(19) Europäisches Patentamt
European Patent Office
Office européen des brevets

(11) EP 2 855 884 B1

(12) EUROPEAN PATENT SPECIFICATION

(45) Date of publication and mention of the grant of the patent:
14.08.2019 Bulletin 2019/33

(21) Application number: **13797827.6**

(22) Date of filing: **15.05.2013**

(51) Int Cl.:
F02C 7/12 *(2006.01)* ***F01D 25/08*** *(2006.01)*
F01D 5/08 *(2006.01)* ***F01D 25/12*** *(2006.01)*

(86) International application number:
PCT/US2013/041127

(87) International publication number:
WO 2013/180954 (05.12.2013 Gazette 2013/49)

(54) **HIGH PRESSURE TURBINE COOLANT SUPPLY SYSTEM**

KÜHLMITTELVERSORGUNGSSYSTEM FÜR HOCHDRUCKTURBINE

SYSTÈME D’ALIMENTATION EN RÉFRIGÉRANT POUR TURBINE À HAUTE PRESSION

(84) Designated Contracting States:
AL AT BE BG CH CY CZ DE DK EE ES FI FR GB GR HR HU IE IS IT LI LT LU LV MC MK MT NL NO PL PT RO RS SE SI SK SM TR

(30) Priority: **31.05.2012 US 201213485579**

(43) Date of publication of application:
08.04.2015 Bulletin 2015/15

(73) Proprietor: **United Technologies Corporation**
Farmington, CT 06032 (US)

(72) Inventors:
- **MOSLEY, John H.**
  Portland, Connecticut 06480 (US)
- **ALVANOS, Ioannis**
  West Springfield, Massachusetts 01089 (US)
- **STRIPINIS, Philip S.**
  Rocky Hill, Connecticut 06067 (US)
- **FREIBERG, Douglas Paul**
  Glastonbury, Connecticut 06073 (US)
- **PINERO, Hector M.**
  Middletown, Connecticut 06457 (US)
- **O’CONNOR, John J.**
  South Windsor, Connecticut 06074 (US)
- **PIETROBON, Jon**
  Longueuil, Québec J4G 1A1 (CA)

(74) Representative: **Dehns**
St. Bride’s House
10 Salisbury Square
London EC4Y 8JD (GB)

(56) References cited:

EP-A1- 1 533 473	**EP-A1- 1 533 473**
GB-A- 2 420 155	**US-A- 4 217 755**
US-A1- 2002 067 987	**US-A1- 2002 067 987**
US-B1- 6 331 097	**US-B2- 6 916 151**
US-B2- 6 916 151	**US-B2- 7 192 245**

EP 2 855 884 B1

Description

BACKGROUND

[0001] Gas turbine engines operate by passing a volume of high energy gases through a plurality of stages of vanes and blades, each having an airfoil, in order to drive turbines to produce rotational shaft power. The shaft power is used to drive a compressor to provide compressed air to a combustion process to generate the high energy gases. Additionally, the shaft power is used to drive a generator for producing electricity, or to drive a fan for producing high momentum gases for producing thrust. In order to produce gases having sufficient energy to drive the compressor, generator and fan, it is necessary to combust the fuel at elevated temperatures and to compress the air to elevated pressures, which also increases its temperature. Thus, the vanes and blades are subjected to extremely high temperatures, often times exceeding the melting point of the alloys comprising the airfoils. High pressure turbine blades are subject to particularly high temperatures.

[0002] In order to maintain gas turbine engine turbine blades at temperatures below their melting point, it is necessary to, among other things, cool the blades with a supply of relatively cooler air, typically bled from the high pressure compressor. The cooling air is directed into the blade to provide impingement and film cooling. For example, cooling air is passed into interior cooling channels of the airfoil to remove heat from the alloy, and subsequently discharged through cooling holes to pass over the outer surface of the airfoil to prevent the hot gases from contacting the vane or blade directly. Various cooling air channels and hole patterns have been developed to ensure sufficient cooling of various portions of the turbine blade.

[0003] A typical turbine blade is connected at its inner diameter ends to a rotor, which is connected to a shaft that rotates within the engine as the blades interact with the gas flow. The rotor typically comprises a disk having a plurality of axial retention slots that receive mating root portions of the blades to prevent radial dislodgment. The siphoned compressor bleed air is typically routed from the compressor to the turbine blade retention slots for routing into the interior cooling channels of the airfoil. As such, the bleed air must pass through rotating and non-rotating components between the high pressure compressor and high pressure turbine. For example, cooling air is often drawn from the radial outer ends of the high pressure compressor vanes and routed radially inward through a support strut to the high pressure shaft before being directed radially outward for flow across the turbine rotor and into the turbine blade roots. Routing of the cooling air in such a manner incurs aerodynamic losses that reduce the cooling effectiveness of the air and overall gas turbine engine efficiency. Additionally, the bleed air must also pass through high pressure zones within the engine that exceed pressures needed to cool the turbine blades. There is, therefore, a continuing need to improve aerodynamic efficiencies in routing cooling fluid within cooling systems of gas turbine engines.

[0004] US 4,217,755 relates to a cooling air control valve. GB 2,420,155 relates to cooling of turbine blades in a gas turbine engine.

SUMMARY

[0005] The present invention is directed toward a turbine stage for use in a gas turbine engine configured to rotate in a circumferential direction about an axis extending through a center of the gas turbine engine. The turbine stage comprises a disk, a plurality of blades and a mini-disk. The disk comprises an outer diameter edge having slots, an inner diameter bore surrounding the axis, a forward face, and an aft face. The plurality of blades is coupled to the slots. The mini-disk is coupled to the aft face of the rotor to define a cooling plenum therebetween in order to direct cooling air to the slots. In one embodiment of the invention, the cooling plenum is connected to a radially inner compressor bleed air inlet through all rotating components so that cooling air passes against the inner diameter bore.

BRIEF DESCRIPTION OF THE DRAWINGS

[0006]

FIG. 1 shows a gas turbine engine including a high pressure compressor section and a high pressure turbine section having the coolant supply system of the present invention.

