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(19) **United States**(12) **Patent Application Publication**
PARKER et al.(10) **Pub. No.: US 2017/0248155 A1**(43) **Pub. Date: Aug. 31, 2017**(54) **CENTRIFUGAL COMPRESSOR DIFFUSER
PASSAGE BOUNDARY LAYER CONTROL****F04D 27/00** (2006.01)**F04D 17/10** (2006.01)(71) Applicant: **GENERAL ELECTRIC COMPANY**,
Schenectady, NY (US)(52) **U.S. Cl.**CPC **F04D 29/44** (2013.01); **F04D 17/10**
(2013.01); **F04D 29/284** (2013.01); **F04D**
27/009 (2013.01); **F01D 25/12** (2013.01)(72) Inventors: **David Vickery PARKER**, Middleton,
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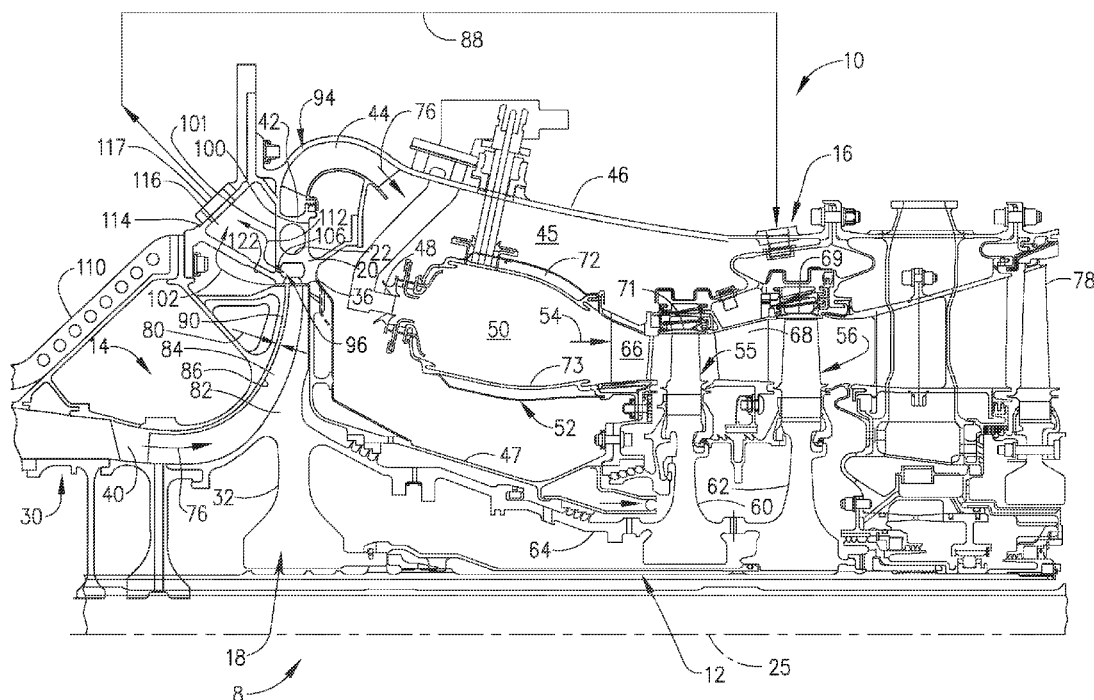
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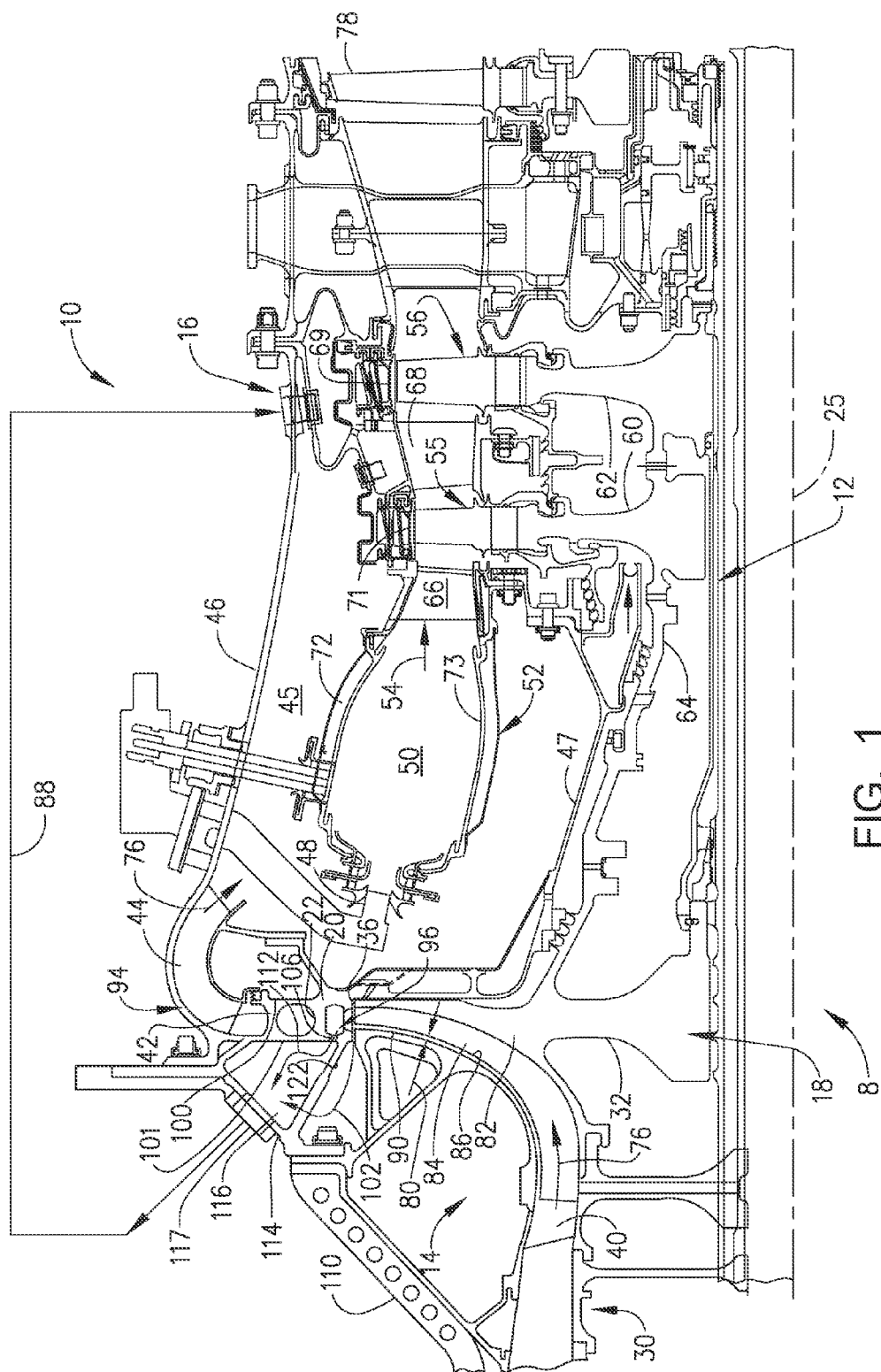
ABSTRACT

A centrifugal compressor diffuser (42) includes a plurality of diffuser flow passages (22) extending through an annular diffuser housing (20) and circumferentially bounded by diffuser vanes (23) and axially bounded by forward and aft walls (101, 100). A diffuser boundary layer bleed (96) for the passages may include boundary layer bleed apertures (106) or slots (130) disposed through the forward wall (101) and a downstream facing wall (142) canted at an acute cant angle to a downstream diffuser airflow direction (103) in the passages. Diffuser bleed flow (112) is bled from a diffuser boundary layer. Boundary layer bleed apertures can be located downstream of throat sections (28) of the flow passages near pressure sides of the vanes. A centrifugal compressor (18) may include the diffuser surrounding an annular centrifugal compressor impeller (32) and apparatus for flowing impeller bleed flow (102) from a radial clearance between an impeller tip (36) and a diffuser annular inlet (27) with diffuser bleed flow either mixed or separately to cool a turbine (16).

