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

(11) EP 1 757 774 A2

(12) **EUROPEAN PATENT APPLICATION**

(43) Date of publication:
28.02.2007 Bulletin 2007/09

(51) Int Cl.:
F01D 5/30 *(2006.01)* ***F01D 5/28*** *(2006.01)*

(21) Application number: **06254334.3**

(22) Date of filing: **17.08.2006**

(84) Designated Contracting States:
AT BE BG CH CY CZ DE DK EE ES FI FR GB GR HU IE IS IT LI LT LU LV MC NL PL PT RO SE SI SK TR
Designated Extension States:
AL BA HR MK YU

(30) Priority: **24.08.2005 US 210520**

(71) Applicant: **General Electric Company**
Schenectady, NY 12345 (US)

(72) Inventors:
- **Farr, Rachel Tavin**
  Covington, KY 41014 (US)
- **Blanton, Lee Alan**
  Cincinnati, OH 45243 (US)
- **Barbe, Roger Owen**
  Cincinnati, OH 45242 (US)
- **Pelletier, Scott Howard**
  Swampscott, MA 01907 (US)

(74) Representative: **Pedder, James Cuthbert et al**
London Patent Operation,
General Electric International, Inc.,
15 John Adam Street
London WC2N 6LU (GB)

(54) **Gas turbine rotor blade assembly and corresponding gas turbine**

(57) A method for assembling a gas turbine engine (12) compressor (50) having a plurality of stages and a plurality of blades (58) coupled to each respective stage includes depositing a silicone oxime sealant (150) onto at least a portion of a compressor blade, and coupling the compressor blade to a compressor disk (56) such that the silicone oxime sealant is between the compressor blade and the compressor disk.

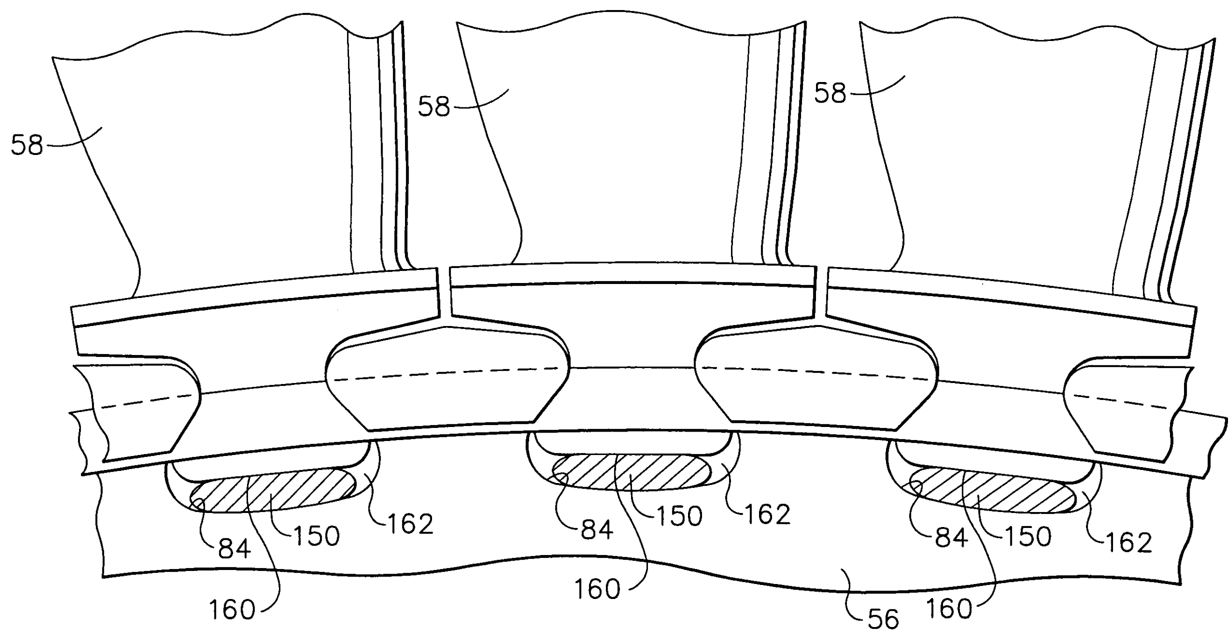

FIG. 4

EP 1 757 774 A2

Description

BACKGROUND OF THE INVENTION

[0001] This invention relates generally to gas turbine engines, and more specifically to methods and apparatus for assembling gas turbine engine components.