FIG. 2 is a schematic view of the high pressure turbine section of FIG. 1 showing a first stage rotor with a forward-mounted mini-disk and a second stage rotor with an aft-mounted mini-disk.

FIG. 3 is a schematic view of the high pressure compressor section of FIG. 1 showing a bleed system having a radially inward-mounted inlet for directing cooling air into a rotating shaft system.

DETAILED DESCRIPTION

[0007] FIG. 1 shows gas turbine engine 10, in which the coolant supply system of the present invention can be used. Gas turbine engine 10 comprises a dual-spool turbofan engine having fan 12, low pressure compressor (LPC) 14, high pressure compressor (HPC) 16, combustor section 18, high pressure turbine (HPT) 20 and low pressure turbine (LPT) 22, which are each concentrically disposed around longitudinal engine centerline CL. Fan 12 is enclosed at its outer diameter within fan case 23A. Likewise, the other engine components are correspondingly enclosed at their outer diameters within various engine casings, including LPC case 23B, HPC case 23C, HPT case 23D and LPT case 23E such that an air flow path is formed around centerline CL. Although depicted as a dual-spool turbofan engine in the disclosed non-limiting embodiment, it should be understood that the concepts described herein are not limited to use with turbofans as the teachings may be applied to other types of turbine engines, such as three-spool turbine engines and geared fan turbine engines.

[0008] Inlet air A enters engine 10 and it is divided into streams of primary air A_P and secondary air A_S after it passes through fan 12. Fan 12 is rotated by low pressure turbine 22 through shaft 24 to accelerate secondary air A_S (also known as bypass air) through exit guide vanes 26, thereby producing a major portion of the thrust output of engine 10. Shaft 24 is supported within engine 10 at ball bearing 25A, roller bearing 25B and roller bearing 25C. Low pressure compressor (LPC) 14 is also driven by shaft 24. Primary air Ap (also known as gas path air) is directed first into LPC 14 and then into high pressure compressor (HPC) 16. LPC 14 and HPC 16 work together to incrementally step-up the pressure of primary air A_P. HPC 16 is rotated by HPT 20 through shaft 28 to provide compressed air to combustor section 18. Shaft 28 is supported within engine 10 at ball bearing 25D and roller bearing 25E. The compressed air is delivered to combustors 18A and 18B, along with fuel through injectors 30A and 30B, such that a combustion process can be carried out to produce the high energy gases necessary to turn turbines 20 and 22, as is known in the art. Primary air A_P continues through gas turbine engine 10 whereby it is typically passed through an exhaust nozzle to further produce thrust.

[0009] HPT 20 and LPT 22 each include a circumferential array of blades extending radially from rotors 34A and 34B connected to shafts 28 and 24, respectively. Similarly, HPT 20 and LPT 22 each include a circumferential array of vanes extending radially from HPT case 23D and LPT case 23E, respectively. In this specific example, HPT 20 comprises a two-stage turbine, which includes inlet guide vanes 29 having blades 32A and 32B extending from rotor disks 34A and 34B of rotor 34, and vanes 35, which extend radially inward from case HPT case 23E between blades 32A and 32B. Blades 32A and 32B include internal channels or passages into which compressed cooling air A_C air from, for example, HPC 16 is directed to provide cooling relative to the hot combustion gasses of primary air A_P. Blades 32B include internal passages into which compressed cooling air A_C from, for example, HPC 16 is routed to provide cooling relative to the hot combustion gasses of primary air Ap.

[0010] Cooling air A_C is directed radially inward to the interior of HPC 16 between adjacent rotor disks, as shown in FIG. 3. From HPC 16, cooling air A_C is directed along shaft 28 within a tie shaft arrangement (FIG. 3) and passed through inner diameter bores of disks 34A and 34B. Finally, as shown in FIG. 1, cooling air A_C is directed radially outward along the aft face of disk 34B and into blades 32B. Blades 32A are provided with cooling air through a separate coolant circuit that is isolated from the flow of cooling air A_C. As such, cooling air A_C can be tailored to the needs of blades 32B. Cooling air A_C can also be used to control the temperature of disk 34B. Furthermore, cooling air A_C is completely contained within rotating components so that dynamic losses are avoided.

[0011] FIG. 2 shows a schematic view of high pressure turbine, or high pressure turbine section, 20 of gas turbine engine 10 in FIG. 1 having inlet guide vane 29, first stage turbine blade 32A, second stage vane 35 and second stage turbine blade 32B disposed within engine case 23D. Inlet guide vane 29 comprises an airfoil that is suspended from turbine case 23D at its outer diameter end. Turbine blade 32A comprises airfoil 40, which extends radially outward from platform 42. Airfoil 40 and platform 42 are coupled to rotor disk 34A through interaction of rim slot 43 with root 44. Second stage vane 35 comprises an airfoil that is suspended from turbine case 23D at its outer diameter end. Turbine blade 32B comprises airfoil 46, which extends radially outward from platform 48. Airfoil 46 and platform 48 are coupled to rotor disk 34B through interaction of rim slot 49 with root 50.

[0012] First stage rotor disk 34A includes forward mini-disk 52A and aft seal plate 54A. Second stage rotor disk 34B includes aft mini-disk 52B and forward seal plate 54B. First stage rotor disk 34A is joined to second stage rotor disk 34B at coupling 56 to define inter-stage cavity 58. Forward mini-disk 52A seals against inlet guide vane 29 and root 44, and directs cooling air (not shown) into rim slot 43. Aft seal plate 54A prevents escape of the cooling air into cavity 58. Aft mini-disk 52B seals against root 50, and directs cooling air A_C into rim slot 49. Forward seal plate 54B prevents escape of cooling air A_C into cavity 58. Aft seal plate 54A and forward seal plate 54B also seal against second stage vane 35 to prevent primary air A_P from entering cavity 58.

