab.

15. The gas turbine engine of claim 13, comprising an engine static structure with a plurality of anti-rotation protrusions and the at least one anti-rotation tab engages one of the plurality anti-rotation protrusions.

16. The gas turbine engine of claim 15, wherein the at least one blade outer air seal includes a pair of tabs for engaging the at least one of anti-rotation tab.

17. The gas turbine engine of claim 9, wherein the seal directly contacts a radially inner surface of the first portion of the shield and includes a "W" shaped cross section.

18. A method assembling a portion of a gas turbine engine comprising:

positioning a shield in abutting contact with a blade outer air seal, wherein the shield includes a first portion extending axially from an axial end of the shield having a blade outer air seal contact surface facing in an axial direction and a second portion extending axially from a radially outer end of the first portion having a vane contact surface facing in a radial direction;

positioning a seal in abutting contact with the shield and the blade outer air seal; and

positioning a vane in abutting contact with the shield.

19. The method of claim 18, wherein the shield is located axially between the vane and the blade outer air seal and the seal is located radially inward from the shield.

20. The method of claim 19, wherein the shield comprises a first portion extending radially on an axially upstream end of the shield and a second portion extending axially from a radially outer end of the first portion.

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