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7, 2014.**Publication Classification**(51) **Int. Cl.****F04D 29/44** (2006.01)**F04D 29/28** (2006.01)



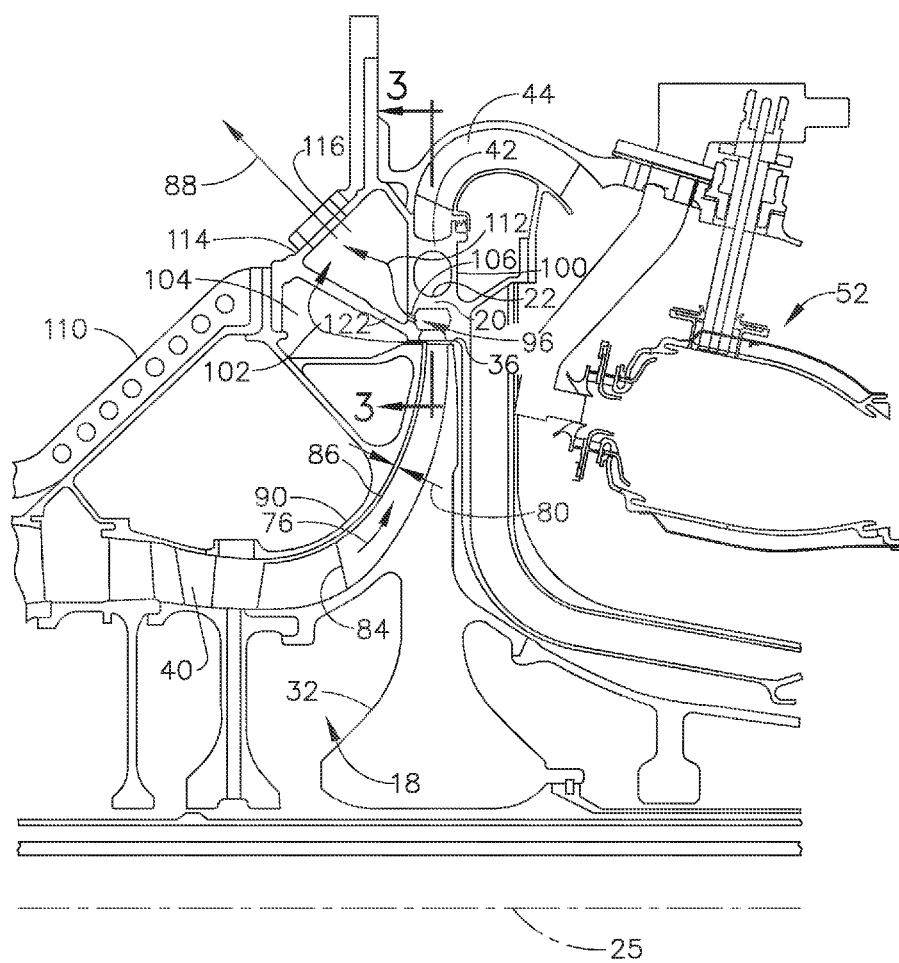


FIG. 2

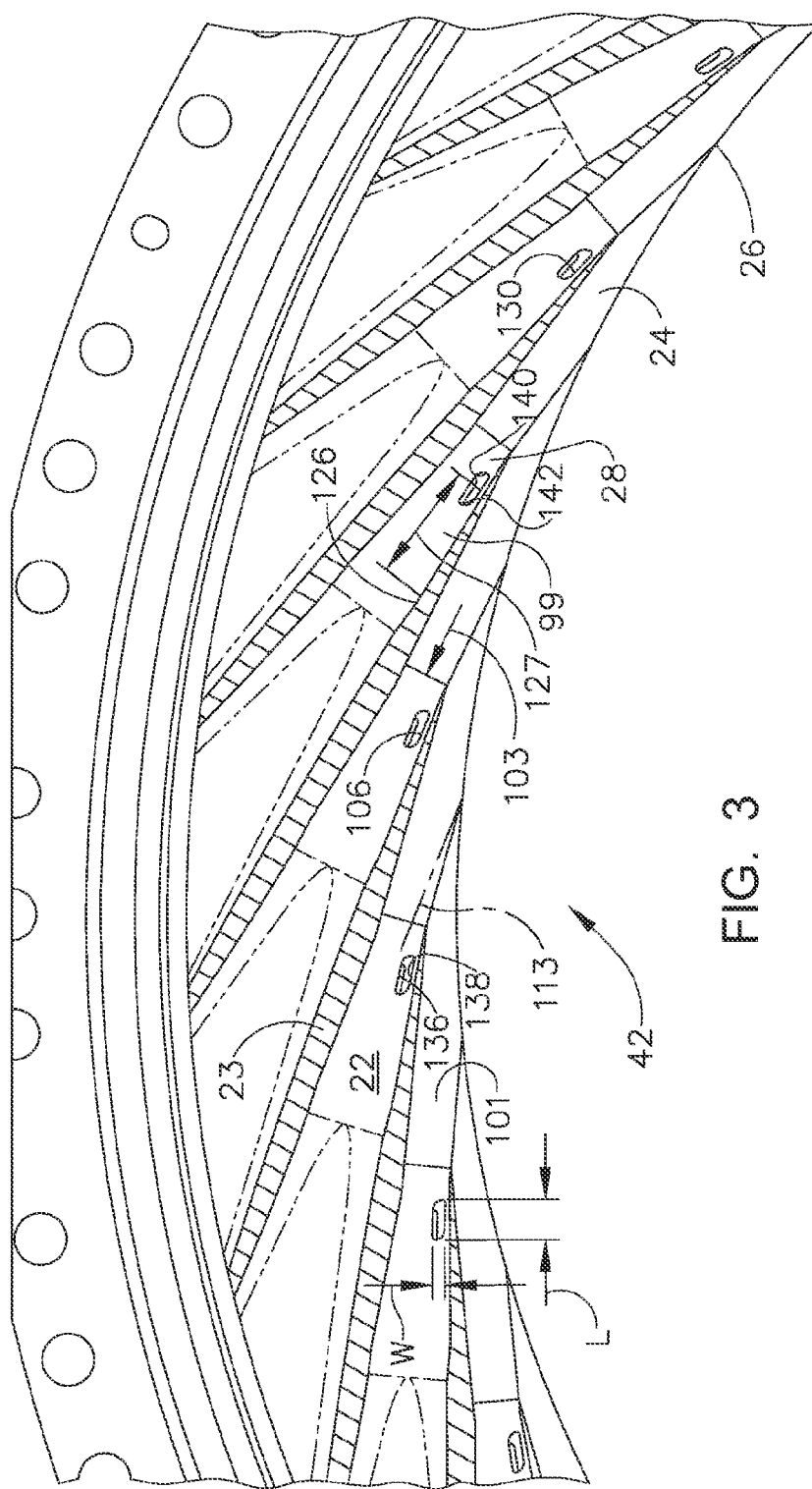
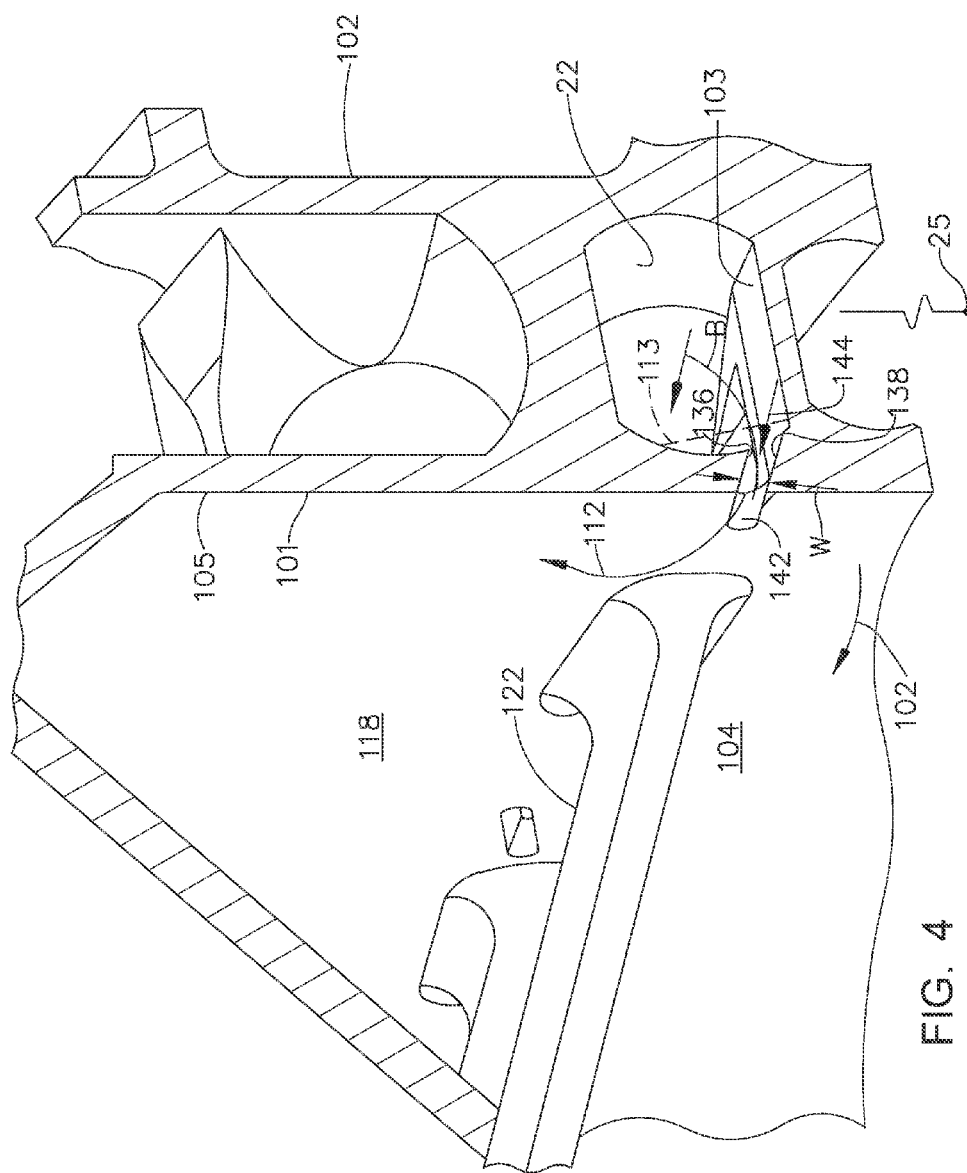
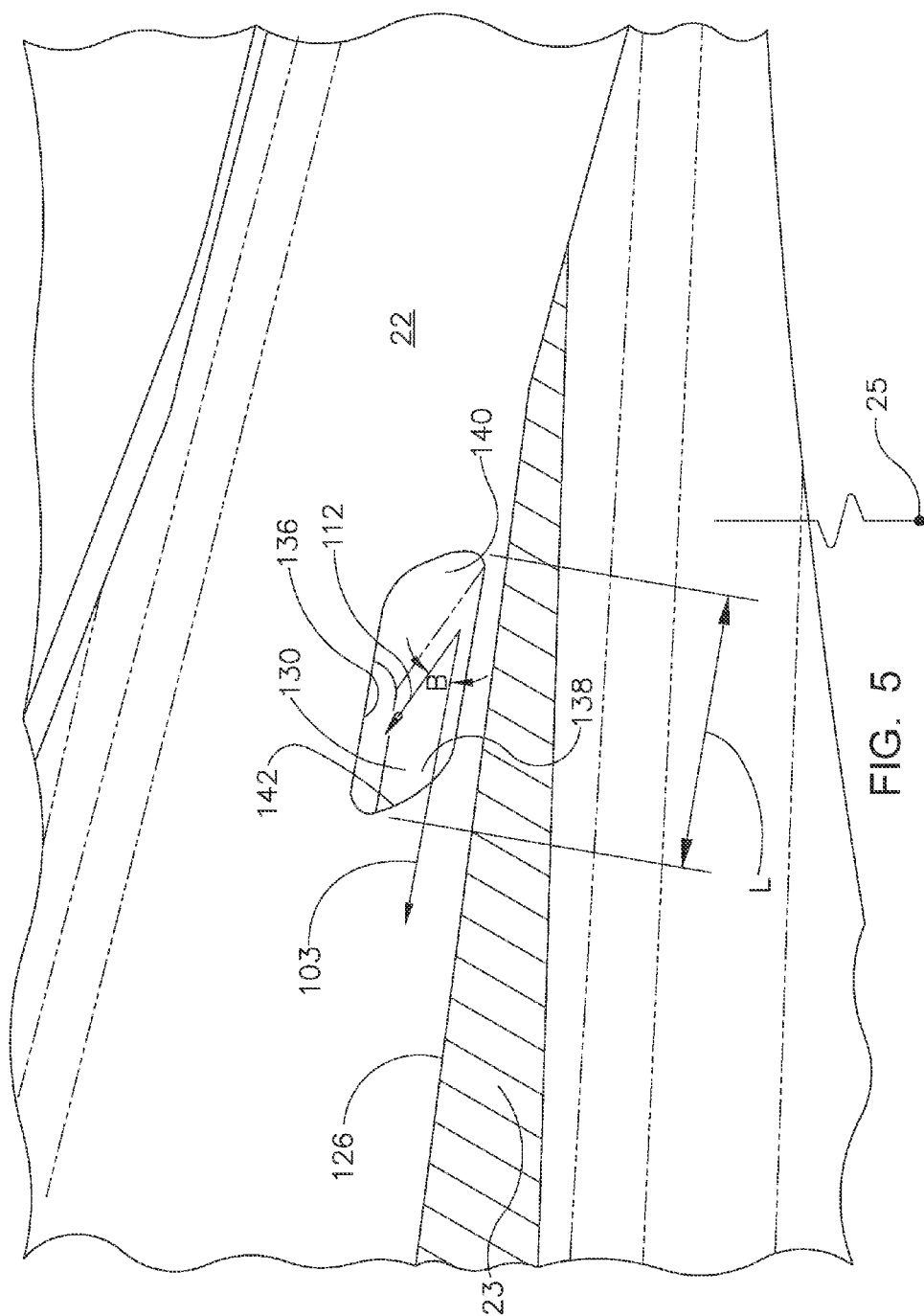