[0002] Accurate manufacturing of a component may be a significant factor in determining a fabricating time of the component. Specifically, when the component is a gas turbine engine blade, accurate manufacturing of the blade may be one of the most significant factors affecting an overall cost of fabrication of the gas turbine engine, as well as subsequent modifications, repairs, and inspections of the blade. For example, at least some known gas turbine engines include a compressor for compressing air which is mixed with a fuel and channeled to a combustor wherein the mixture is ignited within a combustion chamber for generating hot combustion gases. At least some known compressors include a rotor assembly that includes at least one row of circumferentially spaced rotor blades. Each rotor blade includes an airfoil that includes a pressure side, and a suction side connected together at leading and trailing edges. Each airfoil extends radially outward from a rotor blade platform. Each rotor blade also includes a dovetail that extends radially inward from a shank coupled to the platform. The dovetail is used to mount the rotor blade within the rotor assembly to a rotor disk or spool.

[0003] During operation, a pressure differential is created between the compressor blade pressure side and the compressor blade suction side which may result in an undesirable leakage flow between the upstream and downstream portions of the rotor. One such possible leakage path may form at an interconnection between each rotor blade and the rotor disk, where a gap may be defined between a blade base member, usually a dovetail design, and a rotor disk groove in which the rotor blades are carried.

[0004] Accordingly, at least one known gas turbine engine includes a silicone acetoxy sealant to facilitate sealing the blade base and the rotor disk. However, as engine performance requirements have increased, resulting in increased operating temperatures, however the known sealant may not withstand the increased operating temperatures for an extended period of time. As a consequence, the sealant degrades causing leakage to occur between the blade and the disk.

BRIEF DESCRIPTION OF THE INVENTION

[0005] In one aspect of the invention, a method for assembling a gas turbine engine compressor having a plurality of stages and a plurality of blades coupled to each respective stage is provided. The method includes depositing a silicone oxime sealant onto at least a portion of a compressor blade, and coupling the compressor blade to a compressor disk such that the silicone oxime sealant is between the compressor blade and the compressor disk.

[0006] In another aspect of the invention, a gas turbine engine rotor assembly is provided. The gas turbine engine rotor assembly includes a rotor disk, a plurality of circumferentially-space rotor blades coupled to the rotor disk, and a silicone oxime sealant deposited onto at least a portion of the rotor blade such that the silicone oxime sealant is between the rotor blade and the disk.

[0007] In a further aspect of the invention, a gas turbine engine is provided. The gas turbine engine includes a rotor disk, a plurality of circumferentially-space rotor blades coupled to the rotor disk, and a silicone oxime sealant deposited onto at least a portion of the rotor blade such that the silicone oxime sealant is between the rotor blade and the disk.

BRIEF DESCRIPTION OF THE DRAWINGS

[0008] The invention will now be described in greater detail, by way of example, with reference to the drawings, in which:-

Figure 1 is a schematic illustration of a gas turbine engine;

Figure 2 is a cross-sectional view of the compressor shown in Figure 1;

Figure 3 is an end view of an exemplary gas turbine engine blade coupled to an exemplary disk; and

Figure 4 is an end view of the exemplary gas turbine engine blade coupled to an exemplary disk shown in Figure 3 including an exemplary sealant.

DETAILED DESCRIPTION OF THE INVENTION

[0009] As used herein, the terms "manufacture" and "manufacturing" may include any manufacturing process. For example, manufacturing processes may include grinding, finishing, polishing, cutting, machining, inspecting, and/or casting. The above examples are intended as exemplary only, and thus are not intended to limit in any way the definition and/or meaning of the terms "manufacture" and "manufacturing". In addition, as used herein the term "component" may include any object to which a manufacturing process is applied. Furthermore, although the invention is described herein in association with a gas turbine engine, and more specifically for use with a compressor blade for a gas turbine engine, it should be understood that the present invention may be applicable to any component and/or any manufacturing process. Accordingly, practice of the present invention is not limited to the manufacture of compressor blades or other components of gas turbine engines.

[0010] Figure 1 is a schematic illustration of a gas turbine engine 10 having a longitudinal axis 11, and including a core gas turbine engine 12 and a fan section 14 positioned upstream of core engine 12. Core engine 12 includes a generally tubular outer casing 16 that defines an annular core engine inlet 18. Casing 16 surrounds a low-pressure booster 20 for raising the pressure of the incoming air to a first pressure level. In one embodiment, engine 10 is a CFM56 engine available from General Electric Aircraft Engines, Cincinnati, Ohio.

