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

(11) EP 2 578 805 A1

(12) EUROPEAN PATENT APPLICATION

(43) Date of publication:
10.04.2013 Bulletin 2013/15

(51) Int Cl.:
F01D 5/20 *(2006.01)*

(21) Application number: **12187063.8**

(22) Date of filing: **03.10.2012**

(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
Designated Extension States:
BA ME

(30) Priority: **05.10.2011 US 201113253328**

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

(72) Inventors:
- **Nussbaum, Jeffery Howard**
Wilmington, MA Massachusetts 01887 (US)
- **Cox, Andrea MacPherson**
Haverhill, MA Massachusetts 01830 (US)
- **Pommer, Leslie Martin**
Danvers, MA Massachusetts 01923-3216 (US)

(74) Representative: **Williams, Andrew Richard**
Global Patent Operation - Europe
GE International Inc
15 John Adam Street
London WC2N 6LU (GB)

(54) Gas turbine engine airfoil with tip recesses

(57) A gas turbine engine airfoil (54, 225, 88) extending from an airfoil base (56, 228,91) to an airfoil tip (58, 138, 86) at a free end (87) of the airfoil (54, 225, 88) and from a leading edge (LE) to a trailing edge (TE) of the airfoil (54, 225, 88) has at least one recess (92) extending into and circumferentially completely through the airfoil tip (58, 138, 86) and located inwardly of the leading and trailing edges (LE, TE). The recess (92) is located in an area (93) of the airfoil tip (58, 138, 86) subject to high tip vibratory stress and or rubs and may be a circular scallop (94) having a scallop radius (R) and a maximum depth (D) of a few mills in a range of about .005 - .01 inches as measured from a nominal tip edge (106) without the recess. The airfoil may be on a radial blade (200), a stator vane (204), or an impeller blade (84). An annular tip seal (144) may surround a plurality of such airfoils with an airfoil tip (142). A corresponding method of reducing vibratory stress at an airfoil tip is also disclosed.

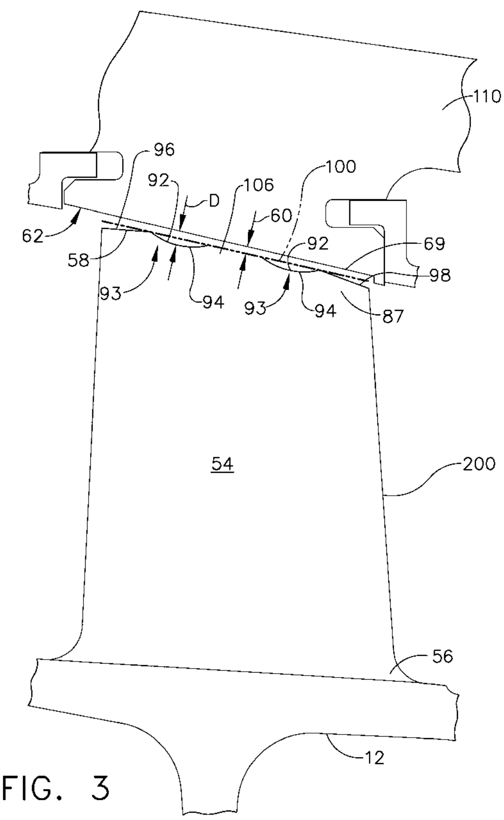

FIG. 3

EP 2 578 805 A1

Description

BACKGROUND OF THE INVENTION

FIELD OF THE INVENTION

[0001] This application relates generally to gas turbine engine clearances between relatively rotating airfoil tips and seals and, more particularly, to clearances at airfoil tips of blades and cantilevered vanes.

DESCRIPTION OF RELATED ART

[0002] Gas turbine engine blades have airfoils have gaps between the rotatable blades and static seals or casings and static cantilevered vanes and rotating seals or rotors in both turbines and the compressors. Often referred to as tip clearances, it is desirable for an efficiency standpoint to reduce the gap between the rotating component and the radially adjacent static part in order to reduce leakage of the gas stream across this gap. The leakage not only reduces efficiency of the compressor or the turbine, but also reduces the life of the turbine blade tips and shroud members because of high temperatures acting on the parts.

[0003] Some gas turbine engine designers will set the gap such that the blades will not rub at all. Some designers provide a negative gap in order to produce rub during the initial engine break-in in order to allow for the normal wear from the rub to produce a smooth and close to zero gap as possible. However, this rub can have undesirable results such as high vibratory stress at an airfoil tip of a rotatable blade or static cantilevered vane or at a tip of an impeller. A rub can have undesirable result of a loss of material fatigue strength.

[0004] Blade tip clearances are a compromise between avoiding rubs and minimizing leakage for best engine performance. Rubs can lead to loss of strength, cracking and additional maintenance. Currently, clearances are often set more open, especially at LE and TE where blades are thin. This avoids rubs in areas of high stress, maintains material properties, and avoids tip cracking. Thus, it is highly desirable to provide an airfoil tip that has low leakage and avoids rubs in areas of high stress in order to maintain material properties and avoid tip cracking.

SUMMARY OF THE INVENTION

[0005] A gas turbine engine component includes an airfoil extending from an airfoil base to an airfoil tip at a free end of the airfoil and extending downstream from a leading edge to a trailing edge of the airfoil. At least one recess extends into and circumferentially completely through the airfoil tip and is located inwardly of the leading and trailing edges of the airfoil.

[0006] The recess may be located in an area of the airfoil tip subject to high tip vibratory stress and or rubs and may be a circular scallop having a scallop radius. The recess may have a maximum depth of a few mills in a range of about 0.127-0.254mm (5-10 mills (.005 - .01 inches)) as measured from a nominal tip edge without the recess.

[0007] The airfoil may be on a gas turbine engine radial or impeller compressor blade. The airfoil may be on a gas turbine engine stator vane and extend radially inwardly from a base of the vane airfoil at an outward end of the stator vane to a vane airfoil tip of the vane airfoil at a radial inward end of the stator vane and extend downstream from a leading edge to a trailing edge of the vane airfoil. The recess extends into and circumferentially completely through the vane airfoil tip and the recess is located inwardly of the leading and trailing edges of the vane airfoil.

