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(54) **Serpentine microcircuits for local heat removal**

(57) A turbine engine component (90), such as a turbine blade, has an airfoil portion (104) with a pressure side and a suction side. The turbine engine component (90) also has a first cooling circuit (100) within the pressure side for cooling the pressure side of the airfoil portion

(110) and a second cooling circuit (120) within the suction side for cooling the suction side of the airfoil portion (104) and for cooperating with a wrap around leading edge cooling circuit (126) for creating a cooling film over the pressure side.

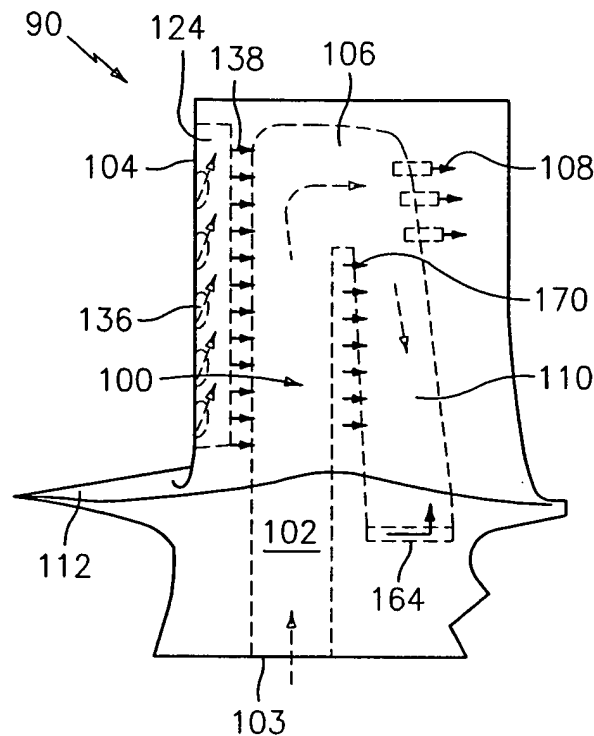


FIG. 5

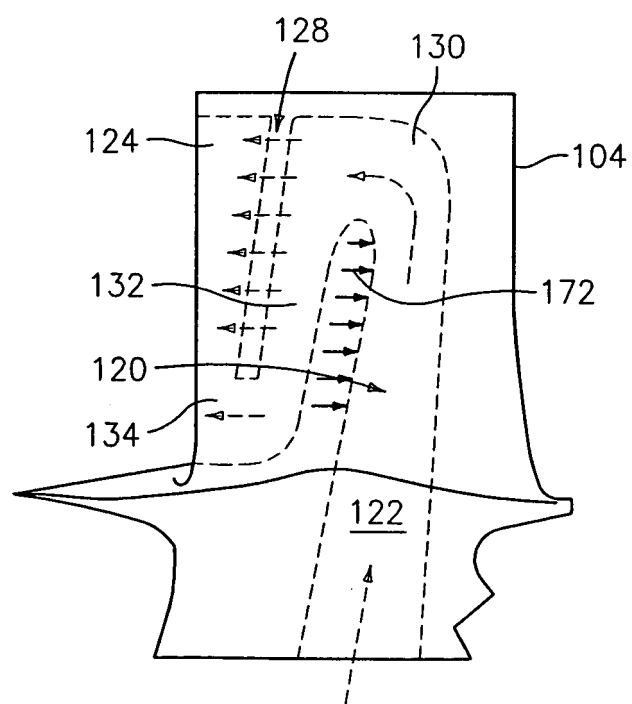


FIG. 6

Description

BACKGROUND

5 (1) Field of the Invention

[0001] The present invention relates to a turbine engine component having an improved scheme for cooling an airfoil portion.

