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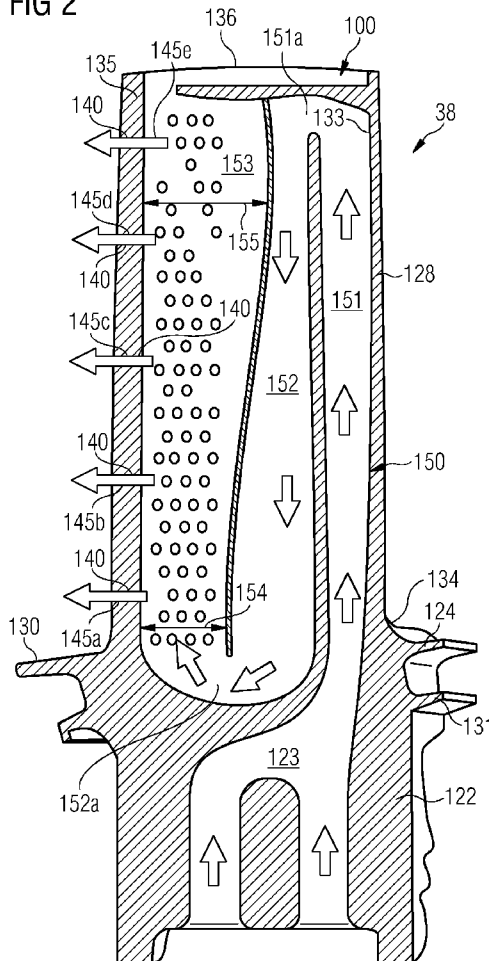
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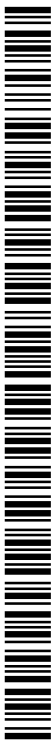
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(54) Title: GAS TURBINE AEROFOIL

FIG 2



(57) Abstract: An aerofoil (100, 101, 102) for a gas turbine (10) comprises: an external surface (128) axially extending between a leading edge (133) and a trailing edge (135) and radially extending between a base (134) and a tip (136); an internal passage (150) for channelling a cooling medium to a plurality of outlet cooling holes (140, 141, 142) the internal passage comprising a cavity (153, 163, 173) having a variable cross-sectional area and communicating with at least a portion of the outlet cooling holes (140, 141, 142).





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DESCRIPTION

Gas turbine aerofoil

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Field of invention

The present invention relates to gas turbine aerofoils and, more particularly, to gas turbine aerofoil provided with
10 internal cooling passages.

Art Background

In a gas turbine engine, air is pressurized in a compressor
15 and mixed with fuel in a combustor for generating hot combustion gases. The hot gases are then channelled towards a turbine which transforms the energy from the hot gases into work for powering the compressor and other devices which converts power, for example an upstream fan in a typical
20 aircraft turbofan engine application, or a generator in power generation application.

The turbine stages include stationary turbine nozzles having a row of vanes which channel the combustion gases into a
25 corresponding row of rotor blades extending radially outwardly from a supporting rotor disk. The vanes and blades may have corresponding hollow aerofoils. Aerofoils may be designed and manufactured hollow in order to save weight, to change its eigenfrequency or to include a cooling circuit
30 therein. In the latter case, the cooling gas which circulates inside the cooling circuit or circuits is typically bleed air from the compressor discharge.

The present invention relates to a turbine hollow aerofoil
35 which can be used in a gas turbine vane or blade and which include an internal passage for cooling the aerofoil.
The external surface of a turbine hollow aerofoil is also

typically provided with a plurality of outlet cooling holes. The cooling gas from the inside of the hollow aerofoil flows through the outlet cooling holes forming a cooling film on the external surface of the turbine hollow .

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For a set cooling hole size, the pressure ratio between inlet and outlet of the cooling hole is the principle factor that determines the mass flow rate and hence the effectiveness of the cooling. The exit pressure is determined by
10 mainstream gas flow conditions, and cannot therefore be easily varied. The inlet pressure is determined by the cooling air supply pressure to the component and by the pressure losses encountered by the flow up to the entry to the cooling hole. The inlet pressure may be further
15 influenced by pressure gains due to the pumping effect induced by the machine rotation. Hence the film cooling mass flow rate is determined by several variables which cannot usually be varied.

20 It is currently known to control the variation of film cooling flow in a specific region of the aerofoil by controlling the size and number of the cooling holes in that region. However, in some case the size and number of the outlet cooling holes may be restricted by casting, machining,
25 or stress limitations.

EP2902589A1 discloses an aerofoil for a gas turbine comprising an internal passage for channelling coolant and an insert with a variable cross section.

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US7955053B1 discloses a turbine blade having an aft flowing triple pass serpentine cooling circuit with all convection cooled blade. The three passes or legs of the serpentine flow circuit are formed by a leading edge rib and a trailing edge
35 rib that are both slanted in order to provide decreasing flow cross sectional areas in the three passes of legs.

EP0207799A2 discloses a combustion turbine rotor blade provided with an airfoil portion having a plurality of coolant holes extending radially outwardly therethrough. The coolant holes are tapered to a smaller flow cross-section in the radially outward direction to produce more uniform cooling action over the length of the holes over which tapering is provided.

It is therefore desirable to provide a new design for gas turbine aerofoils where the pressure in the cavity at the entry to the cooling hole may be changed, hence varying the film cooling mass flow rate and the cooling effectiveness as required by the cooling design criteria, independently from the size and number of the outlet cooling holes.

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Summary of the Invention

It may be an object of the present invention to provide a gas turbine aerofoil where the film cooling mass flow rate and the cooling effectiveness as required by the cooling design criteria is varied without changing the size and number of the outlet cooling holes.

In order to achieve the object defined above, an aerofoil for a gas turbine is provided in accordance to the independent claim. The dependent claims describe advantageous developments and modifications of the invention.