[0008] A gas turbine engine assembly may include a plurality of the airfoils and an airfoil tip clearance between airfoil tips and an annular tip seal surrounding the airfoil tips containing the recesses or scallops.

[0009] A method of reducing vibratory stress at an airfoil tip at a free end of an airfoil of a gas turbine engine component includes determining an area or areas of high tip vibratory stress and machining, cutting, or otherwise forming at least one recess or scallop extending into and circumferentially completely through the airfoil tip in the area or areas of high tip vibratory stress.

[0010] The airfoil tip and airfoil may be in pluralities of airfoil tips and airfoils from a single stage or circumferential row of blades or vanes surrounded by annular tip seals and the determining an area or areas of high tip vibratory stress includes running or rotating a rotor containing the blades or an annular tip seal surrounding the vanes respectively and observing nicks or scratches in the annular seal or airfoil tips. The recesses or scallops are formed in area or areas of high tip vibratory stresses adjacent the nicks or scratches.

BRIEF DESCRIPTION OF THE DRAWINGS

[0011] The foregoing aspects and other features of the invention are explained in the following description, taken in connection with the accompanying drawings where:

FIG. 1 is a sectional view illustration of a gas turbine engine having axial and centrifugal compressor stages.

FIG. 2 is an enlarged sectional view illustration of an axial compressor stage of the engine illustrated in FIG. 1.

FIG. 3 is a schematical sectional illustration of an airfoil tip with recesses in the axial compressor stage of the engine illustrated in FIG. 2.

FIG. 4 is an enlarged sectional view illustration of the centrifugal compressor stage of the engine illustrated in FIG. 1.

FIG. 5 is an enlarged sectional view illustration of the recess in centrifugal compressor stage of the engine illustrated in FIG. 4.

DETAILED DESCRIPTION OF THE INVENTION

[0012] Illustrated in FIG. 1 is a gas turbine engine 8 with a high pressure gas generator 10. The high pressure gas generator 10 has a high pressure rotor 12 including, in downstream flow relationship, a high pressure compressor 14, a combustor 52, and a high pressure turbine 16. The high pressure compressor 14 includes high pressure multiple stage axial compressor stages and a single stage centrifugal compressor 18 as a final compressor stage. The rotor 12 is rotatably supported about an engine centerline 28 by a forward bearing 20 in a front frame 22 and a rear bearing 24 disposed downstream of high pressure turbine 16 in a turbine frame 26.

[0013] The exemplary embodiment of the compressor 14 illustrated herein includes a five stage axial compressor 30 followed by the single stage 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. The compressor 14 includes a forward casing 110 and an aft casing 114. The forward casing 110 generally surrounds the axial compressor 30 and the aft casing 114 generally surrounds the centrifugal compressor 18 and supports the diffuser 42 directly downstream of the centrifugal compressor 18.

[0014] Referring further to FIG. 2, the five stage axial compressor 30 includes third and fourth compressor stages 190, 193 of rotatable compressor blades 200 circumferentially surrounded by a blade shroud 69. A circumferential row of non-rotatable stator vanes 204 is disposed axially between the third and fourth compressor stages 190, 193 of the rotatable compressor blades 200. Illustrated in FIGS. 2 and 3 is a compressor blade 200 having a blade airfoil 54 extending radially outwardly from an airfoil base 56 located on the high pressure rotor 12 to a radially outer blade airfoil tip 58 at a free end 87 of the blade airfoil 54 as measured along a span S of the blade airfoil 54. The blade airfoil 54 extends downstream from a leading edge LE to a trailing edge TE. Each radially outer blade airfoil tip 58 is radially spaced apart and inwardly from and adjacent to a blade rub land 62 of the blade shroud 69 mounted on the compressor forward casing 110. An annular rotor airfoil tip clearance 60 is defined between the radially outer blade airfoil tip 58 and the blade rub land 62 on the compressor forward casing 110.

[0015] The stator vanes 204 are cantilevered from and fixed to the forward casing 110 of their radial outward ends 234 and are unsupported at their radial inward ends 236 which are free ends 87. A vane airfoil 225 extends radially between the opposite radial outward and inward ends 234, 236. Each vane airfoil 225 extends radially inwardly from a base 228 of the vane airfoil 225 at the outward end 234 of the stator vane 204 to a vane airfoil tip 138 of the vane airfoil 225 at the radial inward end 236 of the stator vane 204. The vane airfoil 225 extends downstream from a leading edge LE to a trailing edge TE. Each vane airfoil tip 138 is radially spaced apart and outwardly from and adjacent to a rotor seal land 232 on the high pressure rotor 12. An annular vane airfoil tip clearance 66 is defined between the vane airfoil tip 138 and the rotor seal land 232 on the high pressure rotor 12.

[0016] Referring to FIGS. 1 and 4, compressor discharge pressure (CDP) air 76 is discharged from the impeller 32 of the centrifugal compressor 18 and directly into a diffuser 42 and then through a deswirl cascade 44 into a combustion chamber 45 within the combustor 52. Referring more particularly to FIG. 1, the impeller 32 includes a plurality of impeller compressor blades 84 including impeller airfoils 88 extending outwardly from impeller airfoil bases 91 on a rotor disc portion 82 to impeller airfoil tips 86 at free ends 87 of the impeller airfoils 88. Each of the impeller airfoils 88 extends downstream from a leading edge LE to a trailing edge TE of the impeller airfoil 88. An annular centrifugal blade tip shroud 90 surrounds the impeller compressor blades 84 and impeller airfoils 88. The centrifugal blade tip shroud 90 is adjacent to the impeller airfoil tips 86 defining an annular impeller airfoil tip clearance 80 therebetween. The impeller airfoil tip clearance 80 varies in axial width W in a radial direction R as measured from the engine centerline 28.