[0013] Airfoil 40 and airfoil 46 extend from their respective inner diameter platforms toward engine case 23D, across gas path 60. Hot combustion gases of primary air Ap are generated within combustor 18 (FIG. 1) upstream of high pressure turbine 20 and flow through gas path 60. Inlet guide vane 29 straightens the flow of primary air A_P to improve incidence on airfoil 40 of turbine blade 32A. As such, airfoil 40 is better able to extract energy from primary air Ap. Likewise, second stage vane 35 straightens the flow of primary air Ap from airfoil 40 to improve incidence on airfoil 46. Primary air Ap impacts airfoils 40 and 46 to cause rotation of rotor disk 34A and rotor disk 34B about centerline C_L. Cooling air A_C, which is relatively cooler than primary air A_P, is routed from high pressure compressor 16 (FIG. 1) to high pressure turbine 20. Specifically, cooling air A_C is provided to rim slot 49 so that the air can enter internal cooling channels of blade 32B without having to pass through any non-rotating components when engine 10 is operating.

[0014] Second stage turbine rotor disk 34B of FIG. 1 includes wheel 62 and hub 64, through which holes 66 extend. Wheel 62 includes a plurality of slots 49 that extend through an outer diameter rim of wheel 62. Wheel 62 also includes inner diameter bore 68 through which engine centerline CL extends. First stage turbine rotor disk 34A includes slots 43 and a similar inner diameter bore. Hub 64 extends axially from wheel 62 at inner diameter bore 68 to form an annular body surrounding centerline CL. Rotor disk 34B is also attached to aft mini-disk 52B, which includes axially extending portion 70A and radially extending portion 70B. Mini-disk 52B forms cooling passage 72 along rotor disk 34B. Mini-disk 52B is coupled to hub 64 at joint 74, which comprises a pair of overlapping flanges from hub 64 and axially extending portion 70A. Mini-disk 52B adjoins slots 49 at face seal 76, which comprises a flattened portion that abuts slots 49 and roots 50 of blade 32B.

[0015] Rotor disks 34A and 34B, when rotated during operation of engine 10 via high pressure shaft 28, rotate about centerline CL. Low pressure shaft 24 rotates within high pressure shaft 28. Hub 64 of rotor disk 34B is coupled to high pressure shaft 28, which couples to HPC 16 (FIG. 1) through a rotor hub (not shown). Rotor disk 34A is coupled to a rotor hub (FIG. 3) through tie shaft 78 to define cooling passage 80 between tie shaft 78 and high pressure shaft 28. Cooling air A_C from HPC 16 (FIG. 1) is routed into cooling passage 80 where, due to pressure differentials within engine 10, the air turns to enter holes 66. Within holes 66, the air is bent by the rotation of hub 64 and distributed into cooling passage, or plenum, 72. From cooling passage 72, cooling air A_C flows toward face seal 76, which prevents cooling air A_C from escaping rotor disk 34B, and into slots 49. From slots 49 cooling air A_C enters interior cooling channels of blade 32B to cool airfoil 46 relative to primary air A_P. As such, cooling air A_C is completely contained within rotating components between high pressure turbine stage 20 and high pressure compressor stage 16, as is explained with reference to FIG. 3.

[0016] FIG. 3 is a schematic view of high pressure compressor, or high pressure compressor section, 16 of FIG. 1 showing bleed system 82 having radially inward-mounted inlet 84 for directing cooling air A_C between high pressure shaft 28 and tie shaft 78. High pressure compressor 16 comprises disks 86A and 86B, from which blades 88A and 88B extend. HPC 16 also includes vanes 90A and 90B that extend from HPC case 23C between blades 88A and 88B. Disk 86B is coupled to disk 86A at coupling 92 between rim shrouds 94A and 94B. Disk 86A is coupled to high pressure turbine disk 34A via rotor hub 96 and tie shaft 78. Rotor hub 96 also couples to high pressure shaft 28. High pressure shaft 28 couples second stage high pressure turbine disk 34B to a forward stage (not shown) of HPC 16 in any conventional manner, such as through a rotor hub.

[0017] Cooling air A_C flows from between blade 88B and vane 90A radially inward through inlet 84. In the embodiment shown, inlet 84 comprises a bore through rim shroud 94A, but may extend through rim shroud 94B or be positioned between rim shrouds 94A and 94B. Cooling air A_C is directed radially inward through anti-vortex tube 98, which distributes cooling air within the inter-disk space between disks 86A and 86B. From anti-vortex tube 98, cooling air A_C impacts high pressure shaft 28 and is turned axially downstream to passage 99 in rotor hub 96. Portions of cooling air A_C travel upstream to cool other parts of HPC 16. Passage 99 feeds cooling air A_C into cooling passage 80 between tie shaft 78 and high pressure shaft 28. As such, cooling air A_C is completely bounded by components configured to rotate during operation of gas turbine engine 10. In the embodiment shown, cooling air A_C is bounded by rim shroud 94A, rim shroud 94B, disk 86A, disk 86B, rotor hub 96, shaft 28 and a rotor hub (not shown) joining shaft 28 to a disk of HPC 16. For example, a rotor hub having the opposite orientation as rotor hub 96 could extend between shaft 28 and disk 86B, although HPC 16 would typically include many more stages than two. Although the invention has been described with reference to inlet bore 84, in other embodiments other bleed air inlets that siphon air from HPC 16 and direct the air radially inward toward shaft 28 within rotating components may be used, as are known in the art.

[0018] As discussed previously with reference to FIG. 2, cooling air A_C continues through cooling passage 80 underneath rotor disks 34A and 34B to flow along inner diameter bores, such as inner diameter bore 68 of rotor disk 34B. From cooling passage 80, cooling air A_C flows through holes 66 into plenum 72 between wheel 62 and aft mini-disk 52B. From plenum 72 cooling air A_C travels into slots 49 and into blade 46. Cooling air A_C is thus completely bounded by components configured to rotate during operation of gas turbine engine 10, before being discharged into gas path 60. In the embodiment shown, cooling air A_C is bounded by tie shaft 78, shaft 28 rotor disk 34A, rotor disk 34B, hub 64, aft-mini disk 52B, forward seal plate 54B and blade 32B.