According to an aspect of the present invention, an aerofoil for a gas turbine comprises:

- an external surface axially extending between a leading edge and a trailing edge and radially extending between a base and a tip;
- an internal passage for channelling a cooling medium to a plurality of outlet cooling holes the internal passage comprising a cavity communicating with at least a portion of the outlet cooling holes. The cavity has a variable cross-

sectional area.

Variations of the cross-sectional area at a specific portion of the outlet cooling holes is used to increase or decrease the internal pressures and hence the film cooling mass flow rate in a region of the aerofoil affected by that specific portion of the outlet cooling holes. Therefore, the cooling effectiveness in that region can be controlled without changing the size and number of the outlet cooling holes.

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According to an exemplary embodiment of the present invention, the cavity extends radially between the base and the tip of the aerofoil. The outlet cooling holes may be provided on the external surface of the hollow aerofoil, in particular close to the trailing edge and/or to the leading edge.

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Advantageously, this permits to control the cooling film on the external side of the hollow aerofoil.

20

According to another exemplary embodiment of the present invention, the internal passage comprises a plurality of cavities in series, the last cavity of the plurality of cavities communicating with at least a portion of the outlet cooling holes provided on the external surface of the hollow aerofoil.

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In all the embodiments defined above, the cavity may comprise at least a first section having a first cross-sectional area and a second section having a second cross-sectional area, the second cross-sectional area being greater than the first cross-sectional area. The second section may be closer to the tip of the aerofoil and the first section may be closer to the base, in such a way that the increase of the cross-sectional area compensates for the pressure losses along the flow path of the cooling gas. Particularly, the values of the first, second and third cross-sectional areas may be defined

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in such a way that the flow rate through at least a portion of the outlet cooling holes is constant and equal to a predefined desired value.

5 According to an alternative embodiment, the cavity comprises a third section having a third cross-sectional area, the second cross-sectional area being greater than the first cross-sectional area and greater than the third cross-sectional area. The first section is closer to the base, the
10 third section is closer to the tip and the second section is intermediate between the first section and the third section.

According to yet another exemplary embodiment of the present invention, the outlet cooling holes are provided on an
15 external surface of an impingement tube extending radially inside the hollow aerofoil, the cavity, which communicates with at least a portion of the outlet cooling holes, extending radially inside the impingement tube. An insert may be provided inside the impingement tube, in such a way that
20 the cavity is comprised between the insert and the external surface of the impingement tube, the insert having a shape such that the cavity has a variable cross-sectional area. Advantageously, this permits to locally increasing or decreasing the impingement flow, thus controlling the amount
25 of cooling to the target wall, i.e. to an internal surface of the hollow aerofoil.

At least a part of the cavity may vary in cross-sectional area at a constant rate between or from the first section and
30 the second section and/or where applicable between the second section and the third section.

At least a part of the cavity may vary in cross-sectional area at a non-constant rate between or from the first section
35 and the second section and/or where applicable between the second section and the third section.

The non-constant rate of change of area of the cavity may comprise a sudden or step changes in the area.

The internal passage may be at least partly formed by the
5 aerofoil and the cooling holes extend from the internal passage to the external surface.

Brief Description of the Drawings

10 The above mentioned attributes and other features and advantages of this invention and the manner of attaining them will become more apparent and the invention itself will be better understood by reference to the following description of embodiments of the invention taken in conjunction with the
15 accompanying drawings, wherein:

FIG. 1 shows part of a turbine engine in a sectional view and in which the present inventive aerofoil is incorporated,

20 FIG. 2 shows a longitudinal sectional view of a gas turbine aerofoil (a blade) according to the present invention,

FIG. 3 shows a longitudinal sectional view of another embodiment of a gas turbine aerofoil according to the present
25 invention,

FIG. 4 shows a cross-sectional view of the gas turbine aerofoil of fig. 3, taken along the sectional line IV-IV in Fig. 3,
30

FIG. 5 shows an isometric partially-sectioned view of yet another embodiment of a gas turbine aerofoil according to the present invention,

35 FIG. 6 shows a detailed view of the gas turbine aerofoil of fig. 5.

Detailed Description

Hereinafter, above-mentioned and other features of the present invention are described in details. Various
5 embodiments are described with reference to the drawings, wherein the same reference numerals are used to refer to the same elements throughout. The illustrated embodiments are intended to explain, and not to limit the invention.

10 FIG. 1 shows an example of a gas turbine engine 10 in a sectional view. The gas turbine engine 10 comprises, in flow series, an inlet 12, a compressor section 14, a combustor section 16 and a turbine section 18 which are generally
15 arranged in flow series and generally about and in the direction of a longitudinal or rotational axis 20. The gas turbine engine 10 further comprises a shaft 22 which is rotatable about the rotational axis 20 and which extends longitudinally through the gas turbine engine 10. The shaft 22 drivingly connects the turbine section 18 to the
20 compressor section 14.

In operation of the gas turbine engine 10, air 24, which is taken in through the air inlet 12 is compressed by the compressor section 14 and delivered to the combustion section
25 or burner section 16. The burner section 16 comprises a burner plenum 26, one or more combustion chambers 28 and at least one burner 30 fixed to each combustion chamber 28. The combustion chambers 28 and the burners 30 are located inside the burner plenum 26. The compressed air passing through the
30 compressor 14 enters a diffuser 32 and is discharged from the diffuser 32 into the burner plenum 26 from where a portion of the air enters the burner 30 and is mixed with a gaseous or liquid fuel. The air/fuel mixture is then burned and the combustion gas 34 or working gas from the combustion is
35 channelled through the combustion chamber 28 to the turbine section 18 via a transition duct 17.

This exemplary gas turbine engine 10 has a cannular combustor section arrangement 16, which is constituted by an annular array of combustor cans 19 each having the burner 30 and the combustion chamber 28, the transition duct 17 has a generally
5 circular inlet that interfaces with the combustor chamber 28 and an outlet in the form of an annular segment. An annular array of transition duct outlets form an annulus for channelling the combustion gases to the turbine 18.