[0017] It is desirable to minimize the rotor airfoil tip clearance 60, the stator airfoil tip clearance 66, and the impeller airfoil tip clearance 80 during the engine operating cycle and avoid or minimize rubs between the associated lands and airfoil and blade tips, particularly, during engine accelerations such as during cold bursts. In more general terms, it is desirable to minimize airfoil tip clearances 140 (illustrated herein as the rotor, vane, and impeller airfoil tip clearances 60, 66, 80) between airfoil tips 142 (illustrated herein as the blade, vane, and impeller airfoil tips 58, 138, 86) and annular tip seals 144 (illustrated herein as the blade rub land 62 on the compressor forward casing 110, the rotor seal land 232 on the high pressure rotor 12, and the centrifugal blade tip shroud 90). To this end, the rotor and vane airfoil tips 58, 138 and the impeller airfoil tips 86, as illustrated in FIGS. 3 and 4, have recesses 92 located in areas 93 subject to high tip vibratory stress to avoid rubs in the areas 93 of high stress, maintain material properties, and avoid tip cracking.

[0018] As exemplified in FIG. 3, the recesses 92 in the airfoil tips and impeller blade tips are located inwardly of the leading and trailing edges LE, TE of the airfoils and are illustrated herein the form of circular scallops 94. The airfoil and impeller are solid and the recesses 92 and scallops 94 extend circumferentially completely through the tips. The airfoil tips may also have leading and trailing edge corner cuts 96, 98 to further reduce vibratory stress deterioration of the airfoil tips which is particularly useful for thin leading and trailing edges LE, TE.

[0019] The recesses 92 and scallops 94 are formed in or cut into nominal airfoil tip edges 106 of the rotor airfoil tips 58 which is illustrated as a dashed line in FIG. 3. The recesses 92 and scallops 94 are also illustrated as formed from or cut into impeller tip edges 108 of the impeller airfoil tips 86 as illustrated in FIGS. 4 and 5. Nominal airfoil tip and impeller blade tip shapes 100, 102 are illustrated in dashed lines in FIGS. 3 and 5. The nominal shapes are the originally designed tip shapes before the recesses or scallops are formed or cut in the tips. The nominal airfoil tip edge 106 are equidistantly spaced apart from the blade rub land 62 and are, thus, conical as illustrated in FIG. 3 or cylindrical. The nominal impeller tip edge 108 has an impeller blade tip shape 102 that may be a simple or a compound curve including at least a first sector 120 having a first radius of curvature R1 as illustrated in FIG. 5.

[0020] The recesses 92 and scallops 94 are very shallow having a maximum depth D of only a few mills such as in a range of about 5-10 mills (.005 - .01 inches) as measured from the nominal airfoil and impeller tip edges 106, 108 respectively for the exemplary engine illustrated herein. Larger engines may have larger recesses or scallops. The scallops 94 are circular having a scallop radius SR. The scallop radius SR of the scallops 94 in the impeller tip edges 108 of the impeller airfoil tips 86 is smaller than the first radius of curvature R1 of the impeller blade tip shape 102.

[0021] There are known methods of determining areas of high tip vibratory stress in the airfoil tips and impeller blade tips. There are analytical methods such as finite element analysis, empirical methods, and semi-empirical in which a combination of testing and analytical methods are used to determine if stresses will produce cracks or if there is undesirable rubbing. During engine production and overhaul, the engine is often run or rotated and rubbing may occur between airfoil tips and surrounding seals indicating areas of high tip vibratory stress in the airfoil tips and impeller blade tips. Nicks and scratches observed after such engine runs indicate such areas and where the recesses or scallops should be placed on the airfoil tips and impeller blade tips. The recesses and scallops can be machined or cut into the airfoil tips adjacent the nicks and scratches. The size and depth of the recesses and scallops may be tailored for each individual airfoil and, thus, vary from airfoil to airfoil within a stage or annular row of blades, vanes or impeller blades.

[0022] The present invention has been described in an illustrative manner. It is to be understood that the terminology which has been used is intended to be in the nature of words of description rather than of limitation. 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 scope of the invention.

[0023] Various aspects and embodiments of the invention are indicated in the following clauses:

1. A gas turbine engine component comprising:

    an airfoil extending from an airfoil base to an airfoil tip at a free end of the airfoil,

    the airfoil extending downstream from a leading edge to a trailing edge of the airfoil,

    at least one recess extending into and circumferentially completely through the airfoil tip, and

    the recess located inwardly of the leading and trailing edges of the airfoil.

2. The component as claimed in Clause 1 wherein the recess is located in an area of the airfoil tip subject to high tip vibratory stress and or rubs.

3. The component as claimed in Clause 2 wherein the recess is a circular scallop having a scallop radius.

4. The component as claimed in Clause 2 wherein the recess has a maximum depth of a few mills in a range of about .005 - .01 inches as measured from a nominal tip edge without the recess.

5. A gas turbine engine compressor blade comprising:

    an airfoil extending radially outwardly from an airfoil base located on a gas turbine engine rotor to an airfoil tip at a free end of the airfoil,

    the airfoil extending downstream from a leading edge to a trailing edge of the airfoil,

    at least one recess extending into and circumferentially completely through the airfoil tip, and

    the recess located inwardly of the leading and trailing edges of the airfoil.

6. The gas turbine engine compressor blade as claimed in Clause 5 wherein the recess is located in an area of the airfoil tip subject to high tip vibratory stress and or rubs.

7. The gas turbine engine compressor blade as claimed in Clause 6 wherein the recess is a circular scallop having a scallop radius.

8. The gas turbine engine compressor blade as

claimed in Clause 6 wherein the recess has a maximum depth of a few mills in a range of about .005 - .01 inches as measured from a nominal tip edge without the recess.

9. The gas turbine engine compressor blade as claimed in Clause 6 wherein the gas turbine engine compressor blade is a radial compressor blade or an impeller compressor blade.