10 (2) Prior Art

[0002] The overall cooling effectiveness is a measure used to determine the cooling characteristics of a particular design. The ideal non-achievable goal is unity, which implies that the metal temperature is the same as the coolant temperature inside an airfoil. The opposite can also occur when the cooling effectiveness is zero implying that the metal temperature is the same as the gas temperature. In that case, the blade material will certainly melt and burn away. In general, existing cooling technology allows the cooling effectiveness to be between 0.5 and 0.6. More advanced technology such as supercooling should be between 0.6 and 0.7. Microcircuit cooling as the most advanced cooling technology in existence today can be made to produce cooling effectiveness higher than 0.7.

[0003] Fig. 1 shows a durability map of cooling effectiveness (x-axis) vs. the film effectiveness (y-axis) for different lines of convective efficiency. Placed in the map is a point 10 related to a new advanced serpentine microcircuit shown in FIGS. 2a - 2c. This serpentine microcircuit includes a pressure side serpentine circuit 20 and a suction side serpentine circuit 22 embedded in the airfoil walls 24 and 26.

[0004] The Table I below provides the operational parameters used to plot the design point in the durability map.

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Operational Parameters for serpentine microcircuit	
beta	2.898
Tg	2581 [F]
Tc	1365 [F]
Tm	2050 [F]
Tm_bulk	1709 [F]
Phi_loc	0.437
Phi_bulk	0.717
Tco	1640 [F]
Tci	1090 [F]
eta_c_loc	0.573
eta_f	0.296
Cooling Flow	3.503%
Total WAE	10.8
Legend for Table I Beta = heat load Phi_loc = local cooling effectiveness Phi_bulk = bulk cooling effectiveness Eta_c_loc = local cooling efficiency Eta_f = film effectiveness Tg = gas temperature Tc = coolant temperature Tm = metal temperature Tm_bulk = bulk metal temperature Tco = exit coolant temperature Tci = inlet coolant temperature WAE = compressor engine flow, pps	

[0005] It should be noted that the overall cooling effectiveness from the table is 0.717 for a film effectiveness of 0.296 and a convective efficiency (or ability to pick-up heat) of 0.573. Also note that the corresponding cooling flow for a turbine blade having this cooling microcircuit is 3.5% engine flow. FIG. 3 illustrates the cooling flow distribution for a turbine blade with the serpentine microcircuits of FIGS. 2a - 2c embedded in the airfoils walls.

[0006] There are however field problems that can be addressed efficiently with peripheral microcircuit designs. One such field problem is illustrated in FIGS. 4A and 4B. In FIG. 4A, the streamlines of the gas path close to the external surface of the airfoil illustrate four different regions in which the gas flow changes direction or migration: a tip region, two midsection regions, and a root region. In between the tip and the upper mid region, the flow transitions through a pseudo stagnation point(s). The momentum of the external gas seems to decelerate in such a way as to impose a local thermal load to the part. This manifests itself by regions where the propensity for erosion and oxidation increase in the airfoil surface. The superposition of FIG. 4B illustrates the local coincidence between the pseudo-stagnation region and the blade distress in the part surface. In the mid region, the upper and lower region also converge onto one another, but even though the space between streamlines decreases, the flow seems to accelerate and there is no pseudo-stagnation regions. A mild manifestation of the same tip-to-mid phenomena seems to initiate in the transition region between the mid-to-root regions. It is therefore necessary to tailor the peripheral microcircuit in such a manner as to address these local high thermal load regions.

SUMMARY OF THE INVENTION

[0007] In accordance with the present invention, a turbine engine component is provided with improved cooling. The turbine engine component broadly comprises an airfoil portion having a pressure side and a suction side. The turbine engine component further has a first cooling circuit within the pressure side for cooling the pressure side of the airfoil portion and a second cooling circuit within the suction side for cooling the suction side of the airfoil portion and for cooperating with means for creating a cooling film over the pressure side.

[0008] Other details of the serpentine microcircuits for hot gas migration of the present invention, as well as other advantages attendant thereto, are set forth in the following detailed description and the accompanying drawings wherein like reference numerals depict like elements.