10 The turbine section 18 comprises a number of blade carrying discs 36 attached to the shaft 22. In the present example, two discs 36 each carry an annular array of turbine blades 38. However, the number of blade carrying discs could be different, i.e. only one disc or more than two discs. In
15 addition, guiding vanes 40, 44 (a first stage of guiding vanes 44 and a second stage of guiding vanes 40), which are fixed to a stator 42 of the gas turbine engine 10, are disposed between the stages of annular arrays of turbine blades 38, 60 (a first stage of turbine blades 60 and a
20 second stage of turbine blades 38). Between the exit of the combustion chamber 28 and the leading turbine blades 60 inlet guiding vanes 40, 44 are provided and turn the flow of working gas onto the turbine blades 38, 60.

25 The combustion gas from the combustion chamber 28 enters the turbine section 18 and drives the turbine blades 38 which in turn rotate the shaft 22. The guiding vanes 40, 44 serve to optimise the angle of the combustion or working gas on the turbine blades 38.

30 The turbine section 18 drives the compressor section 14. The compressor section 14 comprises an axial series of vane stages 46 and rotor blade stages 48. The rotor blade stages 48 comprise a rotor disc supporting an annular array of
35 blades. The compressor section 14 also comprises a casing 50 that surrounds the rotor stages and supports the vane stages 48. The guide vane stages include an annular array of

radially extending vanes that are mounted to the casing 50.

The vanes are provided to present gas flow at an optimal angle for the blades at a given engine operational point.

5 Some of the guide vane stages have variable vanes, where the angle of the vanes, about their own longitudinal axis, can be adjusted for angle according to air flow characteristics that can occur at different engine operations conditions.

10 The casing 50 defines a radially outer surface 52 of the passage 56 of the compressor 14. A radially inner surface 54 of the passage 56 is at least partly defined by a rotor drum 53 of the rotor which is partly defined by the annular array of blades 48.

15

The present invention is described with reference to the above exemplary turbine engine having a single shaft or spool connecting a single, multi-stage compressor and a single, one or more stage turbine. However, it should be appreciated that
20 the present invention is equally applicable to two or three shaft engines and which can be used for industrial, aero or marine applications.

The terms upstream and downstream refer to the flow direction
25 of the airflow and/or working gas flow through the engine unless otherwise stated. The terms forward and rearward refer to the general flow of gas through the engine. The terms axial, radial and circumferential are made with reference to the rotational axis 20 of the engine.

30

Inventive embodiments of a gas turbine aerofoil 100, 101, 102 are shown in **Figs. 2 to 6**.

The inventive turbine aerofoil 100 may be used, in general,
35 in the turbine blades 38, 60 or in the turbine guiding vanes 40, 44.

With reference to **Fig. 2** a first embodiment of a gas turbine aerofoil 100 is shown, applied to a turbine blade 38.

The rotor blade 38 comprises:

- 5 - a root 122 for connecting the blade 38 to the disk 36 of the rotor 22, ,
- a platform 124 having a lower surface 131, from which the root 122 extends and an upper surface 130 opposite to the lower surface 131,
- 10 - the hollow aerofoil 100 extending from the upper surface 130 of the platform 124.

The aerofoil 100 has an external surface 128. The external surface 128 axially extends between a leading edge 133 and a trailing edge 135 and radially extends between a base 134 and a tip 136. On the external surface 128 a first plurality of outlet cooling holes 140 are provided in a portion of the hollow aerofoil 100 including the trailing edge 135. The cooling medium passes through the external surface 128 via the cooling holes to form a film of cooling medium over the external surface of the aerofoil. These cooling holes 140 and the other cooling holes in the external surface of the aerofoil are known as film cooling holes or effusion cooling holes.

25 The aerofoil 100 is hollow and further comprises an internal passage 150 for channelling a cooling medium (for example compressed air from the compressor section 14) to the first plurality of outlet cooling holes 140. The cooling medium flows exiting the first plurality outlet cooling holes 140 is represented by a respective first plurality of arrows 145a, 145b, 145c, 145d, 145e.

35 The internal passage 150 of the aerofoil 100 is connected to a root passage 123, provided in the root 122 of the rotor blade 38. The root passage 123 provides a connection between the internal passage 150 and a source (not shown) of the

cooling medium.

The internal passage 150 comprises a plurality of cavities (three cavities 151, 152, 153) disposed in series between the root passage 123 and the first plurality of outlet cooling holes 140. Each of the cavities 151, 152, 153 extends radially between the base 134 and the tip 136 of the aerofoil 100.

10 The first cavity 151 is connected to the root passage 123 and extends radially from the base 134 to the tip 136, adjacently to the leading edge 133.

The second cavity 152 is connected to the first cavity 151 by means of a first curve 151a of the passage 150, provided at the tip 136. The second cavity 152 extends radially from the tip 136 to the base 134.

The third cavity 153 is connected to the second cavity 152 by means of a second curve 152a of the passage 150, provided at the base 134. The third cavity 153 extends radially from the base 134 to the tip 136, adjacently to the trailing edge 135. The third cavity 153 communicates with the first plurality of outlet cooling holes 140, thus permitting the cooling medium to exit the external surface 128 for cooling the aerofoil 100 at the trailing edge 135.

The third cavity 153 has a variable cross-sectional area. The third cavity 153 comprises a first section 154 and a second section 155, both orthogonal to the radial direction. The first section 154 and the second section 155 are disposed relatively to each other in such a way that the first section 154 is closer to the base 134 and the second section 155 is closer to the tip 136 of the aerofoil 100.

35 The first section 154 has a first cross-sectional area and the second section 155 having a second cross-sectional area,

the second cross-sectional area being greater than the first cross-sectional area.