FIG. 3





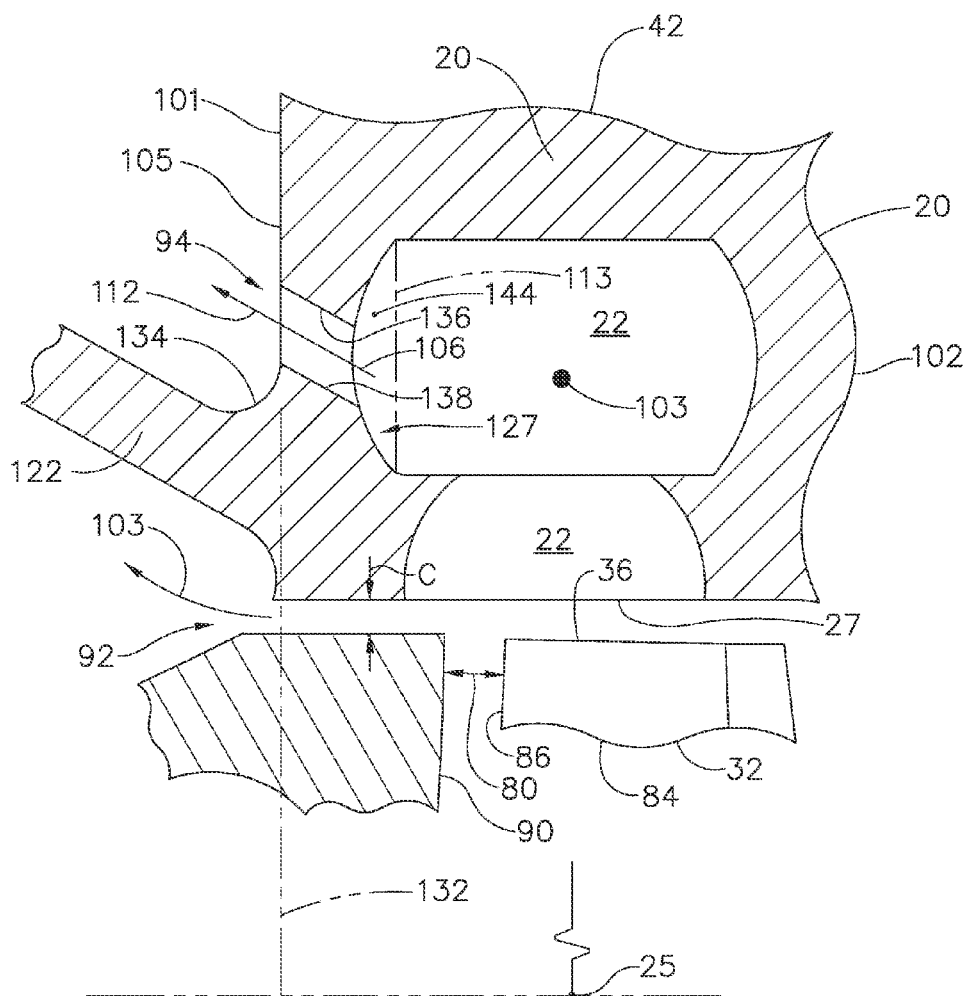


FIG. 6

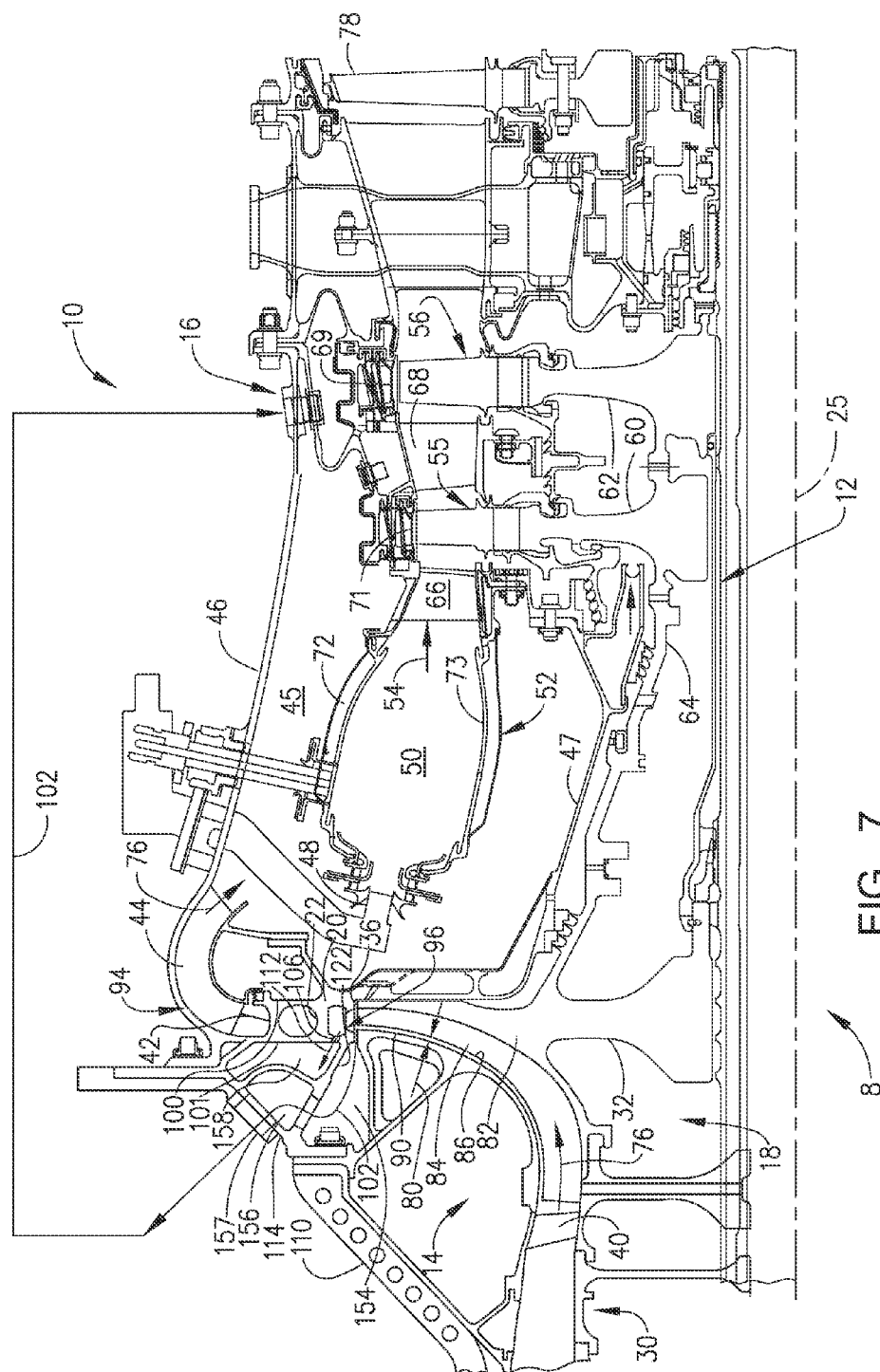
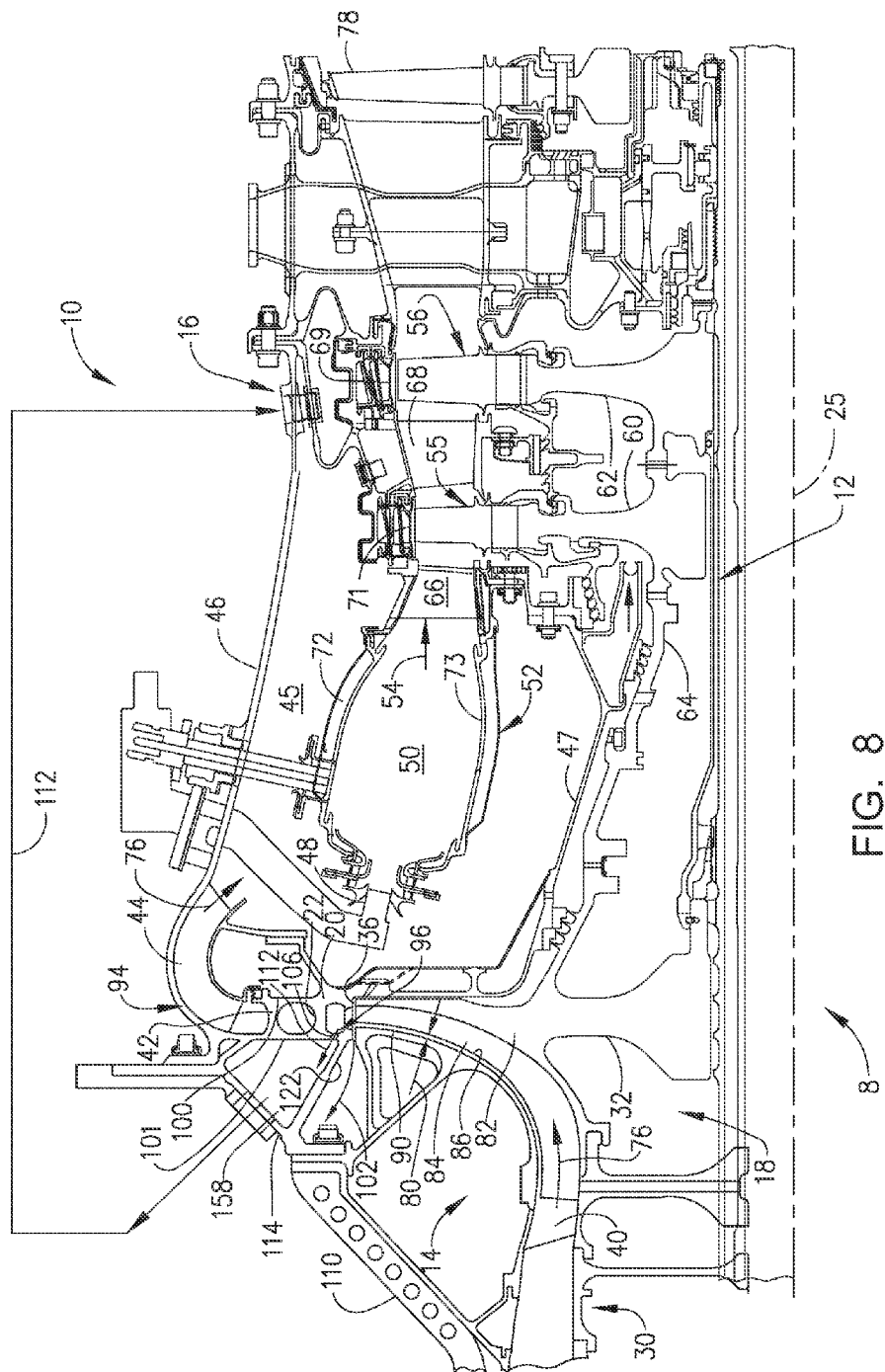
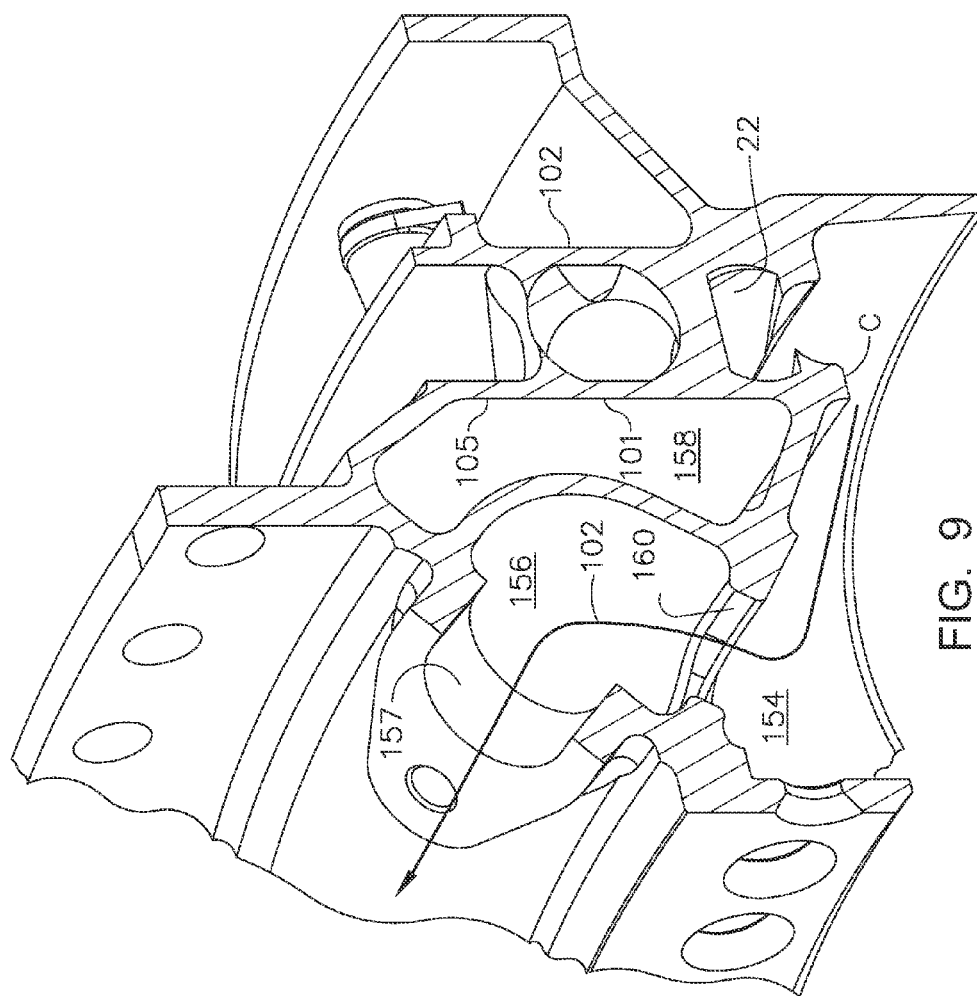