10. A gas turbine engine stator vane comprising:

a vane airfoil extending radially inwardly from a base of the vane airfoil at an outward end of the stator vane to a vane airfoil tip of the vane airfoil at a radial inward end of the stator vane,

the vane airfoil extending downstream from a leading edge to a trailing edge of the vane airfoil,

at least one recess extending into and circumferentially completely through the vane airfoil tip, and

the recess located inwardly of the leading and trailing edges of the vane airfoil.

11. The gas turbine engine stator vane as claimed in Clause 10 wherein the recess is located in an area of the vane airfoil tip subject to high tip vibratory stress and or rubs.

12. The gas turbine engine stator vane as claimed in Clause 11 wherein the recess is a circular scallop having a scallop radius.

13. The gas turbine engine stator vane as claimed in Clause 11 wherein the recess has a maximum depth of a few mills in a range of about .005 - .01 inches as measured from a nominal tip edge without the recess.

14. A gas turbine engine assembly comprising:

a plurality of airfoils,

each of the airfoils extending from an airfoil base to an airfoil tip at a free end of the airfoil,

each of the airfoils extending downstream from a leading edge to a trailing edge of the airfoil,

an airfoil tip clearance between airfoil tips and an annular tip seal surrounding the airfoil tips,

at least one recess extending into and circumferentially completely through each of the airfoil tips, and

the recess located inwardly of the leading and trailing edges of the airfoil.

15. The assembly as claimed in Clause 14 wherein the recess is located in an area of the airfoil tip subject to high tip vibratory stress and or rubs.

16. The assembly as claimed in Clause 15 wherein the recess is a circular scallop having a scallop radius.

17. The assembly as claimed in Clause 15 wherein the recess has a maximum depth of a few mills in a range of about .005 - .01 inches as measured from a nominal tip edge without the recess.

18. The assembly as claimed in Clause 14 further comprising:

a plurality of blades including the airfoils,

the airfoils extending radially outwardly from the airfoil base to the airfoil tip,

the airfoil base mounted on a gas turbine engine rotor, and

wherein the annular tip seal is a blade rub land.

19. The assembly as claimed in Clause 18 wherein the recess is located in an area of the airfoil tip subject to high tip vibratory stress and or rubs.

20. The assembly as claimed in Clause 18 wherein the recess is a circular scallop having a scallop radius.

21. The assembly as claimed in Clause 14 further comprising:

the airfoils being vane airfoils on stator vanes cantilevered from and fixed to an engine casing at radial outward ends of the stator vane,

the airfoils unsupported at free radial inward ends of the airfoils, and

wherein the annular tip seal is a rotor seal land on a rotor of the assembly.

22. The assembly as claimed in Clause 21 wherein the recesses are located in an area of the airfoil tips subject to high tip vibratory stress and or rubs.

23. The assembly as claimed in Clause 21 wherein the recesses are circular scallops having scallop radii.

24. The assembly as claimed in Clause 14 further comprising:

the airfoils being impeller airfoils extending outwardly from impeller airfoil bases on a rotor disc portion of an impeller to impeller airfoil tips at free ends of the impeller airfoils, and
wherein the annular tip seal is an annular centrifugal blade tip shroud.

25. The assembly as claimed in Clause 24 wherein the recesses are located in an area of the airfoil tips subject to high tip vibratory stress and or rubs.

26. The assembly as claimed in Clause 24 wherein the recesses are circular scallops having scallop radii.

27. A method of reducing vibratory stress at an airfoil tip at a free end of an airfoil of a gas turbine engine component, the method comprising determining an area or areas of high tip vibratory stress and machining, cutting, or otherwise forming at least one recess or scallop extending into and circumferentially completely through the airfoil tip in the area or areas of high tip vibratory stress.

28. The as claimed in Clause 27 wherein the recess or scallop is formed to a maximum depth of a few mills in a range of about .005 - .01 inches as measured from a nominal tip edge without the recess or scallop.

29. The as claimed in Clause 27 wherein:

the airfoil tip and airfoil are from pluralities of airfoil tips and airfoils from a single stage or circumferential row of blades or vanes surrounded by annular tip seals,

the determining an area or areas of high tip vibratory stress includes running or rotating a rotor containing the blades or an annular tip seal surrounding the vanes respectively and observing nicks or scratches in the annular seal or airfoil tips, and

forming the recesses or scallops in area or areas of high tip vibratory stresses adjacent the nicks or scratches.

Claims

1. A gas turbine engine component (200, 203, 84) comprising:

an airfoil (54, 225, 88) extending from an airfoil base (56, 228, 91) to an airfoil tip (58, 138, 86) at a free end (87) of the airfoil (54, 225, 88),
the airfoil (54, 225, 88) extending downstream from a leading edge (LE) to a trailing edge (TE) of the airfoil (54, 225, 88),
at least one recess (92) extending into and circumferentially completely through the airfoil tip (58, 138, 86), and
the recess (92) located inwardly of the leading and trailing edges (LE, TE) of the airfoil (54, 225, 88).

2. The component as claimed in Claim 1, wherein the recess (92) is located in an area (93) of the airfoil tip (58, 138, 86) subject to high tip vibratory stress and or rubs.

3. The component as claimed in either of Claim 1 or 2, wherein the recess (92) is a circular scallop (94) having a scallop radius (SR).

4. The component as claimed in any preceding Claim, wherein the recess (92) has a maximum depth (D) in a range of 0.127-0.254mm (.005 - .01 inches) as measured from a nominal tip edge (106, 108) without the recess.

5. A gas turbine engine component as claimed in any preceding claim, wherein the component is a compressor blade (200, 84),
the airfoil (54, 88) extending radially outwardly from the airfoil base (56, 91) and wherein the airfoil base (56, 91) is located on a gas turbine engine rotor (12).

6. The gas turbine engine component of Claim 5, wherein the compressor blade (200, 84) is a radial compressor blade (200) or an impeller compressor blade (84).