According to the Bernoulli principle, pressure and velocity
5 in a moving flow are inversely proportional. Therefore, at
the second section 155 the local velocity decreases with
respect to the first section 154. The pressure
correspondently increases and the flow through the respective
cooling hole 140 increases. The increase of the cross-
10 sectional area in the third cavity 153 from the base 134 to
the tip 136 compensate for the pressure losses along the flow
path of the cooling medium.

The values of the cross-sectional areas along the third
15 cavity 153 may be preferably arranged in such a way that the
flow rate of the cooling medium flows 145a, 145b, 145c, 145d,
145e through the first plurality of cooling holes 140 is
constant and equals a predefined desired value. As an
alternative, the flow rate of the cooling medium flows 145a,
20 145b, 145c, 145d, 145e through the first plurality of cooling
holes 140 may vary as required by design needs.

According to another embodiment of the present invention (not
shown) the gas turbine aerofoil 100 of **Fig. 2** may be applied
25 to a guiding vane 40.

With reference to **Figs. 3** and **4**, a second embodiment of a gas
turbine aerofoil 101 is shown, applied to a turbine blade 38.
On the external surface 128 of the turbine aerofoil 101 a
30 second plurality of outlet cooling holes 141 are provided in
a portion of the hollow aerofoil 101 including the leading
edge 133.

The internal passage 150 comprises a cavity 163 directly
35 connected to the root passage 123 and extending radially
between the base 134 and the tip 136 of the aerofoil 101,
adjacently to the leading edge 133.

The cavity 163 communicates with the second plurality of outlet cooling holes 141, thus permitting the cooling medium to exit the external surface 128 for cooling the aerofoil 101 at the leading edge 133.

The cavity 163 has a variable cross-sectional area. The cavity 163 comprises a first section 154 having a first cross-sectional area, a second section 155 having a second cross-sectional area and a third section 156 having a third cross-sectional area. The first, second and third section 154, 155, 156 are all orthogonal to the radial direction.

The first, second and third section 154, 155, 156 are disposed relatively to each other in such a way that the first section 154 is closer to the base 134, the third section 156 is closer to the tip 136 and the second section 155 is intermediate between the first section 154 and the third section 156. The second cross-sectional area is greater than the first cross-sectional area and greater than the third cross-sectional area.

The values of the cross-sectional areas along the cavity 163 are arranged in such a way that the flow rate of the cooling medium flows 145c through holes of the second plurality of cooling holes 141 which are intermediate between the base 134 and the tip 136 is greater than the flow rate of the cooling medium flows 145a, 145b which are closer to the base 134 or than the flow rate of the cooling medium flows 145d, 145e which are closer to the tip 136. Hence, flow rates through holes 145a, 145b, 145c, 145d, 145e may be varied as necessary for optimum design by varying cross-section areas.

According to another embodiment of the present invention (not shown) the gas turbine aerofoil 101 of **Figs. 3** and **4** may be applied to a guiding vane 40.

According to a further embodiment of the present invention (not shown) the gas turbine aerofoil 100 and 101 of **Figs. 2** and **2 to 4**, respectively may be combined in the same the turbine blade 38 or guiding vane 40, i.e. both the cavities
5 153 and 163 may be present in the same aerofoil for controlling the flow rate of the cooling medium flows at trailing edge 135 and at the leading edge 133.

With reference to **Figs. 5** and **6**, a third embodiment of a gas
10 turbine aerofoil 102 is shown, applied to a turbine blade 38. The turbine aerofoil 102 comprises, inside the external surface 128, an impingement tube 170 radially extending between the base 134 and the tip 136. On the external surface 174 of the impingement tube 170 a third plurality of outlet
15 cooling holes 142 are provided.

The internal passage 150 comprises a cavity 173 inside the impingement tube 170 directly connected to the root passage 123 and extending radially between the base 134 and the tip
20 136 of the aerofoil 102.

The cavity 173 communicates with the second plurality of outlet cooling holes 142, thus permitting the cooling medium to exit the external surface 174 of the impingement tube 170
25 for cooling from the inside a target portion of the external surface 128 of the aerofoil 102.

The cavity 173 has a variable cross-sectional area. For varying the cross-sectional area, an insert 171 is provided
30 inside the impingement tube 170, in such a way that the cavity 173 is therefore comprised between the insert 171 and the external surface 174 of the impingement tube 170.

The insert 171 has a smaller thickness in a region
35 intermediate between the base 134 and the tip 136. Consequently, the cavity 173 comprises a first section 154 having a first cross-sectional area, a second section 155

having a second cross-sectional area and a third section 156 having a third cross-sectional area. The first, second and third section 154, 155, 156 are all orthogonal to the radial direction. The first, second and third section 154, 155, 156 are disposed relatively to each other in such a way that the first section 154 is closer to the base 134, the third section 156 is closer to the tip 136 and the second section 155 is intermediate between the first section 154 and the third section 156. The second cross-sectional area is greater than the first cross-sectional area and greater than the third cross-sectional area.

The values of the cross-sectional areas along the cavity 173 are arranged in such a way that the flow rate of the cooling medium flows 145c through holes of the second plurality of cooling holes 142 which are intermediate between the base 134 and the tip 136 is greater than the flow rate of the cooling medium flows 145a, 145b which are closer to the base 134 or than the flow rate of the cooling medium flows 145d, 145e which are closer to the tip 136. Hence, flow rates through holes 145a, 145b, 145c, 145d, 145e may be varied as necessary for optimum design by varying cross-section areas.

According to another embodiment of the present invention (not shown) the gas turbine aerofoil 102 of **Figs. 5** and **6** may be applied to a guiding vane 40.