BRIEF DESCRIPTION OF THE DRAWINGS

[0009]

FIG. 1 is a graph showing cooling effectiveness versus film effectiveness for a turbine engine component;
 FIG. 2A shows an airfoil portion of a turbine engine component having a pressure side cooling microcircuit embedded in the pressure side wall and a suction side cooling microcircuit embedded in the suction side wall;
 FIG. 2B is a schematic representation of a pressure side cooling microcircuit used in the airfoil portion of FIG. 2A;
 FIG. 2C is a schematic representation of a suction side cooling microcircuit used in the airfoil portion of FIG. 2A;
 FIG. 3 illustrates the cooling flow distribution for a turbine engine component with serpentine microcircuits embedded in the airfoil walls;
 FIG. 4A is a schematic representation illustrating the pressure side distress on an airfoil surface;
 FIG. 4B is a schematic representation of the local coincidence between the pseudo-stagnation region and the blade distress;
 FIG. 5 is a schematic representation of a peripheral pressure side cooling circuit;
 FIG. 6 is a schematic representation of a peripheral suction side cooling circuit; and
 FIG. 7 is a schematic representation of main body internal cooling circuits.

DETAILED DESCRIPTION OF THE PREFERRED EMBODIMENT(S)

[0010] Referring now to FIGS. 5 and 6, there are depicted two peripheral cooling arrangements which may be used to address local increases in the airfoil thermal load of a turbine engine component 90 such as a turbine blade. The two peripheral cooling arrangements include a peripheral pressure side microcircuit 100 which may be incorporated or embedded within the wall forming the pressure side of an airfoil portion 104 and a suction side microcircuit 120 which may be incorporated or embedded within the wall forming the suction side of the airfoil portion 104.

[0011] In FIG. 5, the pressure side peripheral microcircuit 100 is shown. In this circuit, the first leg 102 has an inlet 103 which receives cooling fluid from a source (not shown). The leg 102 provides a flow of cooling fluid which quenches the hot spot in the tip-to-mid region of the airfoil portion 104 shown in FIG. 4B. The cooling fluid within the leg 102 proceeds around a 180 degree bend 106 which is supplemented with a plurality of film holes 108, preferably three film holes. The film holes 108 ensure flow acceleration through the bend 106 to a second downstream leg 110 which ends

below the platform 112 of the turbine engine component 90 in an exit 164. Cooling fluid from the leg 110 is fed into an internal trailing edge circuit 114 to be discussed hereinafter via the exit 164 where it is used to further cool the airfoil portion 104.

[0012] Referring now to FIG. 6, there is shown a peripheral suction side microcircuit 120. The circuit 120 has a first leg 122 which communicates with a source (not shown) of cooling fluid. In the first leg 122, the cooling flow convects heat away from the suction side. Since the circuit 120 has no film holes, effective cooling may not be done past the external gage point of the airfoil portion 104 where any film cooling would provide high aerodynamic penalties due to mixing. Thus, the circuit 120 is used to feed cooling fluid to a leading edge microcircuit 124 which wraps around the leading edge 126 of the airfoil portion 104. The circuit 120 feeds or supplies cooling fluid to the leading edge wrap around circuit 124 through a plurality of wall cross over holes 128. As can be seen from FIG. 6, the circuit 120 has a bend 130 and a second leg 132. The holes 128 are preferably located in the vicinity of the bend 130 and the second leg 132. The second leg 132 may also communicate with the wrap around circuit 124 via a passageway 134. As the microcircuit 124 wraps around the leading edge, several holes 136 are located in the leading edge and are used to cool the leading edge of the airfoil portion 104. Further, the microcircuit 124 is provided with a plurality of film holes 138 for creating a film of cooling fluid over the pressure side of the airfoil portion.