[0019] Because cooling air A_C is bounded by components that rotate when gas turbine engine 10 operates, dynamic losses, such as drag, are avoided, thereby increasing efficiency of HPC 16, reducing the volume of cooling air A_C required for cooling of blades 32B and increasing the overall operating efficiency of engine 10. Furthermore, cooling air A_C is isolated from other flows of cooling air within engine 10, particularly cooling air used to cool first stage turbine blades 32A. For example, cooling air may be directed from the outer diameter of HPC 16, such as at between the tips of vane 90B and blade 88B (FIG. 3). This cooling air is fed into tangential onboard injector 100 (FIG. 2) after flowing radially outward of tie shaft 78, outside of passage 80. As a result of cooling air A_C being isolated from the cooling air for blade 32A, cooling air A_C need not travel through inter-stage cavity 58 from slots 43 to enter slots 49 as has previously been done in the prior art. Cooling air for blade 32A is typically required to be at higher pressures than cooling air A_C because, among other things, blade 32A requires increased cooling and primary air Ap must be kept out of inter-stage cavity 58 via pressurization from the cooling air of first stage blade 32A.

[0020] A further benefit of the present invention is achieved by the flow of cooling air A_C across bore 68 and aft face 102 of disk 34B. Slots 49 of disk 34B are subject to significantly high temperatures from primary air Ap, while bore 68 is subject to less high temperatures due to spacing from primary air A_P. Thus, a temperature gradient is produced across wheel 62. As temperatures within engine 10 fluctuate due to different operating conditions, the temperature gradient induces low cycle fatigue in wheel 62. Low cycle fatigue from the high temperature gradient reduces the life of disk 34B. The temperature of cooling air A_C can be controlled to heat bore 68 and aft face 102 of disk 34B to reduce the temperature gradient across wheel 62, while still remaining relatively cooler than primary air Ap to cool blade 32B. A reduction in the temperature gradient across wheel 62 produces a corresponding increase in the life of disk 34B.

[0021] Furthermore, bore 68 comprises a large mass of circular material that, when subject to heating, experiences thermal growth that increases the diameter of the circular material. An increase in the diameter of bore 68, and wheel 62, pushes turbine blades 32B radially outward, closer to HPT case 23D. Cooling air A_C can be used to condition the temperature of bore 68 to control the thermal growth rate and change in diameter of the circular material, thereby influencing tip clearance between airfoil 46 of blade 32B and shroud 104 attached to HPT case 23D.

[0022] While the invention has been described with reference to an exemplary embodiment(s), it will be understood by those skilled in the art that various changes may be made and equivalents may be substituted for elements thereof without departing from the scope of the invention. In addition, many modifications may be made to adapt a particular situation or material to the teachings of the invention without departing from the essential scope thereof. Therefore, it is intended that the invention not be limited to the particular embodiment(s) disclosed, but that the invention will include all embodiments falling within the scope of the appended claims.

Claims

1. A turbine stage for a gas turbine engine (10) configured to rotate in a circumferential direction about an axis extending through a center of the gas turbine engine, the turbine stage comprising:
a turbine disk (34B) comprising:

an outer diameter edge having slots (49);
an inner diameter bore surrounding the axis;
a forward face;
an aft face;
a hub (64) extending from the inner diameter bore of the turbine disk to form an annular body; and
a plurality of holes (66) extending through the hub;
a plurality of blades coupled to the slots;
a mini-disk (52B) comprising:

an axially extending portion (70A) disposed opposite the hub (64);
a radially extending portion (70B) disposed opposite the aft face of the turbine disk;
an axial retention flange disposed at a radial distal tip of the radially extending portion to engage the slots; and
a coupling disposed at an axially distal tip of the axially extending portion to engage the hub, wherein the mini-disk couples to the aft face of the turbine disk to define a cooling plenum (72) therebetween to direct cooling air to the slots, and wherein the holes permit cooling air from within the hub to enter the cooling plenum; and

a shaft (24) extending from the hub through the inner diameter bore coupling the turbine disk to a compressor disk, wherein the inner diameter bore and the shaft define a cooling passage fluidly coupled to the holes and the plenum.

2. The turbine stage of claim 1 further comprising a cover plate coupled to the forward face of the disk across the slots.

3. The turbine stage of any preceding claim further comprising:

a first stage turbine rotor coupled to the forward face of the disk to define an inter-stage cavity between the first stage turbine rotor and the disk; and
a first stage mini-disk coupled to a forward-facing side of the first stage turbine rotor.

4. A gas turbine engine incorporating the turbine stage of claim 3, the gas turbine engine further comprising:

a compressor stage; w
herein the shaft couples the compressor stage to the hub of the turbine stage, the shaft passing through the inner diameter bore; and
a bleed air inlet for directing cooling air from the compressor to a space radially outward of the shaft.

5. The gas turbine engine of claim 4 wherein the compressor stage comprises:

a first compressor rotor having a plurality of compressor blades extending from a first rim; and
a second compressor rotor having a plurality of compressor blades extending from a second

rim, the second compressor rotor coupled to the first compressor rotor;
wherein the bleed air inlet extends radially inward between the first and second rims.

6. The gas turbine engine of claim 5 further comprising:

a compressor rotor hub connecting the second compressor rotor to the shaft; and
a tie shaft coupling the compressor rotor hub to the first stage turbine rotor.

7. A method of providing compressor bleed air to a turbine stage of a gas turbine engine as claimed in any preceding claim,
the method comprising:

flowing bleed air axially along a shaft connecting a compressor stage to a hub extending from the inner diameter bore of a turbine stage;
passing the bleed air through bore of a rotor disk of the turbine stage;
directing the bleed air radially along an aft surface of the rotor disk; and
feeding the bleed air into a blade slot in a rim of the rotor disk.

8. The method of claim 7 further comprising heating the bore of the rotor disk with the compressor bleed air to reduce a temperature gradient between the rim and the bore to reduce low cycle fatigue.

9. The method of claim 7 or 8 further comprising controlling thermal growth of the rotor disk with the compressor bleed air to influence blade tip clearance.

10. The method of any of claims 7-9 and further comprising:

originating the bleed air from a rim of the compressor stage; and
routing the bleed air radially inward to the shaft;
and optionally wherein the bleed air is bounded from the compressor stage to the turbine stage by components of the gas turbine engine configured to rotate.