According to other embodiments of the present invention (not shown) values of the cross-sectional areas along the cavities 153 or 163 or 173 of the turbine aerofoil 100, 101, 102, respectively are arranged in order to increase or decrease the flow rate of the cooling medium flows through a specific portion of the outlet cooling holes in order to respectively increase or decrease the cooling effectiveness in a specific region of the turbine aerofoil 100, 101, 102.

It should be appreciated throughout the description that the

direction of cooling medium is in a direction from the first section 154 towards the second section 155 and that the cross-section area of the cavity increases between the first section 154 towards the second section 155. Similarly, the direction of cooling medium is in a direction from the second section 155 towards the third section 156 and that the cross-section area of the cavity decreases between the second section 155 towards the third section 156. As the cooling medium travels in an increasing cross-sectional area cavity its velocity decreases and therefore its static pressure increases. Where there is an increase in static pressure in the cavity there is a corresponding increase in the amount of cooling medium that is ejected through the film cooling holes where there is an increase in static pressure. Similarly, where there is a decrease in the cross-sectional area of the cavity the cooling medium increases in velocity and corresponding the static pressure decrease and therefore less cooling medium is ejected through the film cooling holes where there is a decrease in static pressure. Thus it is possible to control the amount of cooling medium ejected through the film cooling holes dependent on the ambient temperature traverse across the aerofoil. Less coolant can be used to improve efficiency and/or the temperature gradient of the aerofoil can be reduced to increase life of the aerofoil.

CLAIMS

- 5 1. An aerofoil (100, 101, 102) for a gas turbine (10) comprising:
- an external surface (128) axially extending between a leading edge (133) and a trailing edge (135) and radially extending between a base (134) and a tip (136);
 - 10 - an internal passage (150) for channelling a cooling medium to a plurality of outlet cooling holes (140, 141, 142) the internal passage comprising a cavity (153, 163, 173) communicating with at least a portion of the outlet cooling holes (140, 141, 142),
 - 15 wherein the cavity (153, 163, 173) has a variable cross-sectional area.
2. The hollow aerofoil (100, 101, 102) of claim 1, wherein the cavity (153, 163, 173) extends radially between the base
20 and the tip of the aerofoil.
3. The hollow aerofoil (100, 101) of claim 1 or 2, wherein the outlet cooling holes (140, 141) are provided on the external surface (128) of the hollow aerofoil (100).
25
4. The hollow aerofoil (100, 101) of claim 3, wherein the internal passage comprises a plurality of cavities in series, the last cavity (153) of said plurality of cavities communicating with at least a portion of the outlet cooling
30 holes (140, 141) provided on the external surface (128) of the hollow aerofoil (100).
5. The hollow aerofoil (100, 101) of any of the preceding claims, wherein said portion of outlet cooling holes (140)
35 are provided close to the trailing edge (135).
6. The hollow aerofoil (100, 101) of any of the preceding

claims, wherein said portion of outlet cooling holes (141) are provided close to the leading edge (133)

7. The hollow aerofoil (100, 101) of any one of claims 1
5 to 6, wherein the cavity (153, 163) comprises at least a first section (154) having a first cross-sectional area and a second section (155) having a second cross-sectional area, the second cross-sectional area being greater than the first cross-sectional area and wherein the direction of cooling
10 medium is in a direction from the first section (154) towards the second section (155).

8. The hollow aerofoil (101) of claim 7, wherein the cavity
15 (163) comprises a third section (156) having a third cross-sectional area, the second cross-sectional area being greater than the first cross-sectional area and greater than the third cross-sectional area, the second section (155) being intermediate between the first section and the third section and wherein the direction of cooling medium is in a direction
20 from the second section (155) towards the third section (156).

9. The hollow aerofoil (102) of claim 1 or 2, wherein the
25 outlet cooling holes (142) are provided on an external surface (174) of an impingement tube (170) extending radially inside the hollow aerofoil (100), the cavity (173) extending radially inside the impingement tube (170).

10. The hollow aerofoil (102) of claim 9, wherein an insert
30 (171) is provided inside the impingement tube (170), in such a way that the cavity (173) is comprised between the insert (171) and the external surface (174) of the impingement tube (170), the insert (171) having a shape such that the cavity (173) has a variable cross-sectional area.

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11. The hollow aerofoil (100) of any of the claims 7 to 8, wherein the values of the first, second and third cross-

sectional areas are defined in such a way that the flow rate through at least a portion of the outlet cooling holes (140, 141, 142) is constant.

5 12. The hollow aerofoil (100) of any of the claims 1 to 11, wherein at least a part of the cavity (153, 163, 173) varies in cross-sectional area at a constant rate between or from the first section (154) and the second section (155) and where applicable between the second section (155) and/or the
10 third section (156).

13. The hollow aerofoil (100) of any of the claims 1 to 11, wherein at least a part of the cavity (153, 163, 173) varies in cross-sectional area at a non-constant rate between or
15 from the first section (154) and the second section (155) and/or where applicable between the second section (155) and the third section (156).

14. The hollow aerofoil (100) of claim 13, wherein the non-
20 constant rate of change of area of the cavity comprises sudden or step changes in the area.

15. The hollow aerofoil (100) of any one of claims 1-14, wherein the internal passage (150) is at least partly formed
25 by the aerofoil and the cooling holes (140, 141) extend from the internal passage (150) to the external surface (128).