[0011] A high pressure, multi-stage, axial-flow compressor 22 receives pressurized air from booster 20 and further increases the pressure of the air to a second, higher pressure level. The high pressure air flows to a combustor 24 and is mixed with fuel. The fuel-air mixture is ignited to raise the temperature and energy level of the pressurized air. The high energy combustion products flow to a first turbine 26 for driving compressor 22 through a first drive shaft 28, and then to a second turbine 30 for driving booster 20 through a second drive shaft 32 that is coaxial with first drive shaft 28. After driving each of turbines 26 and 30, the combustion products leave core engine 12 through an exhaust nozzle 34 to provide propulsive jet thrust.

[0012] Fan section 14 includes a rotatable, axial-flow fan rotor 36 that is driven by second turbine 30. An annular fan casing 38 surrounds fan rotor 36 and is supported from core engine 12 by a plurality of substantially radially-extending, circumferentially-spaced support struts 44. Fan rotor 36 carries a plurality of radially-extending, circumferentially spaced fan blades 42. Fan casing 38 extends rearwardly from fan rotor 36 over an outer portion of core engine 12 to define a secondary, or bypass airflow conduit. A casing element 39 that is downstream of and connected with fan casing 38 supports a plurality of fan stream outlet guide vanes 40. The air that passes through fan section 14 is propelled in a downstream direction by fan blades 42 to provide additional propulsive thrust to supplement the thrust provided by core engine 12.

[0013] Figure 2 is a cross-sectional view of a portion of a compressor 50 that may be used with core gas turbine 12 (shown in Figure 1). In the exemplary embodiment, compressor 50 includes nine stages 45, wherein each stage 46 includes an array of radially-extending, circumferentially-spaced stator vanes 47 and a plurality of peripherally-carried, radially-extending, circumferentially-spaced rotor blades 48. Inlet guide vanes 51 and stator vanes 52 of stages one through three of compressor 50 are variable in that they are pivotable about an axis that extends radially relative to the compressor axis of rotation. Stator vanes 54 of stages four through eight and outlet guide vanes 55 are fixed in position. Additionally, in stages one through three the respective rotor disks 56 include a series of peripherally-spaced, axially-extending dovetail slots 49 into which rotor blades 58 are inserted and from which rotor blades 58 are removed in an axial direction. Rotor disks 60 for stages four through nine, on the other hand, each have a single, circumferentially-extending dovetail slot 62, into which rotor blades 64 are inserted in a generally tangential direction relative to rotor disk 60.

[0014] Compressor 50 includes an inlet 66 that defines a flow passageway 67 having a relatively large flow area, and an outlet 68 that defines a relatively smaller area flow passageway 69 through which the compressed air passes. An outer boundary of the flow passageway is defined by an outer annular casing 70 and an inner boundary of the flow passageway is defined by the blade platforms of respective blades 58, 64 carried by rotors 56, 60, and also by a stationary annular seal ring 72 that is carried at an inner periphery of each of the respective stator sections 52, 54. As shown, respective rotor disks 56, 60 are ganged together by a suitable disk-to-disk coupling arrangement (not shown), and the third stage disk is connected with a drive shaft 74 that is operatively connected with a turbine rotor (not shown).

[0015] Each stator section 52, 54 includes an annular abradable seal that is carried by a respective annular sealing ring 72 and that is adapted to be engaged by respective labyrinth seals carried by 56, 60 in order to minimize air leakage around the respective stators 52, 54. Sealing rings 72 also serve to confine the flow of air to the flow passageway defined by outer casing 70 and the radially innermost surfaces of the respective stator vanes 47.

[0016] Figure 3 is an end view of a plurality of rotor blades 58 coupled to rotor disk 56. Rotor disk 56 includes a plate-like disk body 76 that terminates in an enlarged outer rim 78. Outer rim 78 includes a rotor-blade-receiving circumferential slot 84 that in the exemplary embodiment is substantially U-shaped. Slot 84 has a cross-sectional form of a dovetail, and includes a slot base 86. Slot 84 is defined by a forward sidewall 88 and an aft sidewall 90 that are spaced axially from each other and that extend in a generally radial direction. Each of forward and aft sidewalls 88, 90 has a respective inward convex projection 92, 94 that defines a generally dovetail-type shape of slot 84.