7. A gas turbine engine component as claimed in any of claims 1 to 4, wherein the component is a stator vane (204),
the airfoil (225) extending radially inwardly from the airfoil base (228) and wherein the airfoil base (228) is located at an outward end (234) of the stator vane (204) and the airfoil tip (138) is located at a radial inward end (236) of the stator vane (204).

8. A gas turbine engine assembly (10) comprising the gas turbine engine component (200, 203, 84) of Claim 1, wherein the component comprises a plurality of the airfoils (54, 225, 88) and wherein
an airfoil tip clearance (140) is present between the airfoil tips (142) and an annular tip seal (144) surrounding the airfoil tips (142).

9. The assembly (10) as claimed in Claim 8, further comprising:

a plurality of blades (200) including the airfoils (54, 88),
the airfoils (54, 225, 88) extending radially outwardly from the airfoil base (56, 228, 91) to the airfoil tip (58, 138, 86),
the airfoil base (56, 91) mounted on a gas turbine engine rotor (12), and
wherein the annular tip seal (144) is a blade rub land (62).

10. The assembly (10) as claimed in Claim 8, further comprising:

the airfoils (225) being vane airfoils (225) on stator vanes (204) cantilevered from and fixed to an engine casing (110) at radial outward ends (234) of the stator vane (204),
the airfoils (225) unsupported at free radial inward ends (236) of the airfoils (225), and
wherein the annular tip seal (144) is a rotor seal land (232) on a rotor (12) of the assembly.

11. The assembly (10) as claimed in Claim 8, further comprising:

the airfoils (88) being impeller airfoils (88) extending outwardly from impeller airfoil bases (91) on a rotor disc portion (82) of an impeller (32) to impeller airfoil tips (86) at free ends (87) of the impeller airfoils (88), and
wherein the annular tip seal (144) is an annular centrifugal blade tip shroud (90).

12. A method of reducing vibratory stress at an airfoil tip (58, 138, 86) at a free end (87) of an airfoil (54, 225, 88) of a gas turbine engine component (200, 203, 84), the method comprising determining an area or areas of high tip vibratory stress and machining, cutting, or otherwise forming at least one recess (92) or scallop (94) extending into and circumferentially completely through the airfoil tip (58, 138, 86) in the area or areas of high tip vibratory stress.