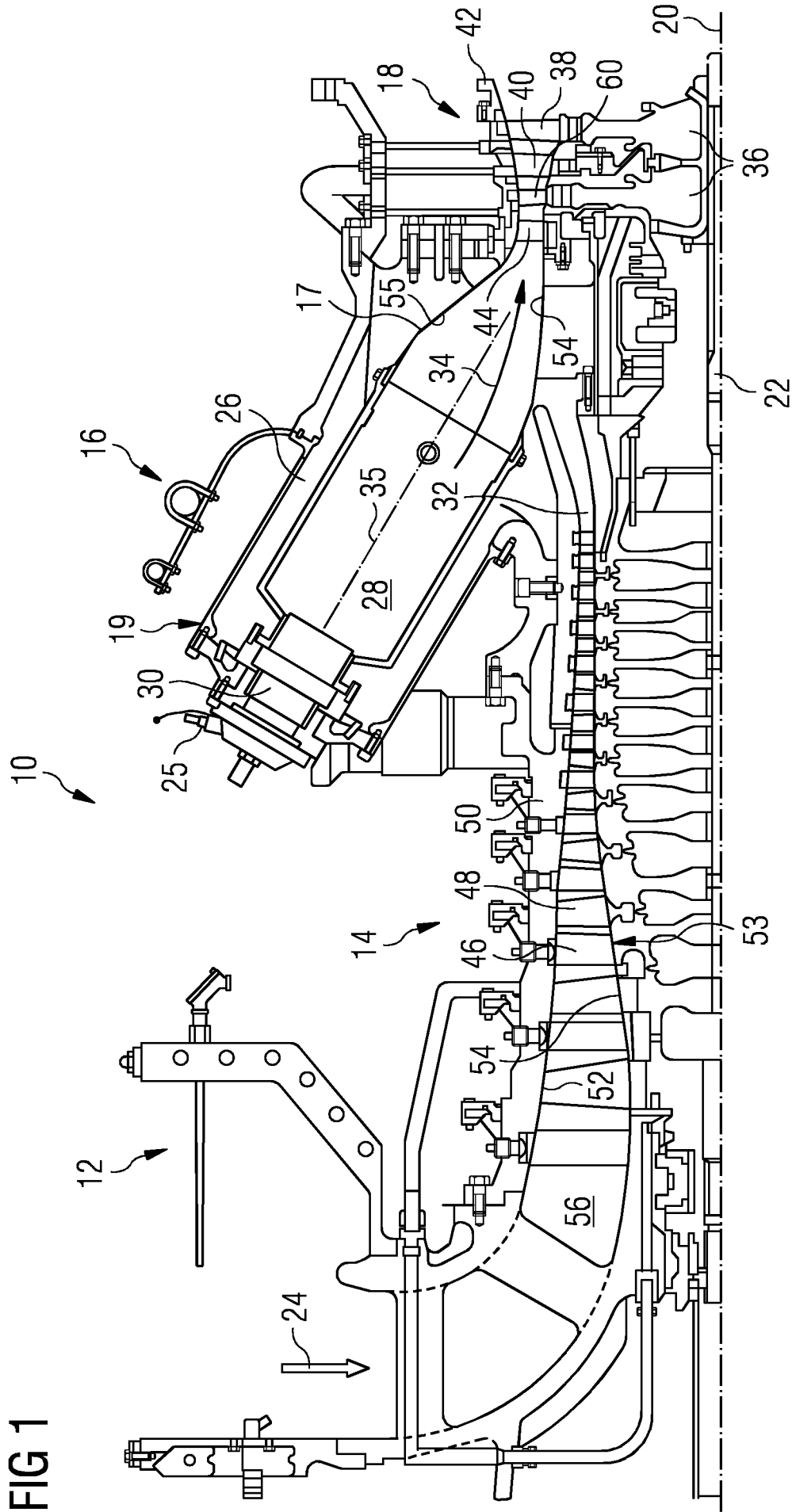


FIG 1

FIG 2

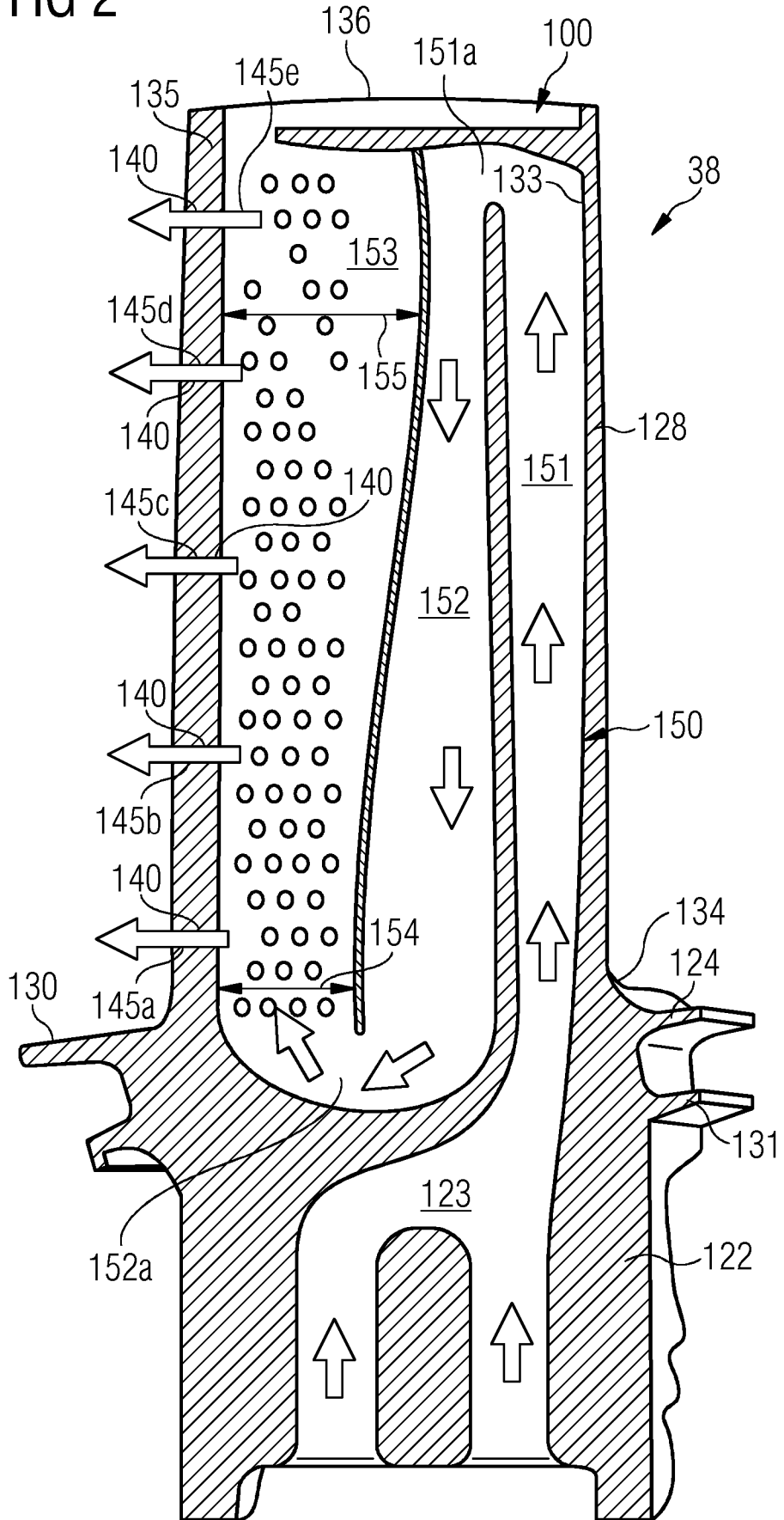


FIG 3

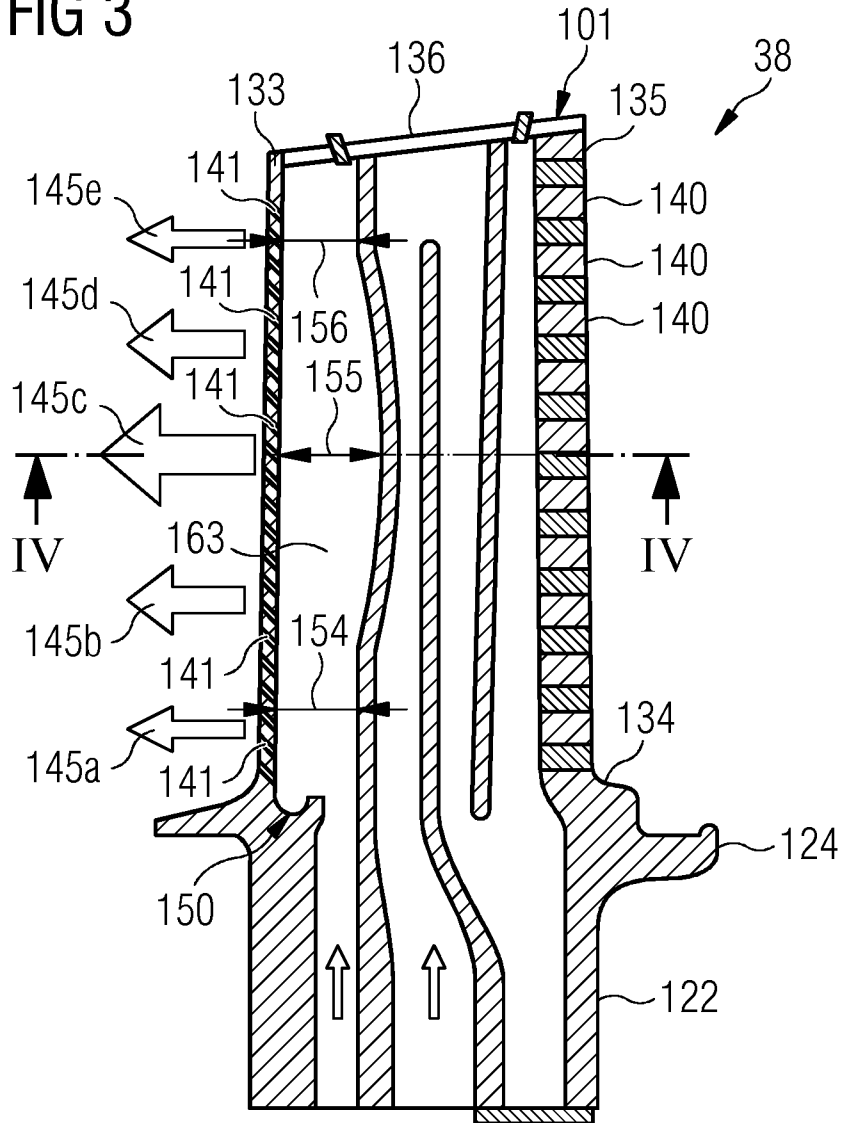


FIG 4

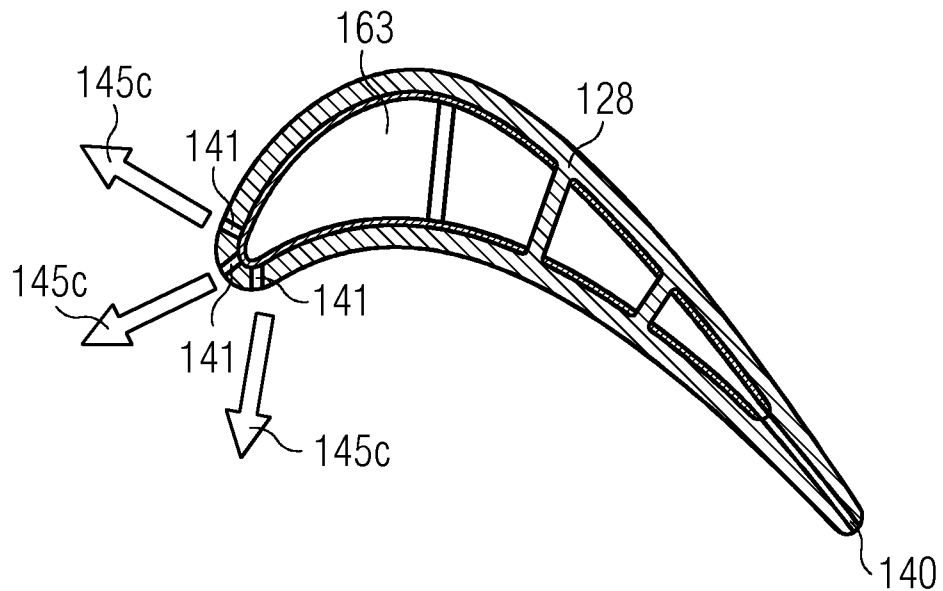


FIG 5

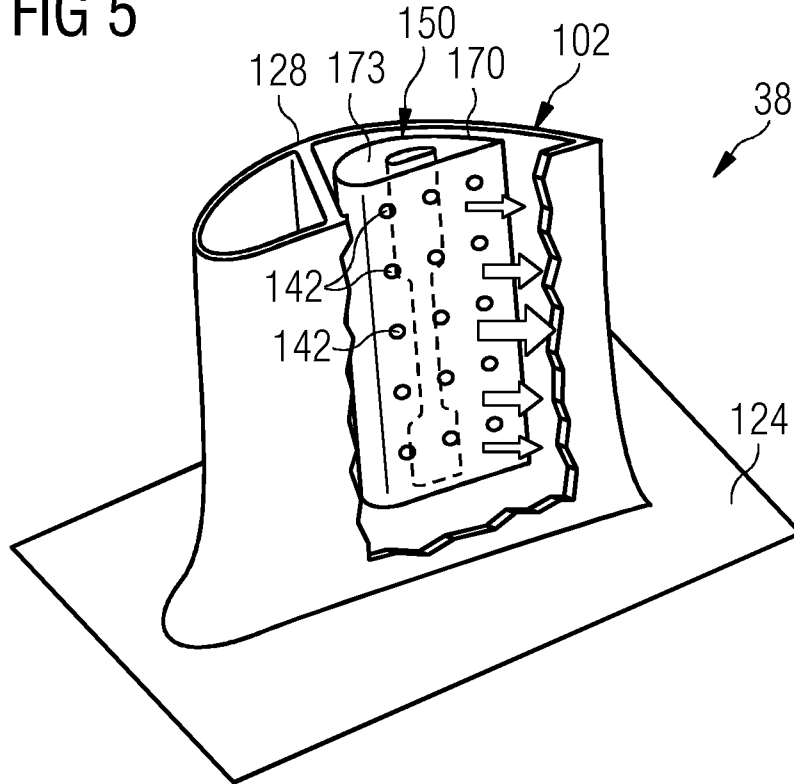
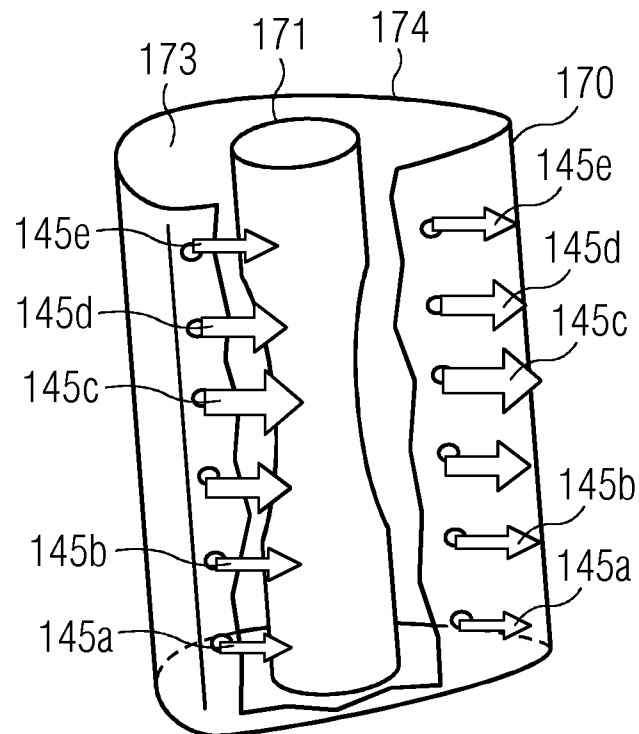


FIG 6



INTERNATIONAL SEARCH REPORT

International application No
PCT/EP2017/050276

A. CLASSIFICATION OF SUBJECT MATTER
INV. F01D5/18
ADD.

According to International Patent Classification (IPC) or to both national classification and IPC

B. FIELDS SEARCHED
Minimum documentation searched (classification system followed by classification symbols)
F01D