[0017] Rotor blade 64 includes a base member 100 that has a shape that corresponds substantially with that of circumferential slot 84. Base member 100 as shown is in the form of a dovetail and includes an enlarged base portion 110 that is received in lateral recesses 112, 114 formed in rotor slot 84. Base member 100 also includes a recessed portion 116, 118 on each side to receive the inwardly-extending convex projections 92, 94 of rotor slot 84. A blade platform 120 is carried on base member 100 and extends in a generally transverse direction relative to the longitudinal axis of base member 100. Extending longitudinally from upper surface 119 of blade platform 120, and in a direction opposite to that of base member 100, is an airfoil portion 122, which is adapted to contact the gases that pass through engine 10.

[0018] Figure 4 is an end view of plurality of rotor blades 58 coupled to rotor disk 56 shown in Figure 3. In the exemplary embodiment, gas turbine engine 10 also includes a sealant 150 that is formed between at least one rotor blade 58 and rotor disk 56. More specifically, sealant 150 is deposited on a lower surface 160 of rotor blade 58 to facilitate sealing a gap 162 that is defined between blade lower surface 160 and dovetail slot 84. Although only a few rotor blades 58 are illustrated, it should be realized that in the exemplary embodiment, sealant 150 can be utilized to seal at least one gap 162 between a respect rotor blade 58 and disk 60. Alternatively, sealant 150 can be utilized to seal a plurality of gaps 162 defined between a plurality of rotor blades 58 and disk 60. Specifically, sealant 150 can be utilized to seal a single blade 58 on a single disk 60, or a plurality of blades 58 on a single disk 60. Moreover, sealant 150 can be utilized to seal a plurality of blades 58 coupled to a plurality of disks 60. In the exemplary embodiment, sealant 150 is utilized on the first three high pressure compressor stages 170, 172, and 174 (shown in Figure 2) to facilitate sealing the gaps 162 defined between each rotor blade 58 and rotor disk 60.

[0019] In the exemplary embodiment, sealant 150 is deposited on blade lower surface 160. After a predetermined quantity of time sufficient to cure sealant 150 has elapsed, blade 58 is coupled to disk 60. According, sealant 150 substantially seals gap 162 such that airflow cannot be channeled through gap 162.

[0020] In the exemplary embodiment, sealant 150 is a room temperature vulcanizing silicone oxime sealant that is deposited onto at least a portion of the compressor blade 58. Oxime as used herein is defined as one in a class of chemical compounds with the general formula R1R2CNOH, where R1 is an organic side chain and R2 is either hydrogen, forming an aldoxime, or another organic group, forming a ketoxime, and can be formed by the action of hydroxylamine on aldehydes or ketones.

[0021] Moreover, during use, sealant 150 is deposited onto at least a portion of blade 58 as a thixotropic paste. Thixotropic as used herein, is defined a gel-like substance that becomes a fluid when subjected to either stirring or shaking, and then returns to a semi-solid state upon standing. Accordingly, sealant 150 is applied to at least a portion of blade 58 in a semi-fluidic state. Sealant 150 is then allowed to cure or harden onto blade 58. After sealant 150 has substantially cured, blade 58 is coupled to disk 60. In the exemplary embodiment, sealant 150 is a room temperature oxime-cure silicone sealant such as for example, Loctite™ 5920. Accordingly, sealant 150 is capable of sealing gap 162 and retaining its elastomeric properties up to temperatures of at least 600 degrees Fahrenheit.

[0022] Described herein is an exemplary sealant that facilitates reducing and/or eliminating the airflow between a high pressure compressor rotor disk and a compressor rotor blade. More specifically, the sealant is applied to a plurality of compressor blades that are coupled to the first three stages of a gas turbine engine compressor assembly. The sealant described herein is a room temperature vulcanizing silicone oxime sealant that is configured to withstand temperatures to at least 600 degrees Fahrenheit.

[0023] More specifically, the sealant described herein facilitates improving gas turbine engine performance by preventing airflow leakage between the compressor blades and the compressor rotor disk. For example, known materials used in such applications cannot withstand operating temperatures of greater than approximately 600 degrees Fahrenheit for an extended period of time. As a consequence, leakage occurs when the known sealants material degrades with time and temperature and effectively disappears, thus eliminating the airflow seal around the component. Whereas the sealant described herein is configured to withstand temperatures greater than 600 degrees Fahrenheit and thus increase engine performance over an extended period of time.

Claims

1. A gas turbine engine rotor assembly (10) comprising:

   a rotor disk (56);
   a plurality of circumferentially-space rotor blades (58) coupled to said rotor disk; and
   a silicone oxime sealant (150) deposited onto at least a portion of said rotor blade or said rotor disk such that said silicone oxime sealant is between said rotor blade and said disk.