[0013] Referring now to FIG. 7, there is shown the main body internal cooling circuits which include a leading edge internal cooling circuit 150 and the trailing edge internal cooling circuit 114. The leading edge internal cooling circuit 150 communicates with a source (not shown) of cooling fluid, such as engine bleed air, via an inlet 151 and has one or more film cooling holes 152 adjacent the tip 154 of the airfoil portion 104 to provide tip cooling. The circuit 150 also has a plurality of cross-over holes 156 for supplying cooling fluid to the leading edge microcircuit 124.

[0014] The trailing edge internal circuit 114 also communicates with a source (not shown) of cooling fluid, such as engine bleed air, via an inlet 157 and has one or more film cooling holes 158 adjacent the tip 154 to provide tip cooling. The circuit 114 also has a plurality of cross-over holes 160 for communicating with a trailing edge cooling circuit 162 for cooling the trailing edge of the airfoil portion 104. As can be seen from FIG. 7, the trailing edge internal circuit 114 also receives cooling fluid from the peripheral pressure side microcircuit 100 via the exit 164.

[0015] Each of the leading edge internal circuit 150 and the trailing edge internal circuit 114 may be provided with a plurality of film cooling holes 170 and 172 respectively to form cooling films over the pressure and suction sides of the airfoil portion 104.

[0016] Using the pressure and suction side cooling circuits of the present invention, the airfoil portion of a turbine engine component may be very effectively convectively cooled. Using the pressure side circuit, the cooling flow is returned to the trailing edge internal circuit for further cooling of the airfoil. Using the suction side circuit, the leading edge of the airfoil is cooled first before discharging in pressure side film. This effective use of coolant allows for positive effects on cycle thermodynamic efficiency, turbine efficiency, rotor inlet temperature impacts, and specific fuel consumption.

Claims

1. A turbine engine component (90) comprising:

an airfoil portion (104) having a pressure side and a suction side;
a first cooling circuit (100) within said pressure side for cooling said pressure side of said airfoil portion (104); and
a second cooling circuit (120) within said suction side for cooling said suction side of said airfoil portion (104) and for cooperating with means for creating a cooling film over said pressure side.

2. The turbine engine component according to claim 1, wherein said means for creating a cooling film over said pressure side comprises a cooling circuit (124) wrapped around a leading edge (126) of said airfoil portion (104).

3. The turbine engine component according to claim 2, further comprising said cooling circuit (124) wrapped around said leading edge (126) having a first set of film holes (138) for cooling said leading edge.

4. The turbine engine component according to claim 3 further comprising said cooling circuit (124) wrapped around said leading edge (126) having a second set of film holes (138) for cooling said pressure side of said airfoil portion (104).

5. The turbine engine component according to any preceding claim, further comprising a leading edge internal circuit (150) and a trailing edge internal circuit (114).

6. The turbine engine component according to claim 5, wherein said first cooling circuit (100) has an exit (164) which

delivers cooling fluid to said trailing edge internal circuit (114).

7. The turbine engine component according to claim 6, wherein said first cooling circuit (100) has a first leg (102), a second leg (110), and a bend (106) between said first leg (102) and said second leg (110).

8. The turbine engine component according to claim 7, wherein said second leg (110) terminates in said exit (164).

9. The turbine engine component according to claim 7 or 8, further comprising means for ensuring flow acceleration through the bend (106).

10. The turbine engine component according to claim 9, wherein said flow acceleration ensuring means comprises a plurality of holes (108).

11. The turbine engine component according to any of claims 5 to 10, wherein each of said internal circuits (150, 114) has a plurality of film holes for creating a flow of cooling fluid over said pressure side and said suction side.

12. The turbine engine component according to any of claims 5 to 11, wherein said leading edge internal circuit (150) has a plurality of cross-over holes (156) for supplying fluid to a leading edge cooling circuit (124).

13. The turbine engine component according to any of claims 5 to 12, wherein said trailing edge internal circuit (114) has a plurality of cross-over holes (160) for supplying fluid to a trailing edge cooling circuit (162).