11. The method of any of claims 7-10 wherein the bleed air bypasses an inter-stage cavity defined by adjacent rotor disk in the turbine stage.

Patentansprüche

1. Turbinenstufe für ein Gasturbinentriebwerk (10), die dazu konfiguriert ist, sich in eine Umfangsrichtung um eine Achse zu drehen, die sich durch eine Mitte eines Gasturbinentriebwerks erstreckt, wobei die Turbinenstufe Folgendes umfasst:
eine Turbinenscheibe (34B), die Folgendes umfasst:

eine Außendurchmesserkante, die Schlitze (49) aufweist;
eine Innendurchmesserbohrung, die die Achse umgibt;
eine vordere Fläche;
eine hintere Fläche;
eine Nabe (64), die sich von der Innendurchmesserbohrung der Turbinenscheibe erstreckt, um einen ringförmigen Körper zu bilden; und
eine Vielzahl von Öffnungen (66), die sich durch die Nabe erstrecken;
eine Vielzahl von Laufschaufeln, die an die Schlitze gekoppelt sind;
eine Minischeibe (52B), die Folgendes umfasst:

einen sich axial erstreckenden Abschnitt (70A), der gegenüber der Nabe (64) angeordnet ist;
einen sich radial erstreckenden Abschnitt (70B), der gegenüber der hinteren Fläche der Turbinenscheibe angeordnet ist;
einen Axialrückhalteflansch, der an einer radial distalen Spitze des sich radial erstreckenden Abschnitts dazu angeordnet ist, die Schlitze in Eingriff zu nehmen; und
eine Kopplungsvorrichtung, die an einer axial distalen Spitze des sich axial erstreckenden Abschnitts dazu angeordnet ist, die Nabe in Eingriff zu nehmen, wobei die Minischeibe sich an die hintere Fläche der Turbinenscheibe koppelt, um eine Kühlkammer (72) dazwischen zu definieren, um Kühlluft an die Schlitze zu leiten, und wobei die Öffnungen einer Kühlluft aus dem Inneren der Nabe ermöglichen, in die Kühlkammer einzutreten; und
eine Welle (24), die sich von der Nabe durch die Innendurchmesserbohrung erstreckt und die Turbinenscheibe an eine Verdichterscheibe koppelt, wobei die Innendurchmesserbohrung und die Welle einen Kühlkanal definieren, der fluidisch mit den Öffnungen und der Kammer gekoppelt ist.

2. Turbinenstufe nach Anspruch 1, ferner umfassend eine Abdeckplatte über den Schlitzen, die an die vordere Fläche der Scheibe gekoppelt ist.

3. Turbinenstufe nach einem der vorhergehenden Ansprüche, ferner umfassend:

einen Turbinenrotor einer ersten Stufe, der an die vordere Fläche der Scheibe gekoppelt ist, um einen Zwischenstufenhohlraum zwischen dem Turbinenrotor der ersten Stufe und der

Scheibe zu definieren; und
eine Minischeibe der ersten Stufe, die an die nach vorn gewandte Seite des Turbinenrotors der ersten Stufe gekoppelt ist.

4. Gasturbinentriebwerk, das eine Turbinenstufe nach Anspruch 3 enthält, wobei das Gasturbinentriebwerk ferner Folgende umfasst:

eine Verdichterstufe;
wobei die Welle die Verdichterstufe an die Nabe der Turbinenstufe koppelt, wobei die Welle die Innendurchmesserbohrung durchquert; und
einen Zusatzlufteintritt zum Leiten von Kühlluft von dem Verdichter an einen Raum, der sich radial außerhalb der Welle befindet.

5. Gasturbinentriebwerk nach Anspruch 4, wobei die Verdichterstufe Folgendes umfasst:

einen ersten Verdichterrotor, der eine Vielzahl von Verdichterlaufschaufeln aufweist, die sich von einem ersten Rand erstrecken; und
einen zweiten Verdichterrotor, der eine Vielzahl von Verdichterlaufschaufeln aufweist, die sich von einem zweiten Rand erstrecken, wobei der zweite Verdichterrotor an den ersten Verdichterrotor gekoppelt ist;
wobei der Zapflufteinlass sich zwischen dem ersten und dem zweiten Rand radial nach innen erstreckt.

6. Gasturbinentriebwerk nach Anspruch 5, ferner umfassend:

eine Verdichterrotornabe, die den zweiten Verdichterrotor mit der Welle verbindet; und
eine Verbindungswelle, die die Verdichterrotornabe an den Turbinenrotor der ersten Stufe koppelt.

7. Verfahren zum Bereitstellen von Verdichterzapfluft an eine Turbinenstufe eines Gasturbinenmotors nach einem der vorhergehenden Ansprüche, wobei das Verfahren Folgendes umfasst:

axiales Strömen von Zapfluft entlang einer Welle, die eine Verdichterstufe mit einer Nabe verbindet, die sich von der Innendurchmesserbohrung einer Turbinenstufe erstreckt;
Führen der Zapfluft durch eine Bohrung einer Rotorscheibe der Turbinenstufe;
radiales Leiten der Zapfluft entlang einer hinteren Oberfläche der Rotorscheibe; und
Einführen der Zapfluft in einen Laufschaufelspalt in einem Rand der Rotorscheibe.

8. Verfahren nach Anspruch 7, ferner umfassend ein Erwärmen der Bohrung der Rotorscheibe mit der Verdichterzapfluft, um ein Temperaturgefälle zwischen dem Rand und der Bohrung zu reduzieren, um eine Ermüdung bei geringer Lastspielzahl zu reduzieren.

9. Verfahren nach Anspruch 7 oder 8, ferner umfassend ein Steuern einer Wärmeausdehnung der Rotorscheibe mit der Verdichterzapfluft, um ein Laufschaufelspitzenspiel zu beeinflussen.

10. Verfahren nach einem der Ansprüche 7-9 und ferner umfassend:

Stammen der Zapfluft von einem Rahmen der Verdichterstufe; und
Leiten der Zapfluft radial nach innen zu der Welle; und wobei die Zapfluft gegebenenfalls von der Verdichterstufe zu der Turbinenstufe durch Bauteile des Gasturbinentriebwerks begrenzt ist, die dazu konfiguriert sind, sich zu drehen.