2. A gas turbine engine rotor assembly (10) in accordance with Claim 1 wherein said rotor disk (56) comprises a compressor rotor disk, and said rotor blades (58, 64) comprise compressor rotor blades.

3. A gas turbine engine rotor assembly (10) in accordance with Claim 1 wherein said silicone oxime sealant (150) comprises a room temperature vulcanizing silicone oxime sealant.

4. A gas turbine engine rotor assembly (10) in accordance with Claim 1 wherein said rotor disk (56) comprises at least one of a first stage (170) compressor rotor disk, a second stage (172) compressor rotor disk, and a third stage (174) compressor rotor disk.

5. A gas turbine engine rotor assembly (10) in accordance with Claim 1 wherein said silicone oxime sealant (150) is operable at a temperature greater than 600 degrees Fahrenheit.

6. A gas turbine engine rotor assembly (10) in accordance with Claim 1 wherein said rotor blade (58) comprises a dovetail, said turbine rotor disk (60) comprises a dovetail slot (62), said silicone oxime sealant (150) deposited between said dovetail and said dovetail slot.

7. A gas turbine engine (12) comprising:

a rotor disk (56);
a plurality of rotor blades (58) coupled to said rotor disk; and
a silicone oxime sealant (150) deposited onto at least a portion of at least one of said rotor blades or said rotor disk such that said silicone oxime sealant is between said rotor blade and said disk.

8. A gas turbine engine (12) in accordance with Claim 7 wherein said rotor disk (56) comprises a compressor rotor disk (58), and said rotor blades comprise compressor rotor blades.

9. A gas turbine engine (12) in accordance with Claim 7 wherein said silicone oxime sealant (150) comprises a room temperature vulcanizing silicone oxime sealant.

10. A gas turbine engine (12) in accordance with Claim 7 wherein said rotor disk (56) comprises at least one of a first stage (170) compressor rotor disk, a second stage (172) compressor rotor disk, and a third stage (174) compressor rotor disk.