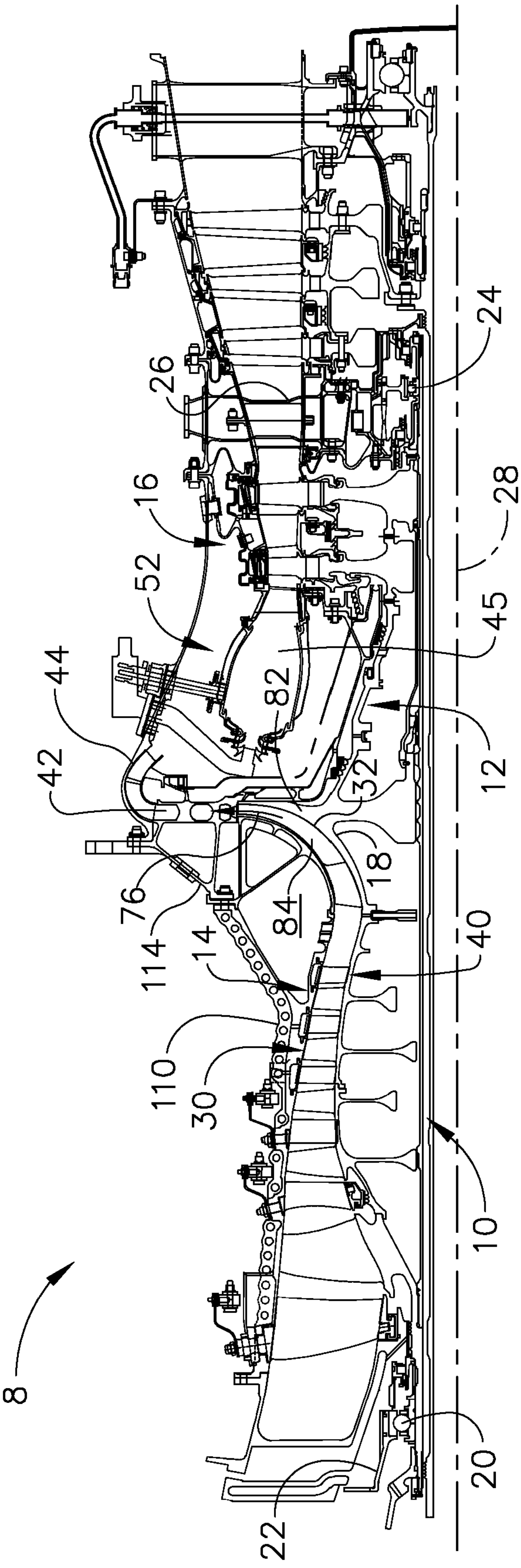

FIG. 1

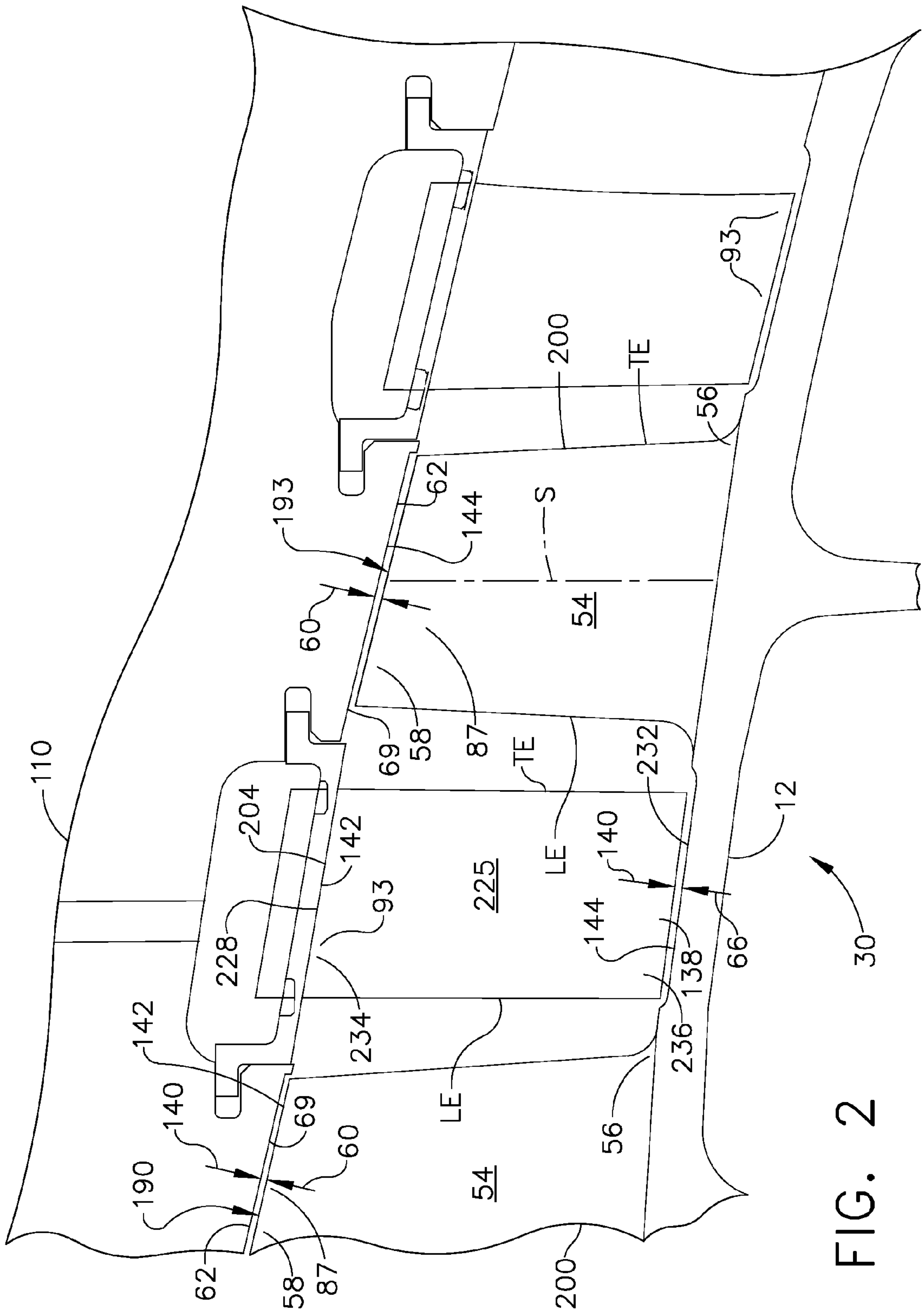
110
193
144
62
S
54
60
200
TE
56
93
69
58
87
204
142
225
228
234
LE
232
140
144
138
236
12
66
30
190
62
58
87
54
200
56