11. Verfahren nach einem de Ansprüche 7-10, wobei die Zapfluft einen Zwischenstufenhohlraum umgeht, der durch eine benachbarte Rotorscheibe in der Turbinenstufe definiert ist.

Revendications

1. Étage de turbine pour un moteur à turbine à gaz (10) configuré pour tourner dans une direction circonférentielle autour d'un axe s'étendant à travers un centre du moteur à turbine à gaz, l'étage de turbine comprenant :
un disque de turbine (34B) comprenant :

un bord de diamètre externe ayant des fentes (49) ;
un alésage de diamètre interne encerclant l'axe ;
une face avant ;
une face arrière ;
un moyeu (64) s'étendant à partir de l'alésage de diamètre interne du disque de turbine pour former un corps annulaire ; et
une pluralité de trous (66) s'étendant à travers le moyeu ;
une pluralité d'aubes couplées aux fentes ;
un minidisque (52B) comprenant :

une partie s'étendant axialement (70A) disposée à l'opposé du moyeu (64) ;
une partie s'étendant radialement (70B) disposée à l'opposé de la face arrière du disque de turbine ;
une bride de retenue axiale disposée à une extrémité distale radiale de la partie s'éten-

dant radialement pour venir en prise avec les fentes ; et

un accouplement disposé à une extrémité axialement distale de la partie s'étendant axialement pour venir en prise avec le moyeu, dans lequel le minidisque est couplé à la face arrière du disque de turbine pour définir un plénum de refroidissement (72) entre ceux-ci pour diriger un air de refroidissement vers les fentes, et dans lequel les trous permettent à l'air de refroidissement provenant de l'intérieur du moyeu d'entrer dans le plénum de refroidissement ; et

un arbre (24) s'étendant à partir du moyeu à travers l'alésage de diamètre interne couplant le disque de turbine à un disque de compresseur, dans lequel l'alésage de diamètre interne et l'arbre définissent un passage de refroidissement couplé de manière fluide aux trous et au plénum.

2. Étage de turbine selon la revendication 1 comprenant en outre une plaque de recouvrement couplée à la face avant du disque sur toutes les fentes.

3. Étage de turbine selon une quelconque revendication précédente comprenant en outre :

un rotor de turbine de premier étage couplé à la face avant du disque pour définir une cavité entre étages entre le rotor de turbine de premier étage et le disque ; et

un minidisque de premier étage couplé à un côté dirigé vers l'avant du rotor de turbine de premier étage.

4. Moteur à turbine à gaz incorporant l'étage de turbine selon la revendication 3, le moteur à turbine à gaz comprenant en outre :

un étage de compresseur ;

dans lequel l'arbre couple l'étage de compresseur au moyeu de l'étage de turbine, l'arbre traversant l'alésage de diamètre interne ; et

une entrée d'air de prélèvement pour diriger un air de refroidissement du compresseur à un espace radialement à l'extérieur de l'arbre.

5. Moteur à turbine à gaz selon la revendication 4 dans lequel l'étage de compresseur comprend :

un premier rotor de compresseur ayant une pluralité d'aubes de compresseur s'étendant à partir d'une première périphérie ; et un second rotor de compresseur ayant une pluralité d'aubes de compresseur s'étendant à partir d'une seconde périphérie, le second rotor de compresseur étant couplé au premier rotor de compresseur ; dans lequel l'entrée d'air de prélèvement s'étend radialement vers l'intérieur entre les première et seconde périphéries.

6. Moteur à turbine à gaz selon la revendication 5 comprenant en outre :

un moyeu de rotor de compresseur reliant le second rotor de compresseur à l'arbre ; et

un arbre de liaison couplant le moyeu de rotor de compresseur au rotor de turbine de premier étage.

7. Procédé de fourniture d'air de prélèvement de compresseur à un étage de turbine d'un moteur à turbine à gaz selon une quelconque revendication précédente, le procédé comprenant :

la mise en écoulement d'un air de prélèvement axialement le long d'un arbre reliant un étage de compresseur à un moyeu s'étendant à partir de l'alésage de diamètre interne d'un étage de turbine ;

le passage de l'air de prélèvement à travers l'alésage d'un disque de rotor de l'étage de turbine ;

la direction de l'air de prélèvement radialement le long d'une surface arrière du disque de rotor ; et

l'alimentation de l'air de prélèvement dans une fente d'aube dans une périphérie du disque de rotor.

8. Procédé selon la revendication 7 comprenant en outre le chauffage de l'alésage de disque de rotor avec l'air de prélèvement de compresseur pour réduire un gradient de température entre la périphérie et l'alésage pour réduire la fatigue oligocyclique.

9. Procédé selon la revendication 7 ou 8 comprenant en outre la commande de la croissance thermique du disque de rotor avec l'air de prélèvement de compresseur pour influencer le jeu à l'extrémité des aubes.

10. Procédé selon l'une quelconque des revendications 7 à 9 et comprenant en outre :

la formation d'origine de l'air de prélèvement à partir d'une périphérie de l'étage de compresseur ; et

l'acheminement de l'air de prélèvement radialement vers l'intérieur de l'arbre ; et facultativement dans lequel l'air de prélèvement est délimité de l'étage de compresseur à l'étage de turbine par des composants du moteur à turbine à gaz configurés pour tourner.

11. Procédé selon l'une quelconque des revendications 7 à 10 dans lequel l'air de prélèvement contourne une cavité entre étages définie par un disque de rotor adjacent dans l'étage de turbine.