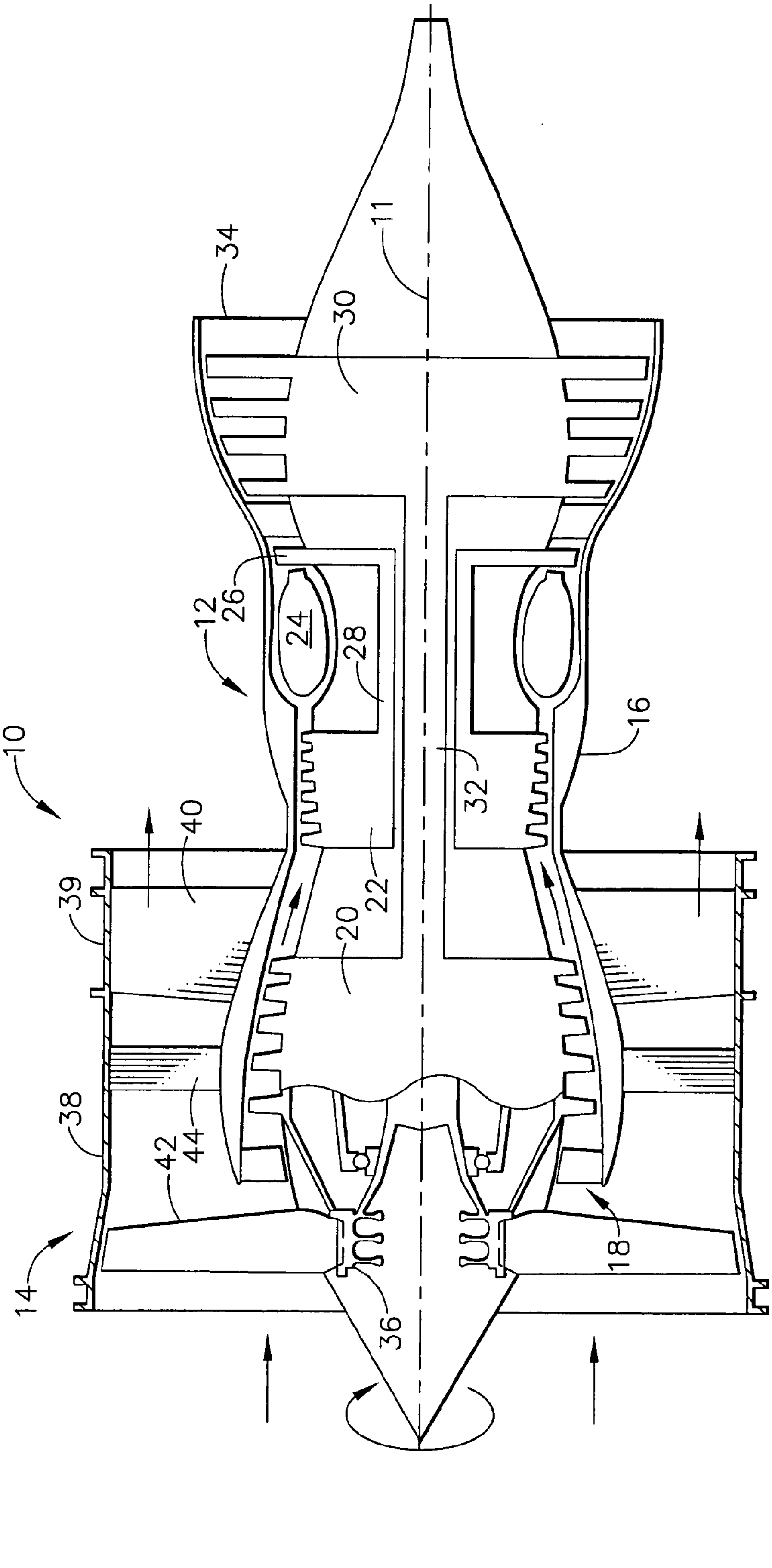

FIG. 1

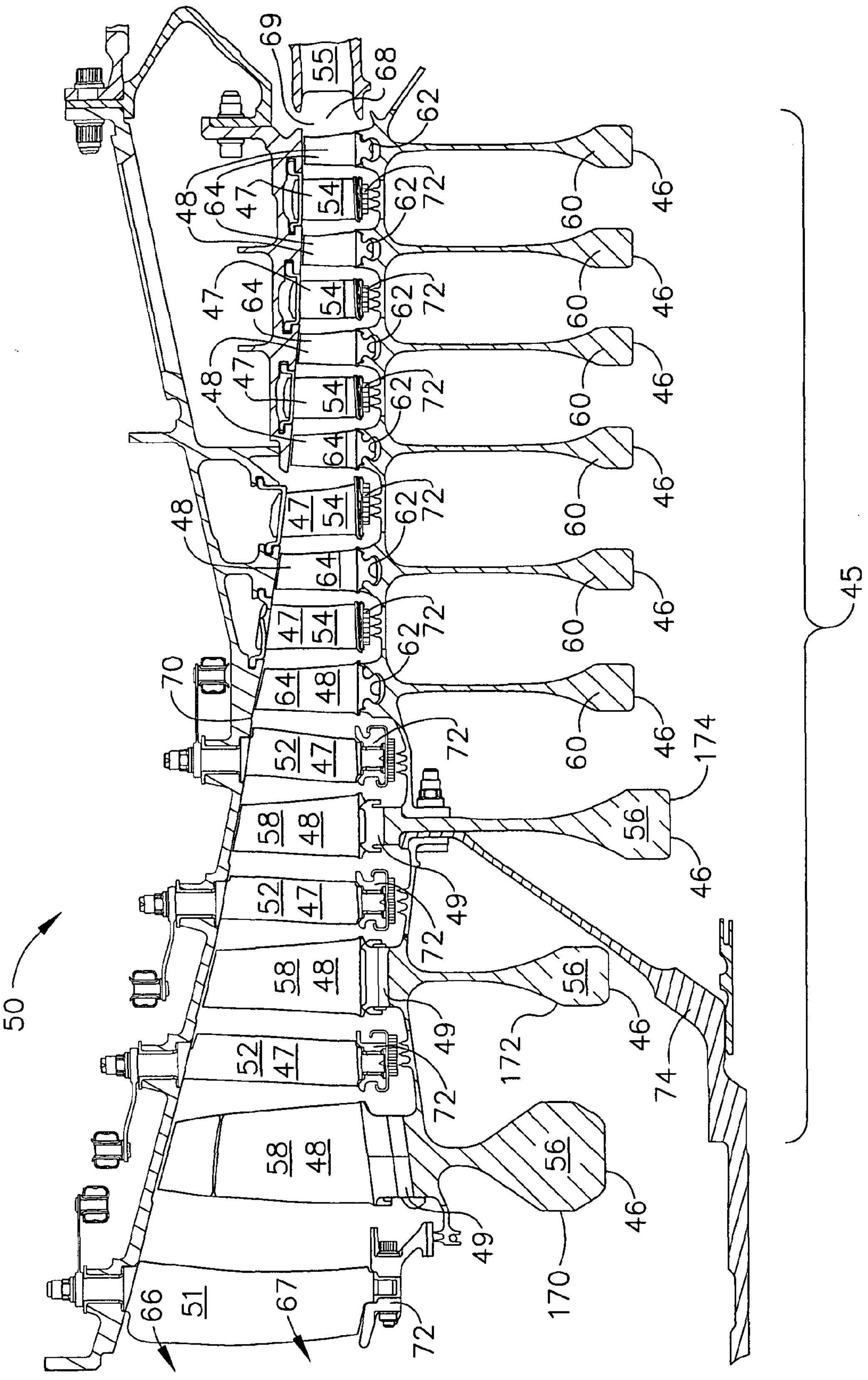
50
45
46
47
48
49
51
52
54
55
56
58
60
62
64
66
67
68
69
70
72
74
170
172
174