FIG. 7





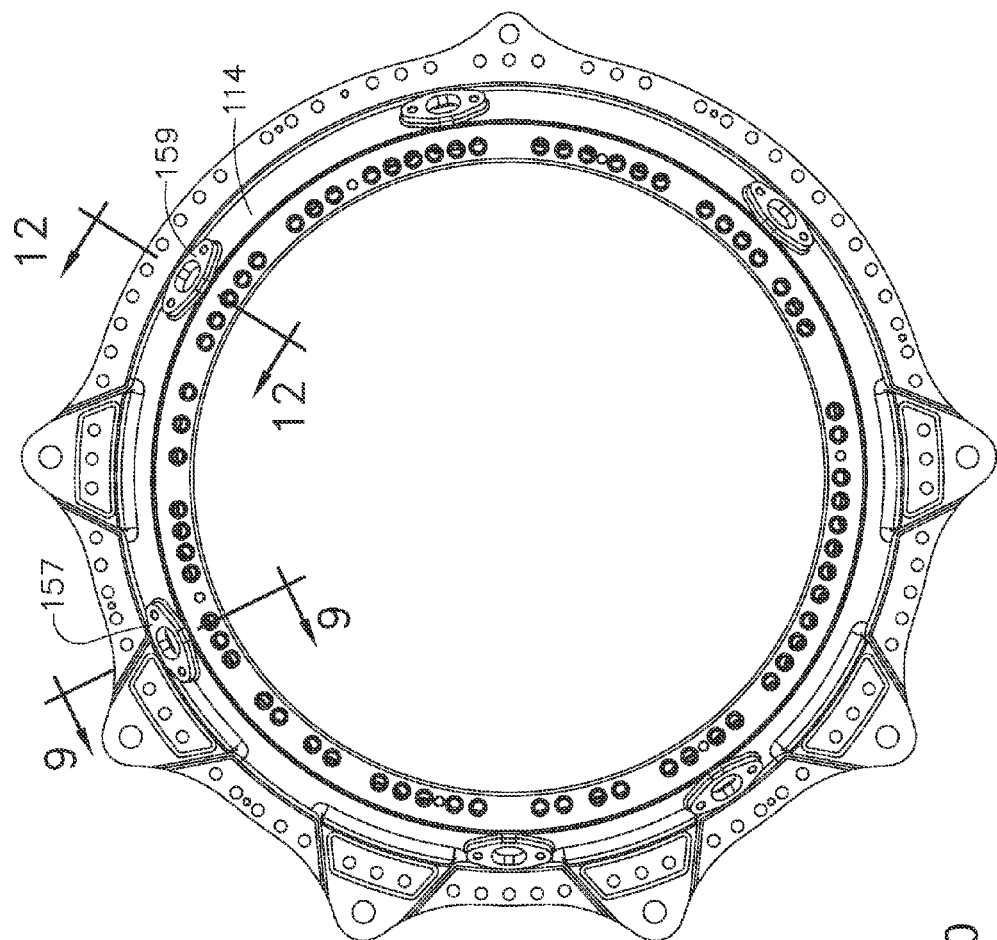


FIG. 10

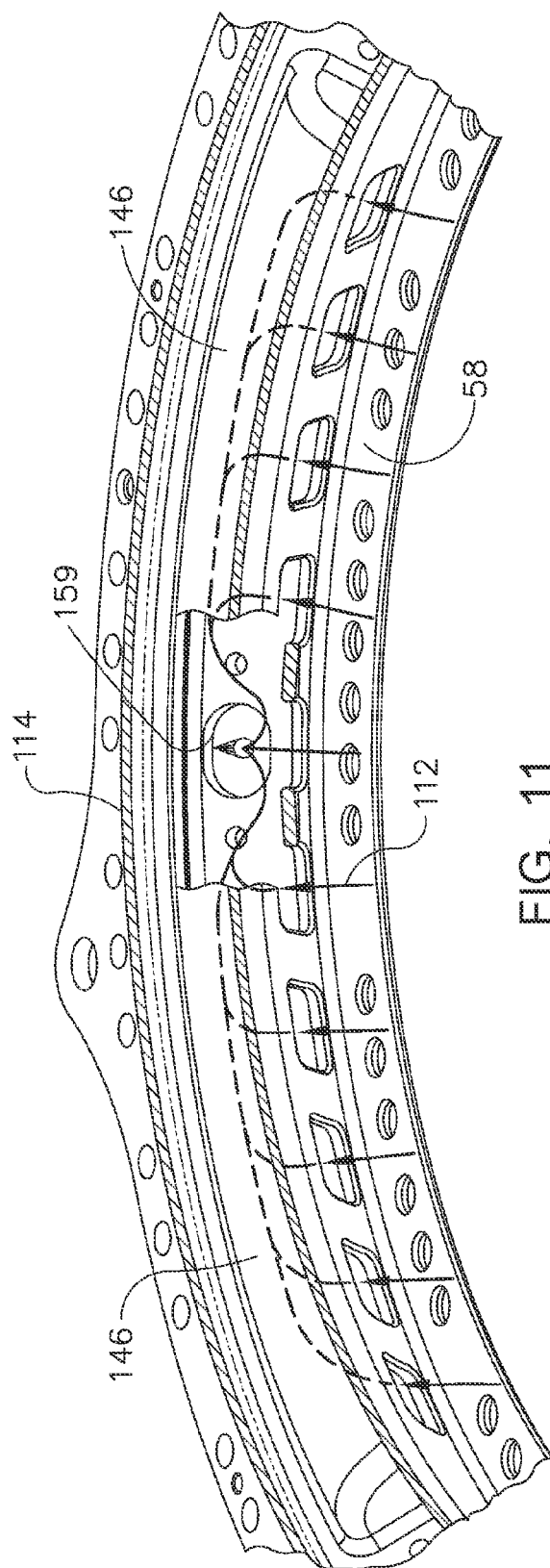
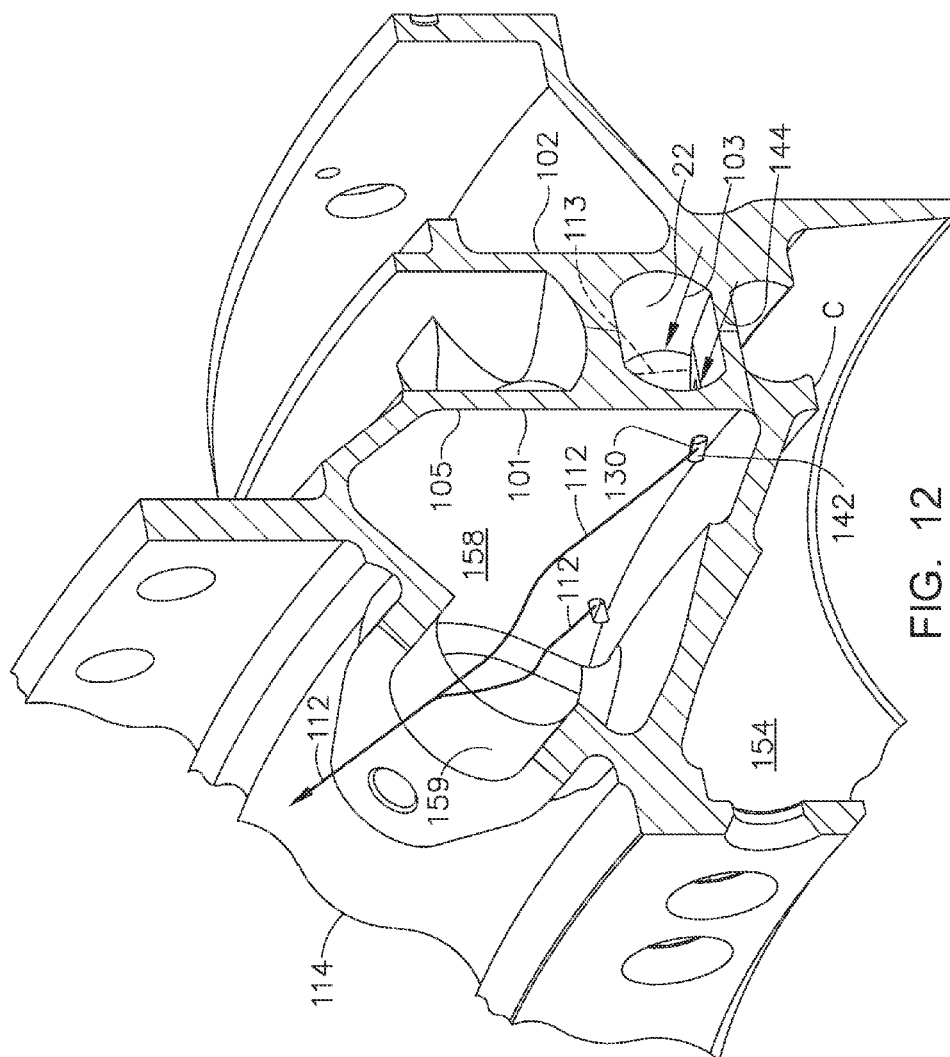