Documentation searched other than minimum documentation to the extent that such documents are included in the fields searched

Electronic data base consulted during the international search (name of data base and, where practicable, search terms used)
EPO-Internal, WPI Data

C. DOCUMENTS CONSIDERED TO BE RELEVANT

Category*	Citation of document, with indication, where appropriate, of the relevant passages	Relevant to claim No.
X	EP 2 902 589 A1 (SIEMENS AG [DE]) 5 August 2015 (2015-08-05) paragraph [0011]; figure 1 -----	1-6, 9-12,15
X	US 7 955 053 B1 (LIANG GEORGE [US]) 7 June 2011 (2011-06-07) column 2, line 65 - column 3, line 24; figure 1 -----	1-5,12, 13
X	EP 0 207 799 A2 (WESTINGHOUSE ELECTRIC CORP [US]) 7 January 1987 (1987-01-07) page 5, line 34 - page 6, line 22; figures 1,4 -----	1-3,7,14

Further documents are listed in the continuation of Box C.

See patent family annex.

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Date of the actual completion of the international search

13 March 2017

Date of mailing of the international search report

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INTERNATIONAL SEARCH REPORT

Information on patent family members

International application No

PCT/EP2017/050276

Patent document cited in search report	Publication date	Patent family member(s)	Publication date
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