FIG. 2

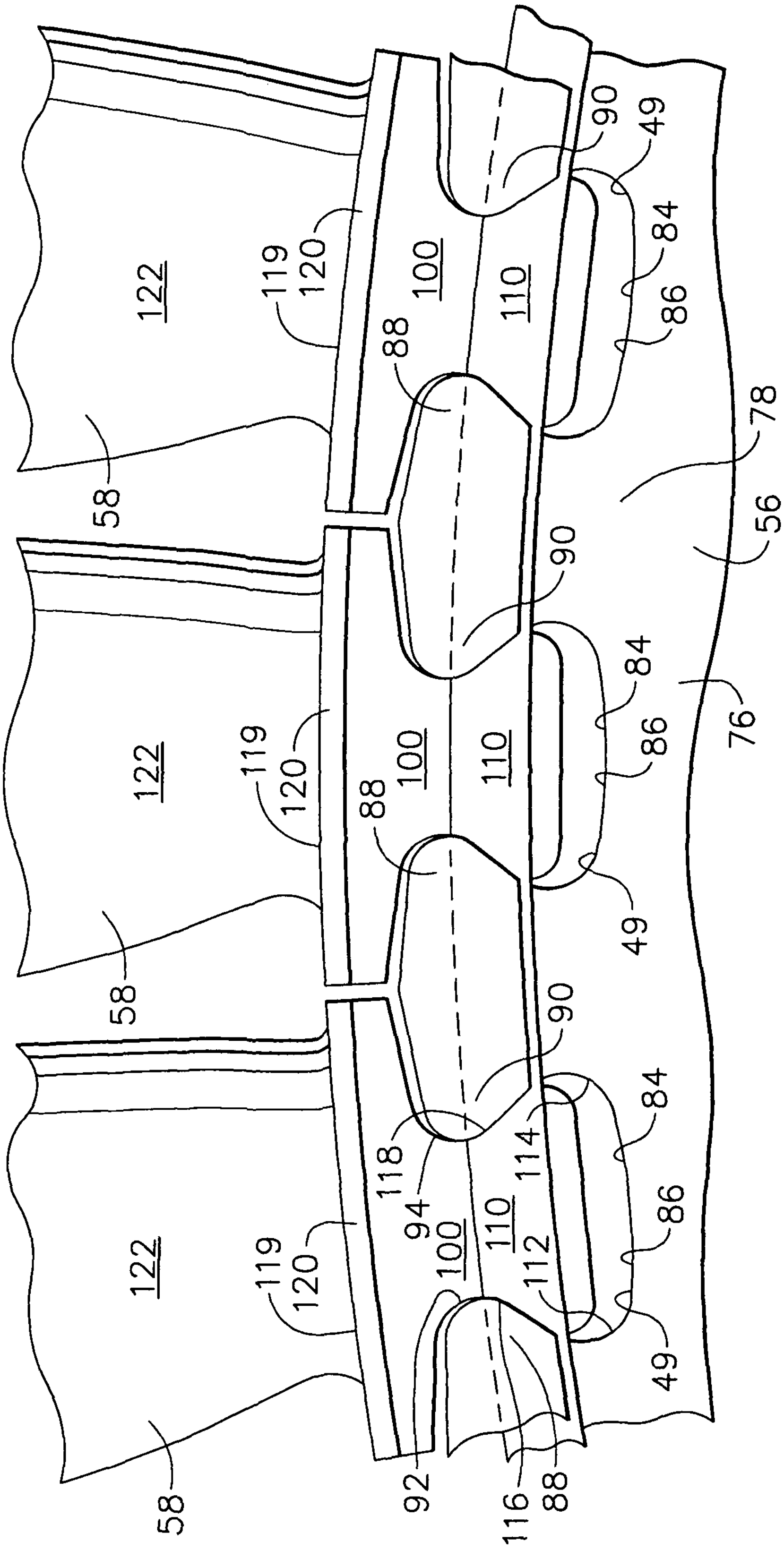

FIG. 3

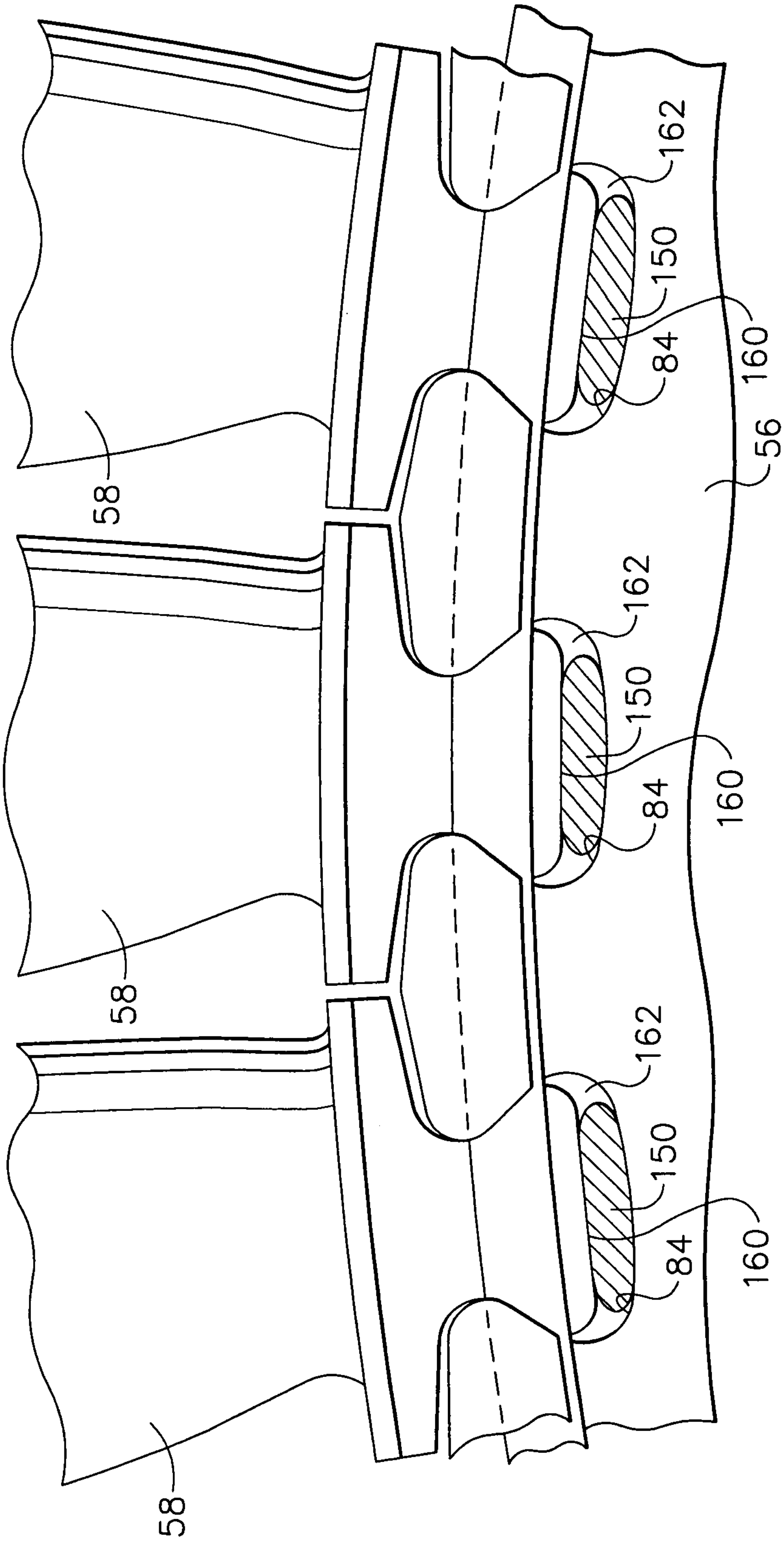

FIG. 4