FIG. 11



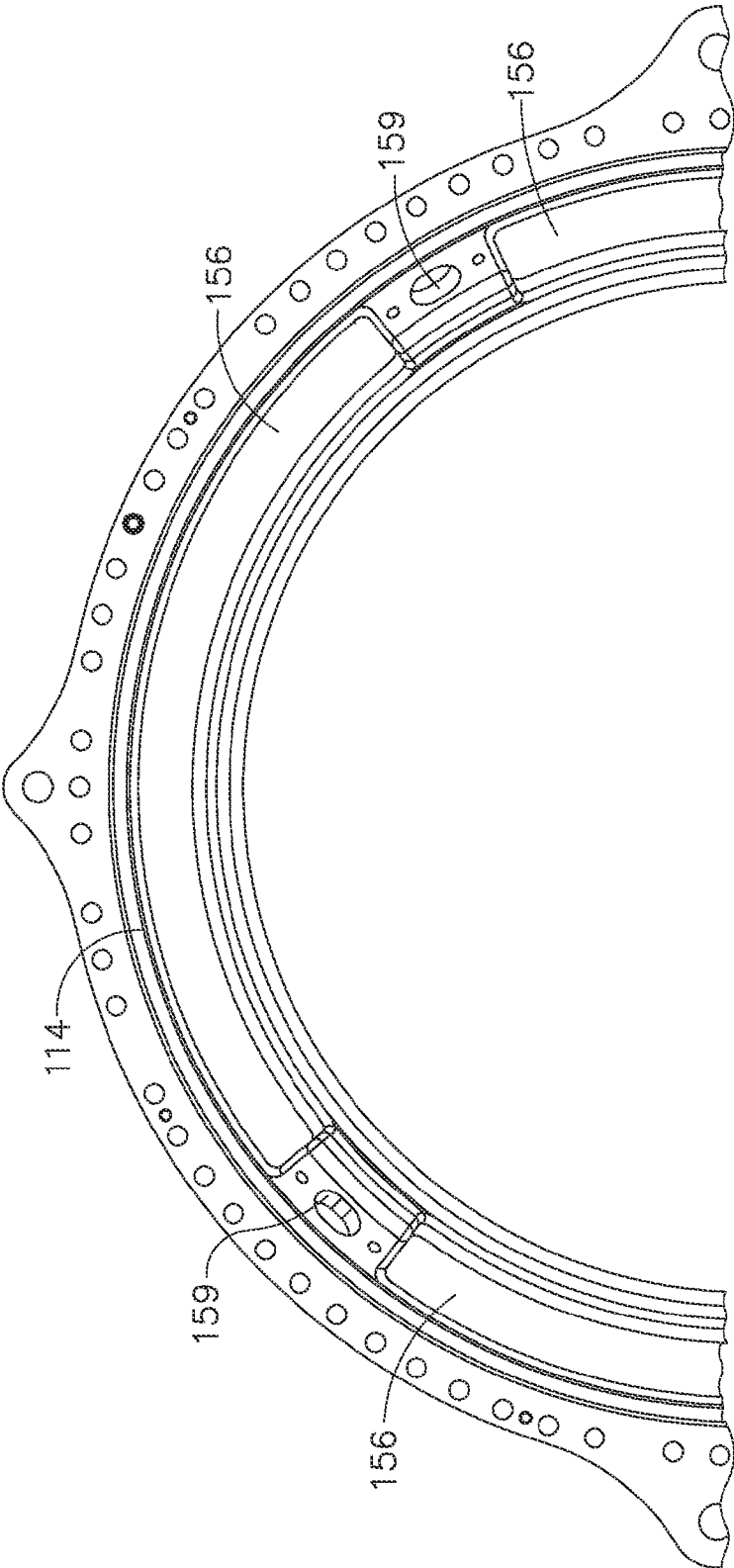


FIG. 13

CENTRIFUGAL COMPRESSOR DIFFUSER PASSAGE BOUNDARY LAYER CONTROL

GOVERNMENT INTERESTS

[0001] This invention was made with government support under government contract No. W911-W6-11-2-0009 by the Department of Defense. The government has certain rights to this invention.

BACKGROUND OF THE INVENTION

Technical Field

[0002] The present invention relates to bleed air from gas turbine engine centrifugal compressors.

[0003] One type of gas turbine engine includes a centrifugal compressor having a rotatable impeller to accelerate and, thereby, increase the kinetic energy of air flowing there-through. A diffuser is generally located immediately downstream of and surrounding the impeller. The diffuser operates to decrease the velocity of the air flow leaving the impeller and transform the energy thereof to an increase in static pressure, thus, pressurizing the air.

[0004] A conventional gas turbine engine typically includes a compressor, combustor, and turbine, both rotating turbine components such as blades, disks and retainers, and stationary turbine components, such as vanes, shrouds, and frames routinely require cooling due to heating thereof by hot combustion gases. Cooling of the turbine, especially the rotating components, is important to the proper function and safe operation of the engine. It is known to bleed cooling air from the centrifugal compressor to help cool the turbine.

[0005] Failure to adequately cool a turbine disk and its blading, for example, by providing cooling air deficient in supply pressure, volumetric flow rate or temperature margin, may be detrimental to the life and mechanical integrity of the turbine. Depending on the nature and extent of the cooling deficiency, the impact on engine operation may range from relatively benign blade tip distress, resulting in a reduction in engine power and useable blade life, to a rupture of a turbine disk, resulting in an unscheduled engine shutdown.

[0006] Balanced with the need to adequately cool the turbine is the desire for higher levels of engine operating efficiency which translate into lower fuel consumption and lower operating costs. Since turbine cooling air is typically drawn from one or more stages of the compressor and channelled by various means, such as pipes, ducts, and internal passageways to the desired components, such air is not available to be mixed with fuel, ignited in the combustor and undergo work extraction in the primary gas flowpath of the turbine.

[0007] Total cooling flow bled from the compressor is a loss in the engine operating cycle and it is desirable to keep such losses to a minimum.

[0008] Some conventional engines employ clean air bleed systems to cool turbine components in gas turbines using an axi-centrifugal compressor as is done in the General Electric CFE738 engine. The turbine cooling supply air exits the centrifugal diffuser through a small gap between the diffuser exit and deswirler inner shroud. Other turbine cooling air methods include extracting cooling from the impeller or from a gap between the impeller and the diffuser exit.

[0009] U.S. Pat. No. 5,555,721 to Bourneuf, et al, which issued on Sep. 17, 1996 and is entitled A Gas Turbine Engine

Cooling Supply Circuit@, discloses using bleed air from an impeller stage of a centrifugal compressor in a turbine cooling supply circuit for a gas turbine. U.S. Pat. No. 5,555,721 discloses impeller tip forward bleed flow and impeller tip aft bleed flow for cooling turbine components. U.S. Pat. No. 5,555,721 is assigned to the General Electric Co., the same assignee as this patent and is incorporated herein by reference.

[0010] U.S. Pat. No. 8,087,249 to Ottaviano, et al. which issued Jan. 3, 2012, and is entitled ATurbine Cooling Air From A Centrifugal Compressor@ discloses a gas turbine engine turbine cooling system including an impeller and a diffuser directly downstream of the impeller and a bleed for bleeding clean cooling air from downstream of the diffuser. U.S. Pat. No. 8,087,249 is assigned to the General Electric Co., the same assignee as this patent and is incorporated herein by reference.

[0011] Thus, there continues to be a demand for advancements in diffuser design and geometry that improves aerodynamic performance and reduces the overall engine radial envelope.

BRIEF DESCRIPTION OF THE INVENTION

[0012] A diffuser for a centrifugal compressor includes an annular diffuser housing, diffuser vanes axially extending between a forward wall and an aft wall of the diffuser housing, a plurality of diffuser flow passages extending through the housing and spaced about a circumference of the housing. The diffuser flow passages are bounded by the diffuser vanes and the forward and aft walls. A diffuser boundary layer bleed is provided for bleeding diffuser bleed flow from a diffuser boundary layer in each of the diffuser flow passages.

[0013] The diffuser boundary layer bleed may be configured for bleeding the diffuser bleed flow from the diffuser boundary layer at a position located in a region of flow weakness in each of the diffuser flow passages.

[0014] The diffuser boundary layer bleed may include boundary layer bleed apertures disposed through the forward wall. Each of the boundary layer bleed apertures may be a slot including a downstream facing wall angled or canted at an acute cant angle with respect to a downstream diffuser airflow direction in each of the diffuser flow passages respectively.

[0015] The boundary layer bleed apertures may be positioned or located downstream of throat sections of the diffuser flow passages near pressure sides of the diffuser vanes.

[0016] A centrifugal compressor including an annular centrifugal compressor impeller, a diffuser annularly surrounding the impeller, and a plurality of diffuser flow passages extending through a housing of the diffuser and spaced about a circumference of the housing. Each of the passages includes a throat section and a diffusing section downstream of the throat section. The diffuser flow passages are circumferentially bounded by diffuser vanes extending axially between forward and aft walls of the diffuser and a diffuser boundary layer bleed is provided for bleeding diffuser bleed flow from a diffuser boundary layer in each of the diffuser flow passages.

[0017] The centrifugal compressor may also include a radial clearance between an impeller tip of the impeller and an annular inlet of the diffuser, a means for mixing impeller bleed flow from the radial clearance with diffuser bleed flow

from the boundary layer bleed apertures for providing turbine cooling air, and a means for flowing the turbine cooling air to a turbine or a means for flowing impeller bleed flow and the diffuser bleed flow separately to the turbine.

BRIEF DESCRIPTION OF THE DRAWINGS

[0018] FIG. 1 is a sectional view illustration of a gas turbine engine with a centrifugal compressor for mixing impeller tip bleed flow and diffuser bleed flow in the compressor section before using the flows for cooling turbine components.

[0019] FIG. 2 is an enlarged sectional view illustration of the centrifugal compressor and a diffuser with diffuser bleed holes illustrated in FIG. 1.

[0020] FIG. 3 is an aft looking forward perspective view illustration of the diffuser and the diffuser bleed holes through 3-3 in FIG. 2.

[0021] FIG. 4 is an enlarged perspective view illustration of the bleed holes illustrated in FIG. 3.

[0022] FIG. 5 is a perspective view illustration of a portion of the diffuser and the diffuser bleed holes illustrated in FIG. 2.

[0023] FIG. 6 is an enlarged sectional view illustration of the centrifugal compressor tip and the diffuser bleed holes illustrated in FIG. 2.

[0024] FIG. 7 is a sectional view illustration of a gas turbine engine centrifugal compressor with an alternative arrangement for separately flowing impeller tip bleed for cooling turbine components.