FIG. 2

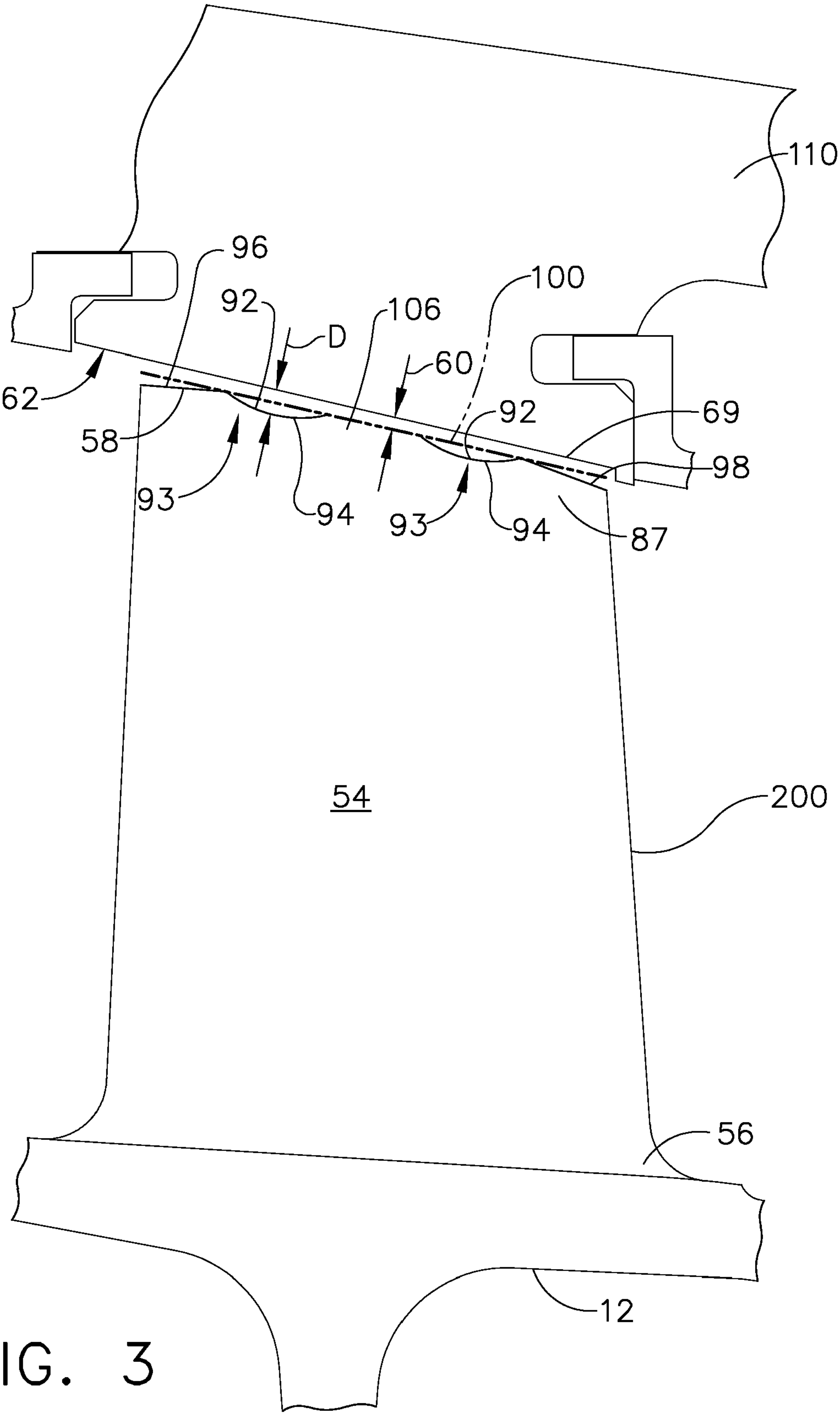

FIG. 3

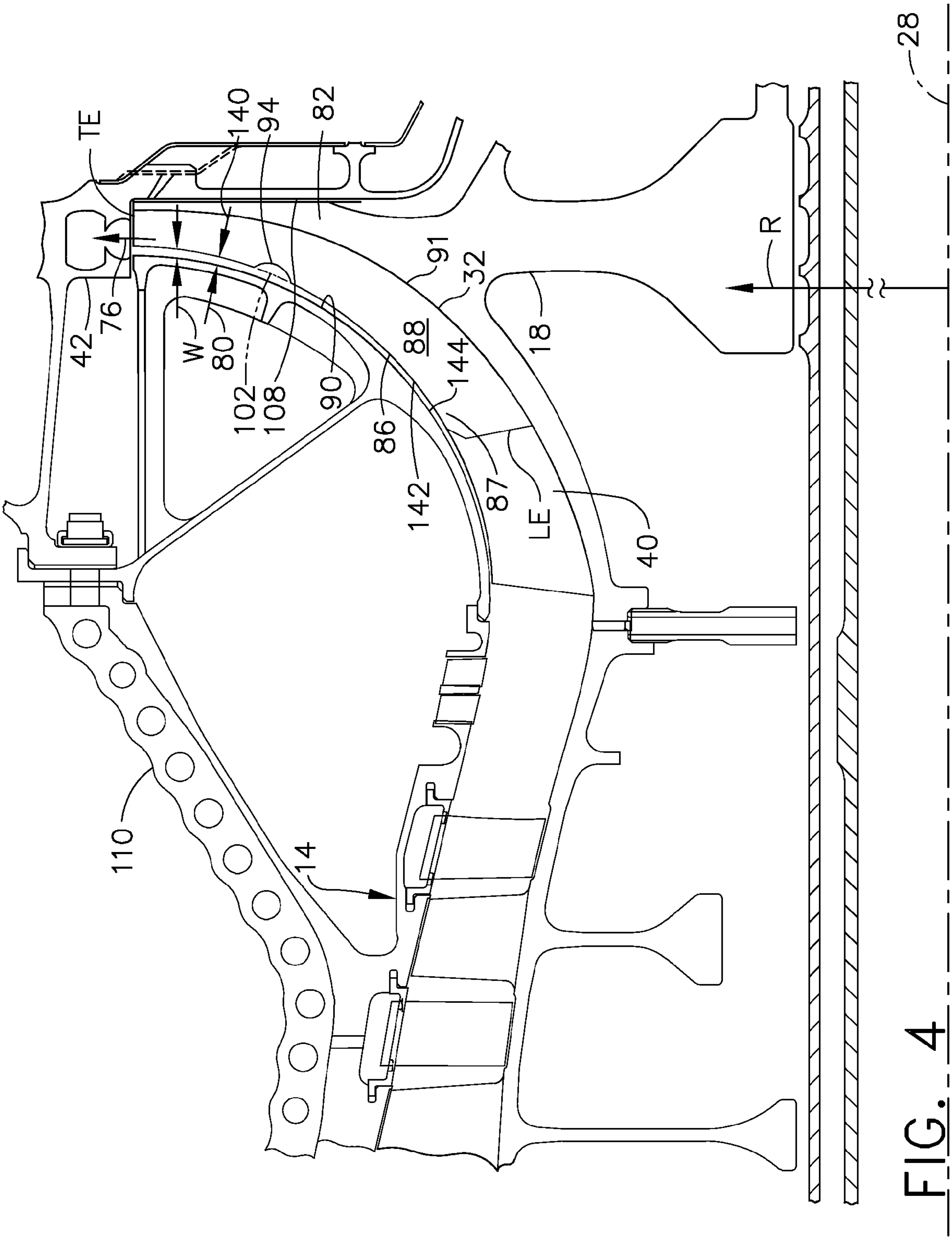
TE
140
94
82
42
76
W
80
102
108
90
86
88
142
144
87
LE
40
91
32
18
R
28
110
14