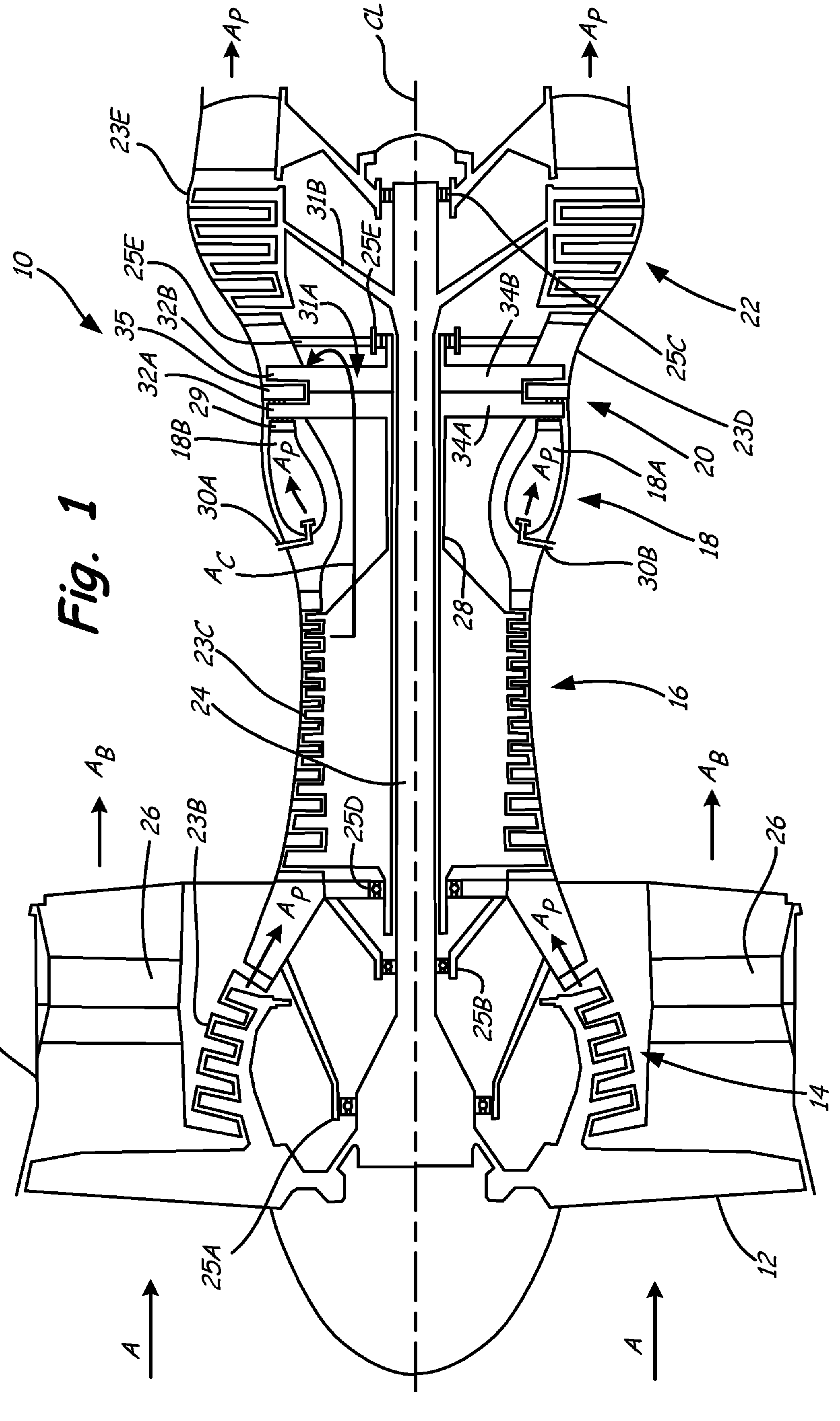

10
12
14
16
18
18A
18B
20
22
23A
23B
23C
23D
23E
24
25A
25B
25C
25D
25E
26
28
29
30A
30B
31A
31B
32A
32B
34A
34B
35
A
A_B
A_P
A_C
CL