[0025] FIG. 8 is a sectional view illustration of the gas turbine engine illustrated in FIG. 7 with an arrangement for separately flowing diffuser bleed flow for cooling turbine components.

[0026] FIG. 9 is an enlarged perspective view illustration of one of the impeller bleed flow ports illustrated in FIG. 7 and as taken through 9-9 in FIG. 10.

[0027] FIG. 10 is a forward looking aft perspective view illustration of an aft casing surrounding the centrifugal compressor and including the impeller and bleed flow ports illustrated in FIGS. 7 and 8 respectively.

[0028] FIG. 11 is cutaway perspective view illustration of impeller bleed flowpaths for one of the impeller bleed flow ports illustrated in FIGS. 7 and 9.

[0029] FIG. 12 is an enlarged perspective view illustration of one of the diffuser bleed flow ports illustrated in FIG. 8 and as taken through 12-12 in FIG. 10.

[0030] FIG. 13 is cutaway perspective view illustration of a diffuser bleed flowpath through one of the diffuser bleed flow ports illustrated in FIG. 8 and as taken through 12-12 in FIG. 10.

DETAILED DESCRIPTION OF THE INVENTION

[0031] Illustrated in FIG. 1 is a gas turbine engine high pressure centrifugal compressor 18 in a high pressure gas generator 10 of a gas turbine engine 8. The high pressure centrifugal compressor 18 is a final compressor stage of a high pressure compressor 4. The high pressure gas generator 10 has a high pressure rotor 12 including, in downstream serial or flow relationship, the high pressure compressor 14, a combustor 52, and a high pressure turbine 16. The rotor 12 is rotatably supported about an engine axis 25 by bearings in engine frames not illustrated herein.

[0032] The exemplary embodiment of the high pressure compressor 14 illustrated herein includes a five stage axial compressor 30 followed by the centrifugal compressor 18 having an annular centrifugal compressor impeller 32. Outlet guide vanes 40 are disposed between the five stage axial compressor 30 and the single stage centrifugal compressor 18. Compressor discharge pressure (CDP) air 76 exits the impeller 32 and passes through a diffuser 42 annularly surrounding the impeller 32 and then through a deswirl cascade 44 into a combustion chamber 45 within the combustor 52. The combustion chamber 45 is surrounded by annular radially outer and inner combustor casings 46, 47. Air 76 is conventionally mixed with fuel provided by a plurality of fuel nozzles 48 and ignited and combusted in an annular combustion zone 50 bounded by annular radially outer and inner combustion liners 72, 73.

[0033] The combustion produces hot combustion gases 54 which flow through the high pressure turbine 16 causing rotation of the high pressure rotor 12 and continue downstream for further work extraction in a low pressure turbine 78 and final exhaust as is conventionally known. In the exemplary embodiment depicted herein, the high pressure turbine 16 includes, in downstream serial flow relationship, first and second high pressure turbine stages 55, 56 having first and second stage disks 60, 62. A high pressure shaft 64 of the high pressure rotor 12 connects the high pressure turbine 16 in rotational driving engagement to the impeller 32. A first stage nozzle 66 is directly upstream of the first high pressure turbine stage 55 and a second stage nozzle 68 is directly upstream of the second high pressure turbine stage.

[0034] Referring to FIG. 1, the compressor discharge pressure (CDP) air 76 is discharged from the impeller 32 of the centrifugal compressor 18, used to combust fuel in the combustor 52, and to cool components of turbine 16 subjected to the hot combustion gases 54; such as, the first stage nozzle 66, first and second stage shrouds 71, 69 surrounding the first and second high pressure turbine stages 55, 56 respectively. The high pressure compressor 14 includes a compressor aft casing 110 and a diffuser forward casing 114 as more fully illustrated in FIGS. 1 and 2. The compressor aft casing 110 generally surrounds the axial compressor 30 and the diffuser forward casing 114 generally surrounds the centrifugal compressor 18 and supports the diffuser 42 directly downstream of the centrifugal compressor 18. The compressor discharge pressure (CDP) air 76 is discharged from the impeller 32 of the centrifugal compressor 18 directly into the diffuser 42.

[0035] Referring to FIGS. 2 and 3, the impeller 32 includes a plurality of centrifugal compressor blades 84 radially extending from a rotor disc portion 82. Opposite and axially forward of the compressor blades 84 is an annular blade tip shroud 90. The shroud 90 is adjacent to blade tips 86 of the compressor blades 84 defining a blade tip clearance 80 therebetween. The diffuser 42 includes an annular diffuser housing 20 having a plurality of tangentially disposed diffuser flow passages 22 extending radially therethrough, spaced about a circumference 26 of the housing 20, and through which diffuser airflow 103 flows in a downstream direction. Diffuser vanes 23 axially extend between a forward wall 101 and the aft wall 100 of the diffuser 42.

[0036] Referring to FIGS. 2 and 3, the diffuser vanes 23 circumferentially extend between adjacent ones of the diffuser flow passages 22. The diffuser flow passages 22 are

partly defined and circumferentially bounded by the circumferentially spaced apart diffuser vanes **23**. Adjacent ones of the passages **22** intersect with each other at radially inner inlet sections **24** of the passages **22** that define a quasi-vaneless annular inlet **27** of the diffuser **42**. Each passage **22** further includes a throat section **28** downstream of and integral with the inner inlet section **24**. Each passage **22** further includes a diffusing section **99** immediately downstream of the throat section **28**.

[0037] Referring to FIGS. **2** and **6**, a centrifugal compressor first cooling air source **92** for turbine cooling air **88** is a small predetermined radial clearance (C) located between an impeller tip **36** of the rotating impeller **32** and the annular inlet **27** of the static diffuser **42**. Impeller bleed flow **102** from the radial clearance (C) is collected in a radially inner manifold **104**. The predetermined radial clearance (C) is designed to accommodate thermal and mechanical growth of the impeller **32** and is open to or in fluid communication with the radially inner manifold **104**.

[0038] Referring to FIGS. **3-6**, we have found that the diffuser airflow **103** on one side of the passage (such as passage **22**) in multi-passage diffusers (such as the diffuser **42**) that follow or are downstream of centrifugal impellers (such as the impeller **32**) is often weak and may be subject to separation. Separation in the passage can generate high losses that lowers engine specific fuel consumption (SFC). This area or region of weak flow **127** is also believed to be a contributor to surge that limits flow range of the compressor.

[0039] A centrifugal compressor stage second cooling air source **94** for turbine cooling air **88** includes a diffuser boundary layer bleed **96** for bleeding diffuser bleed flow **112** from a diffuser boundary layer **113** in each of the diffuser flow passages **22** of the diffuser **42**, illustrated herein as plurality of boundary layer bleed apertures **106**. The diffuser boundary layer bleed **96**, also referred to as fluidic bleed, helps reduce the weak flow and limit or prevent the unwanted flow separation. The diffuser boundary layer bleed **96** bleeds diffuser bleed flow **112** from the diffuser boundary layer **113** into a radially outer manifold **116**.

[0040] The radially inner and outer manifolds **104**, **116** are in fluid communication such that the impeller bleed flow **102** from the radially inner manifold **104** flows into the radially outer manifold **116**. The impeller and diffuser bleed flows **102**, **112** are mixed in the radially outer manifold **116** to provide the turbine cooling air **88** which is then ported or otherwise flowed from radially outer manifold **116** through a plurality of circumferentially distributed manifold ports **117** to the high pressure turbine **16**. The turbine cooling air **88** may be channelled or flowed therefrom by external piping (not shown) to cool the first and second stage shrouds **71**, **69** (illustrated in FIG. **1**).

[0041] Substantially axially extending beams or struts **122** separate the radially inner and outer manifolds **104**, **116** and the impeller bleed flow **102** passes between the struts **122** as it flows from the radially inner manifold **104** into the radially outer manifold **116**. The fluidic bleed flow illustrated herein as the diffuser boundary layer bleed **96** represents a small amount of flow, less than 1% of the engine core flow. The fluidic bleed is strategically removed near the inception of the weak flow to improve the overall performance of the diffuser.

[0042] Referring to FIGS. **3-5**, the boundary layer bleed apertures **106** may be holes or slots **130** through the forward

wall **101** of the diffuser **42** as illustrated herein. The boundary layer bleed apertures **106** or slots **130** lead into and are in flow communication with the radially outer manifold **116**. The slot **130** is positioned or located downstream of the throat section **28** near a pressure side **126** of the diffuser vane **23** at a position where the flow would begin to show weakness or instability in a diffuser without the diffuser boundary layer bleed **96**. This position is located in what is referred to as a region of flow weakness **127**. A slot width W may be sized with manufacturing constraints such as a minimum tool size. A slot length L may be selected to enable up to 3% of the engine core flow to be used.

[0043] The slot **130** should ideally be angled such that the diffuser bleed flow **112** exits the slot perpendicular to a forward surface **105** of the forward wall **101** of the diffuser **42** in a radial plane **132** passing through the engine centerline or axis **25** as illustrated in FIG. **5**. However, because of constraints such as the slot extending through or very near a bend **134** in the forward wall **101** of the diffuser **42** this angle may be different. The slot **130** has radially outer and inner walls **136**, **138**, as illustrated in FIG. **6**, and upstream and downstream facing walls **140**, **142**, as illustrated in FIGS. **4** and **5** respectively, extending through the forward wall **101**. The downstream facing wall **142** is designed to scoop boundary layer air **144** in the diffuser boundary layer **113** only. Thus, the downstream facing wall **142** is angled or canted at an acute cant angle B of less than 90 degrees with respect to the diffuser airflow **103** (parallel to the direction boundary layer air **144** in the downstream direction in the diffuser flow passages **22** of the diffuser **42**). It appears that an acute cant angle B of 45 degrees is desirable. However, the acute cant angle B is limited by geometry and manufacturing constraints on the outside of the diffuser so that an acute cant angle, for example about 22.5 degrees, is more practical.