FIG. 4

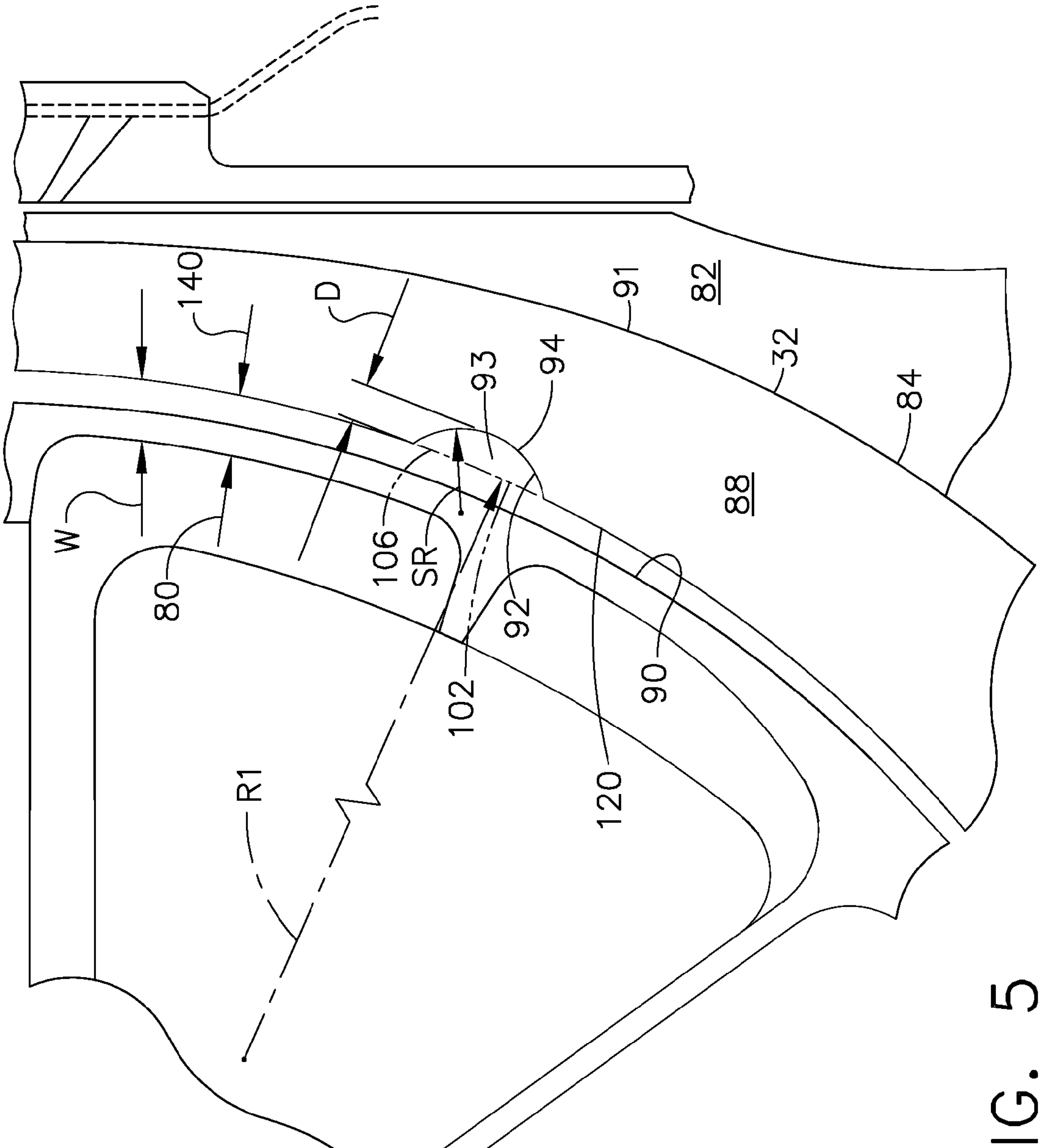

FIG. 5

Europäisches Patentamt
European Patent Office
Office européen des brevets

EUROPEAN SEARCH REPORT

Application Number
EP 12 18 7063

DOCUMENTS CONSIDERED TO BE RELEVANT

Category	Citation of document with indication, where appropriate, of relevant passages	Relevant to claim	CLASSIFICATION OF THE APPLICATION (IPC)
X Y	EP 1 101 947 A2 (GEN ELECTRIC [US]) 23 May 2001 (2001-05-23) * figures 1, 2, 5, 6 * * claims 8, 9 * * column 5, paragraph 40 - column 6, paragraph 41 * -----	1-3,5,6, 8,9,12 7,10	INV. F01D5/20
X	GB 2 427 901 A (ROLLS ROYCE PLC [GB]) 10 January 2007 (2007-01-10) * abstract; figures 1-3 * -----	1,4-6,8, 9	
X	US 2003/059309 A1 (SZUCS PETER NICHOLAS [US] ET AL) 27 March 2003 (2003-03-27) * abstract; figures 1-3 * -----	1,5,6,8, 9	
X	JP 2004 092533 A (MITSUBISHI HEAVY IND LTD) 25 March 2004 (2004-03-25) * figures 1, 4, 9 * -----	1,2,5,6, 8,9,11	
Y	DE 35 00 692 A1 (MTU MUENCHEN GMBH [DE]) 17 July 1986 (1986-07-17) * figure 1 * * page 10, line 1 - page 10, line 13 * -----	7,10	TECHNICAL FIELDS SEARCHED (IPC) F01D

The present search report has been drawn up for all claims

1

Place of search	Date of completion of the search	Examiner
Munich	29 January 2013	Lutoschkin, Eugen

CATEGORY OF CITED DOCUMENTS

X : particularly relevant if taken alone
Y : particularly relevant if combined with another document of the same category
A : technological background
O : non-written disclosure
P : intermediate document

T : theory or principle underlying the invention
E : earlier patent document, but published on, or after the filing date
D : document cited in the application
L : document cited for other reasons

& : member of the same patent family, corresponding document

EPO FORM 1503 03.82 (P04C01)

ANNEX TO THE EUROPEAN SEARCH REPORT ON EUROPEAN PATENT APPLICATION NO. EP 12 18 7063

This annex lists the patent family members relating to the patent documents cited in the above-mentioned European search report.
The members are as contained in the European Patent Office EDP file on
The European Patent Office is in no way liable for these particulars which are merely given for the purpose of information.

29-01-2013

Patent document cited in search report	Publication date	Patent family member(s)	Publication date
EP 1101947 A2	23-05-2001	AT 333591 T	15-08-2006
		DE 60029405 T2	15-02-2007
		EP 1101947 A2	23-05-2001
		ES 2267465 T3	16-03-2007
		IL 137862 A	24-06-2003
		JP 2001182694 A	06-07-2001
		US 6234747 B1	22-05-2001
GB 2427901 A	10-01-2007	GB 2427901 A	10-01-2007
		US 2007098562 A1	03-05-2007
US 2003059309 A1	27-03-2003	NONE	
JP 2004092533 A	25-03-2004	NONE	
DE 3500692 A1	17-07-1986	NONE	

For more details about this annex : see Official Journal of the European Patent Office, No. 12/82