Fig. 1

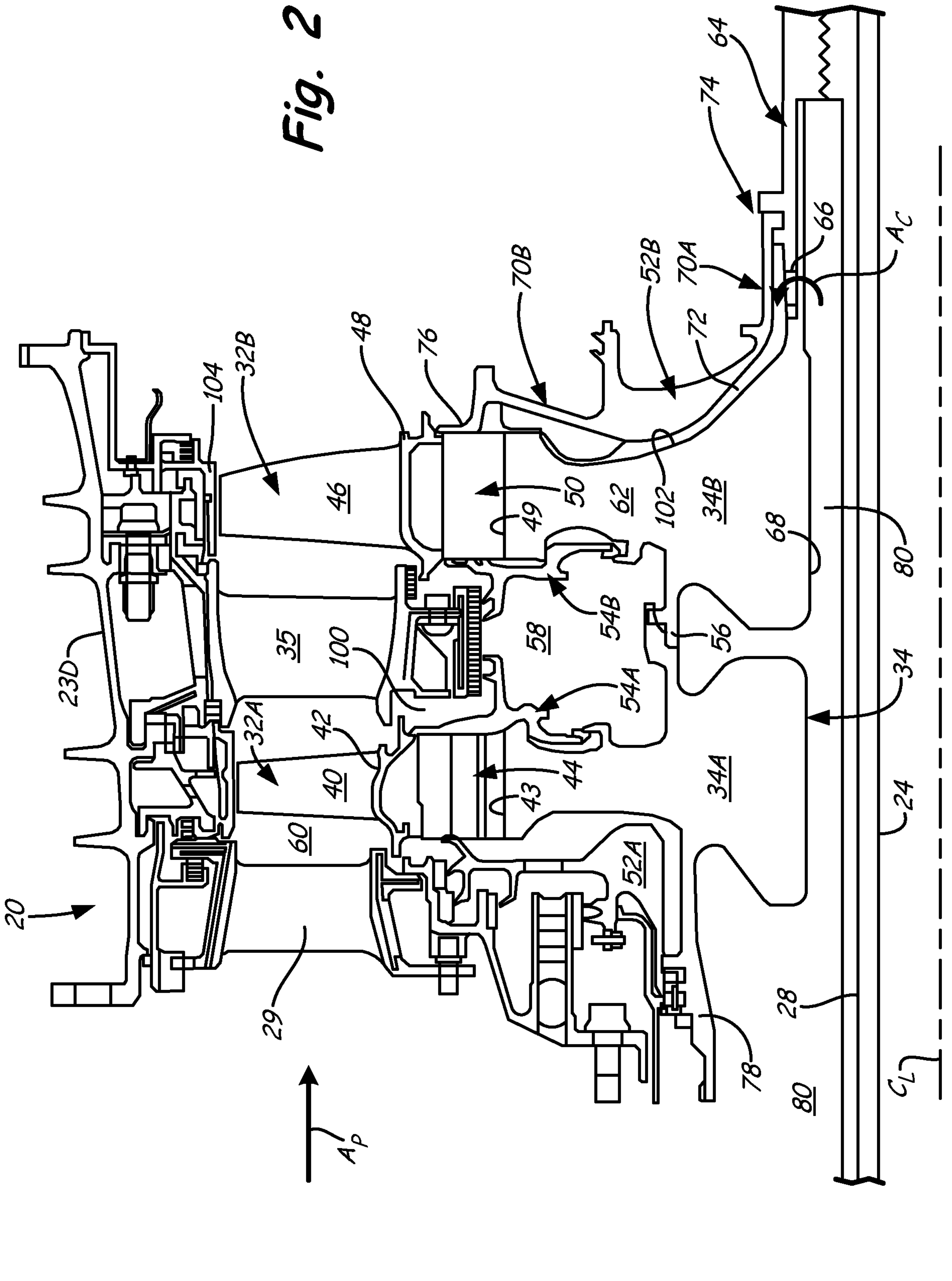
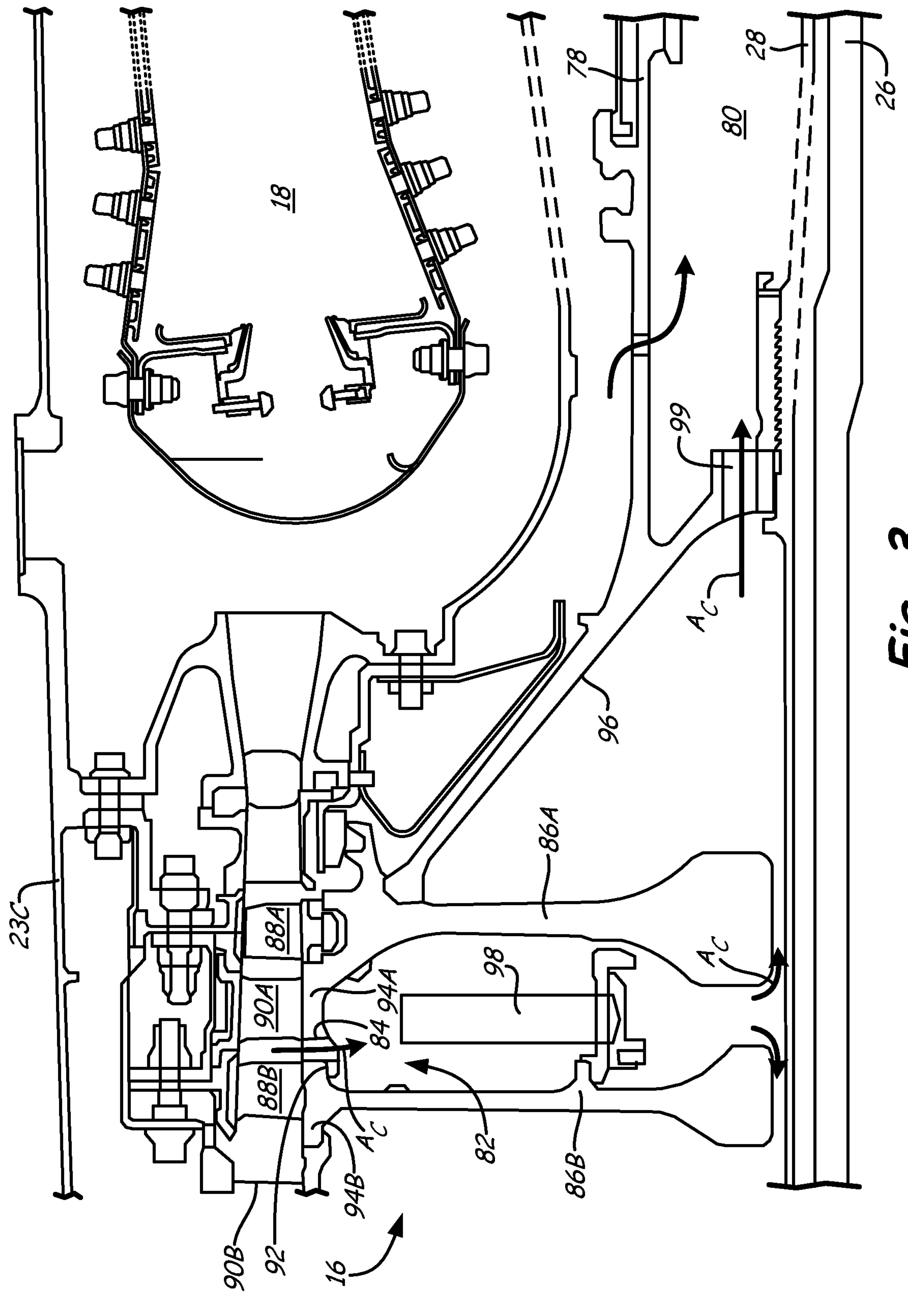

20
29
23D
32A
32B
104
35
40
42
43
44
46
48
49
50
52A
52B
54A
54B
56
58
60
62
64
66
68
70A
70B
72
74
76
78
80
100
102
34
34A
34B
24
28
80
A_P
A_C
C_L

Fig. 2



78
80
28
26
18
99
A_C
96
86A
23C
88A
90A
98
94A
84
A_C
88B
92
A_C
82
86B
94B
90B
16

Fig. 3

REFERENCES CITED IN THE DESCRIPTION

This list of references cited by the applicant is for the reader's convenience only. It does not form part of the European patent document. Even though great care has been taken in compiling the references, errors or omissions cannot be excluded and the EPO disclaims all liability in this regard.

Patent documents cited in the description

- US 4217755 A **[0004]**
- GB 2420155 A **[0004]**