[0044] Illustrated in FIGS. **7-13** is a gas turbine engine with a centrifugal compressor similar to the one illustrated in FIGS. **1-3** but with an alternative arrangement or design for separately gathering and flowing the impeller tip bleed and diffuser bleed flow for cooling turbine components. The impeller bleed flow **102** from the radial clearance (C), illustrated in FIG. **9**, is flowed into and collected in a radially inner annular manifold **154** illustrated in FIGS. **7** and **9**. Inter-manifold apertures **160** are disposed between the inner annular manifold **154** and a plurality of radially outer annular manifolds **156** illustrated in FIGS. **7**, **9**, and **13**. The inter-manifold apertures **160** allow the impeller bleed flow **102** to flow from the inner annular manifold **154** into the outer annular manifolds **156**. The impeller bleed flow **102** from the outer annular manifolds **156** is then ported or otherwise flowed through a plurality of circumferentially distributed impeller bleed flow manifold ports **157**, illustrated in FIG. **10**, to the high pressure turbine **16** for turbine cooling.

[0045] Referring to FIGS. **8**, **10**, and **11-13**, the diffuser boundary layer bleed **96** bleeds diffuser bleed flow **112** from the diffuser boundary layer **113** into an annular diffuser bleed manifold **158** from where the diffuser bleed flow **112** is then ported or otherwise flowed through a plurality of circumferentially distributed diffuser bleed manifold ports **159** to the high pressure turbine **16** for turbine cooling. FIG. **10** illustrates the relative circumferential and axial locations of

the impeller bleed flow manifold ports **157** and the diffuser bleed manifold ports **159** on and through the diffuser forward casing **114**.

[0046] While there have been described herein what are considered to be preferred and exemplary embodiments of the present invention, other modifications of the invention shall be apparent to those skilled in the art from the teachings herein and, it is therefore, desired to be secured in the appended claims all such modifications as fall within the true spirit and scope of the invention. Accordingly, what is desired to be secured by Letters Patent of the United States is the invention as defined and differentiated in the following claims.

What is claimed:

1. A gas turbine engine centrifugal compressor diffuser comprising:

- an annular diffuser housing,
- diffuser vanes axially extending between a forward wall and an aft wall of the diffuser housing,
- a plurality of diffuser flow passages extending through the housing and spaced about a circumference of the housing,
- the diffuser flow passages bounded by the diffuser vanes and the forward and aft walls, and
- a diffuser boundary layer bleed for bleeding diffuser bleed flow from a diffuser boundary layer in each of the diffuser flow passages.

2. The diffuser according to claim **1** further comprising the diffuser boundary layer bleed configured for bleeding the diffuser bleed flow from the diffuser boundary layer at a position located in a region of flow weakness in each of the diffuser flow passages.

3. The diffuser according to claim **1** further comprising the diffuser boundary layer bleed including boundary layer bleed apertures disposed through the forward wall.

4. The diffuser according to claim **3** further comprising each of the boundary layer bleed apertures being a slot including a downstream facing wall angled or canted at an acute cant angle with respect to a downstream diffuser airflow direction in each of the diffuser flow passages respectively.

5. The diffuser according to claim **3** further comprising the boundary layer bleed apertures positioned or located downstream of throat sections of the diffuser flow passages near pressure sides of the diffuser vanes.

6. The diffuser according to claim **5** further comprising each of the boundary layer bleed apertures being a slot including a downstream facing wall angled or canted at an acute cant angle with respect to a downstream diffuser airflow direction in each of the diffuser flow passages respectively.

7. A gas turbine engine centrifugal compressor comprising:

- an annular centrifugal compressor impeller,
- a diffuser annularly surrounding the impeller,
- a plurality of diffuser flow passages extending through a housing of the diffuser and spaced about a circumference of the housing,
- each of the passages including a throat section and a diffusing section downstream of the throat section,
- the diffuser flow passages circumferentially bounded by diffuser vanes extending axially between forward and aft walls of the diffuser, and

a diffuser boundary layer bleed for bleeding diffuser bleed flow from a diffuser boundary layer in each of the diffuser flow passages.

8. The centrifugal compressor according to claim **7** further comprising the diffuser boundary layer bleed configured for bleeding the diffuser bleed flow from the diffuser boundary layer at a position located in a region of flow weakness in each of the diffuser flow passages.

9. The diffuser according to claim **7** further comprising the diffuser boundary layer bleed including boundary layer bleed apertures disposed through the forward wall.

10. The centrifugal compressor according to claim **9** further comprising each of the boundary layer bleed apertures being a slot including a downstream facing wall angled or canted at an acute cant angle with respect to a downstream diffuser airflow direction in each of the diffuser flow passages respectively.

11. The centrifugal compressor according to claim **10** further comprising the boundary layer bleed apertures positioned or located downstream of throat sections of the diffuser flow passages near pressure sides of the diffuser vanes.

12. The centrifugal compressor according to claim **11** further comprising each of the boundary layer bleed apertures being a slot including a downstream facing wall angled or canted at an acute cant angle with respect to a downstream diffuser airflow direction in each of the diffuser flow passages respectively.

13. The centrifugal compressor according to claim **9** further comprising:

- a radial clearance between an impeller tip of the impeller and an annular inlet of the diffuser,
- a means for mixing impeller bleed flow from the radial clearance with the diffuser bleed flow from the boundary layer bleed apertures for providing turbine cooling air and flowing the turbine cooling air to a turbine, or
- a means for flowing the impeller bleed flow and the diffuser bleed flow separately to the turbine.

14. The centrifugal compressor according to claim **13** further comprising each of the boundary layer bleed apertures being a slot including a downstream facing wall angled or canted at an acute cant angle with respect to a downstream diffuser airflow direction in each of the diffuser flow passages respectively.

15. The centrifugal compressor according to claim **13** further comprising the boundary layer bleed apertures positioned or located downstream of throat sections of the diffuser flow passages near pressure sides of the diffuser vanes.

16. The centrifugal compressor according to claim **15** further comprising each of the boundary layer bleed apertures being a slot including a downstream facing wall angled or canted at an acute cant angle with respect to a downstream diffuser airflow direction in each of the diffuser flow passages respectively.

17. The centrifugal compressor according to claim **9** further comprising:

- a radial clearance between an impeller tip of the impeller and an annular inlet of the diffuser,
- the radial clearance in fluid communication with a radially inner manifold,
- the boundary layer bleed apertures in flow communication with a radially outer manifold,

the radially inner manifold in fluid communication with the radially outer manifold such that the impeller bleed flow flows into the radially outer manifold and mixes with the diffuser bleed flow to form turbine cooling air, and

means for flowing turbine cooling air out of the radially outer manifold.

18. The centrifugal compressor according to claim **17** further comprising each of the boundary layer bleed apertures being a slot including a downstream facing wall angled or canted at an acute cant angle with respect to a downstream diffuser airflow direction in each of the diffuser flow passages respectively.

19. The centrifugal compressor according to claim **18** further comprising the boundary layer bleed apertures positioned or located downstream of throat sections of the diffuser flow passages near pressure sides of the diffuser vanes.

20. The centrifugal compressor according to claim **19** further comprising each of the boundary layer bleed apertures being a slot including a downstream facing wall angled or canted at an acute cant angle with respect to a downstream diffuser airflow direction in each of the diffuser flow passages respectively.

21. The centrifugal compressor according to claim **9** further comprising:

a radial clearance between an impeller tip of the impeller and an annular inlet of the diffuser,

the radial clearance in fluid communication with a radially inner annular manifold,

inter-manifold apertures disposed between the inner annular manifold and a plurality of radially outer annular manifolds,

a means for porting and flowing the impeller bleed flow from the radial clearance through a plurality of circumferentially distributed impeller bleed flow manifold ports in and through an diffuser forward casing surrounding the centrifugal compressor to the high pressure turbine for turbine cooling,

the diffuser boundary layer bleed in fluid flow communication with and operable for bleeding the diffuser bleed flow into an annular diffuser bleed manifold, and

a means for porting and flowing the diffuser bleed flow through a plurality of circumferentially distributed diffuser bleed manifold ports in and through the diffuser forward casing to the high pressure turbine for turbine cooling.

22. The centrifugal compressor according to claim **21** further comprising each of the boundary layer bleed apertures being a slot including a downstream facing wall angled or canted at an acute cant angle with respect to a downstream diffuser airflow direction in each of the diffuser flow passages respectively.

23. The centrifugal compressor according to claim **22** further comprising the boundary layer bleed apertures positioned or located downstream of throat sections of the diffuser flow passages near pressure sides of the diffuser vanes.

24. The centrifugal compressor according to claim **23** further comprising each of the boundary layer bleed apertures being a slot including a downstream facing wall angled or canted at an acute cant angle with respect to a downstream diffuser airflow direction in each of the diffuser flow passages respectively.

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