14. The turbine engine component according to any of claims 5 to 13, wherein each of said leading edge and said trailing edge internal circuits (150, 114) has means for cooling a tip (154) of said airfoil portion (104).

15. The turbine engine component according to any preceding claim, wherein said second cooling circuit (120) has a first leg (122), a second leg (132), and a bend (130) between said first leg (122) and said second leg (132).

16. The turbine engine component according to claim 15, wherein said second cooling circuit (120) has a plurality of cross-over holes (128) for supplying cooling fluid to said means for creating a cooling film over said pressure side.

17. The turbine engine component according to claim 15 or 16, wherein said second leg (132) communicates with said means for creating a cooling film over said pressure side.

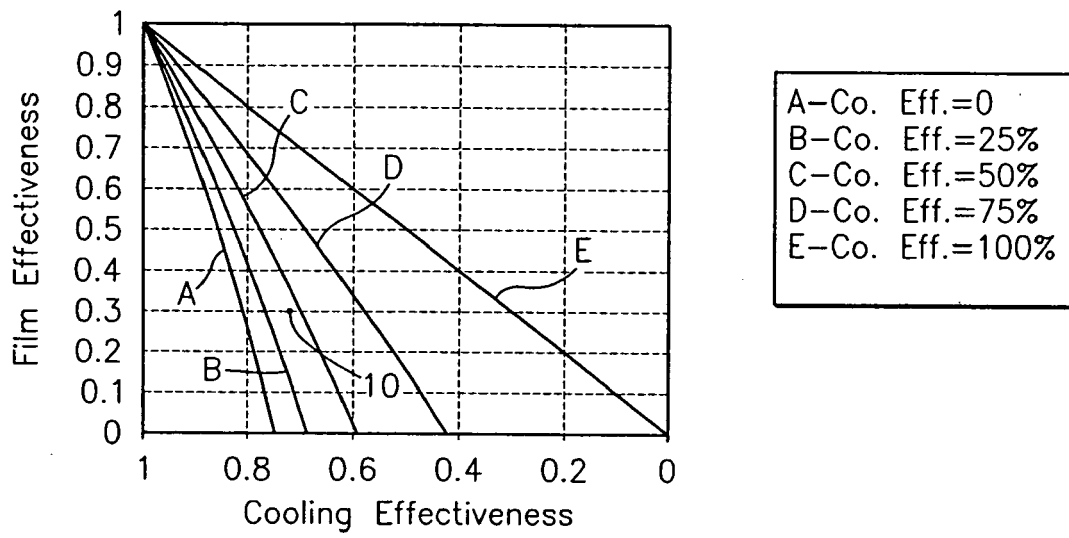


FIG. 1

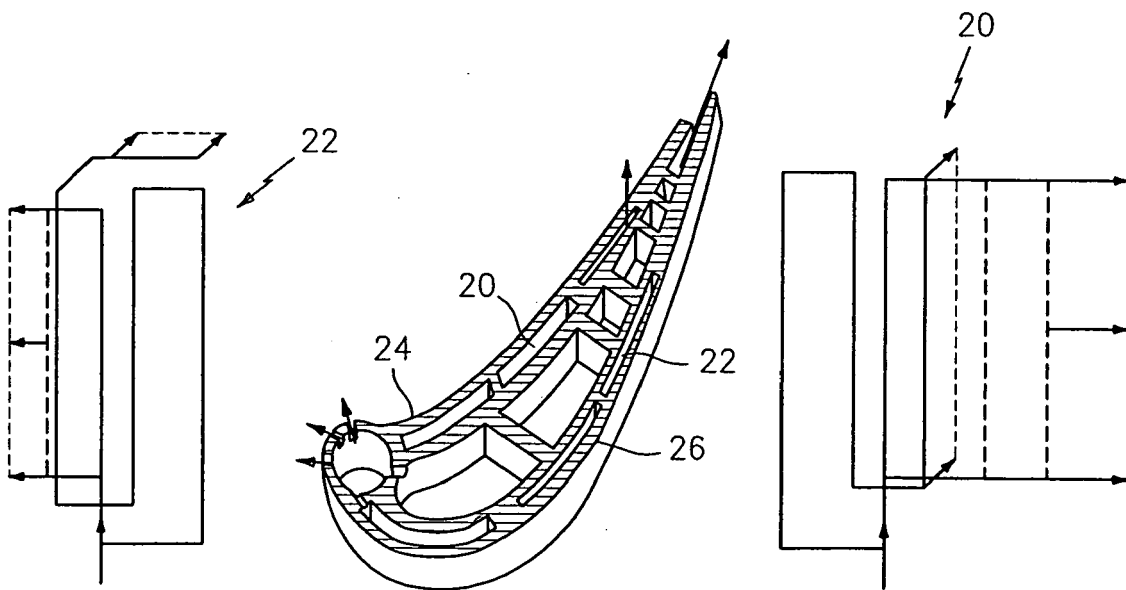


FIG. 2C

FIG. 2A

FIG. 2B

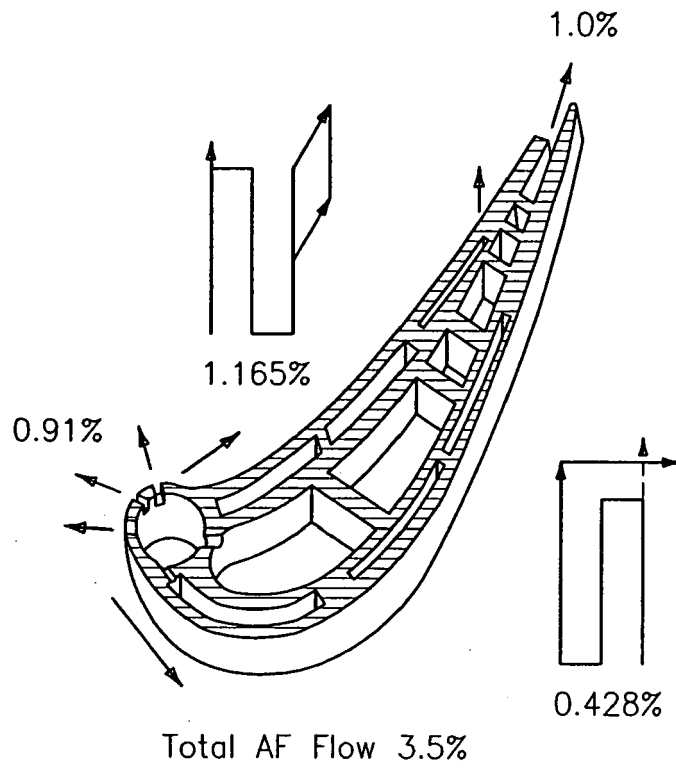


FIG. 3

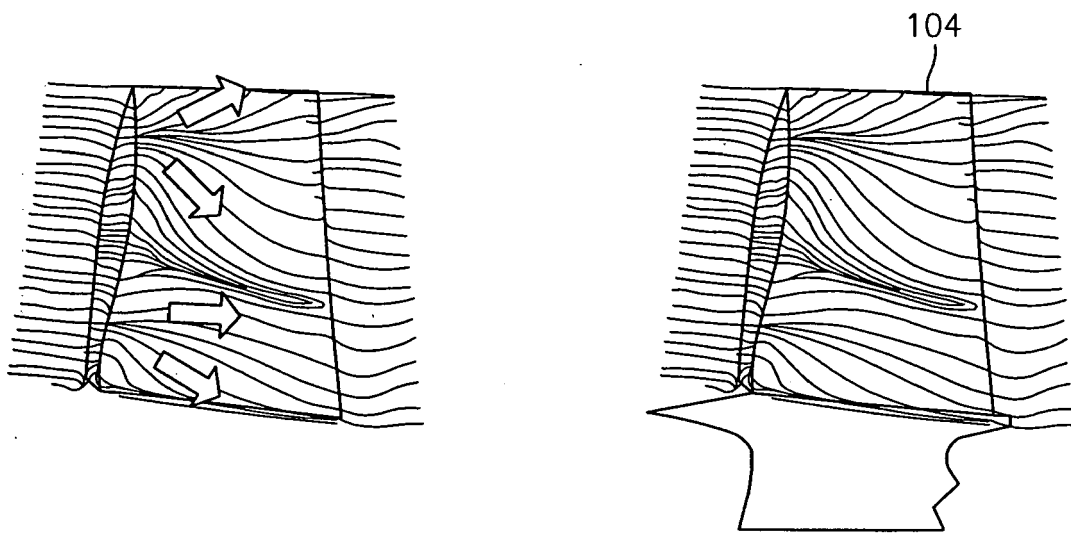


FIG. 4A

FIG. 4B

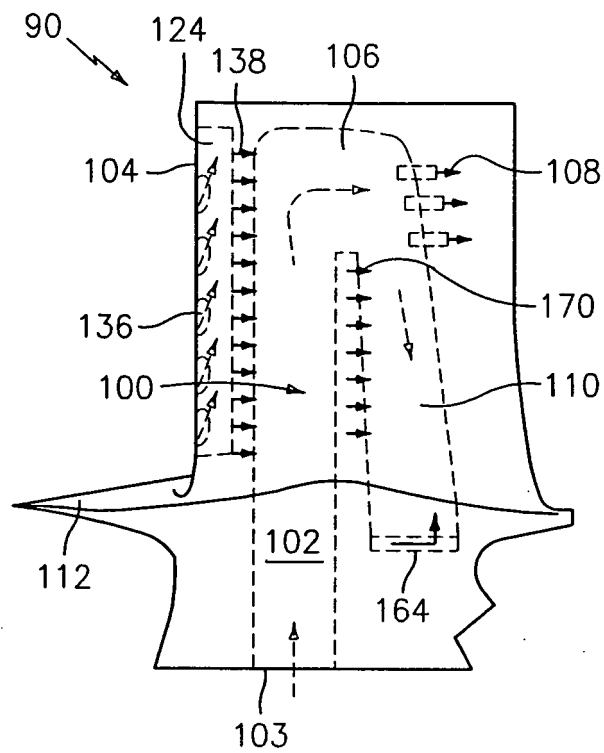


FIG. 5

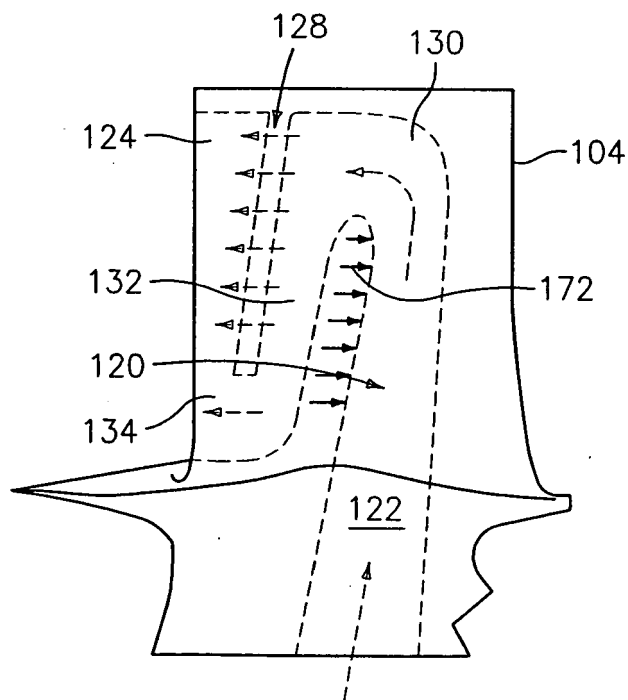


FIG. 6

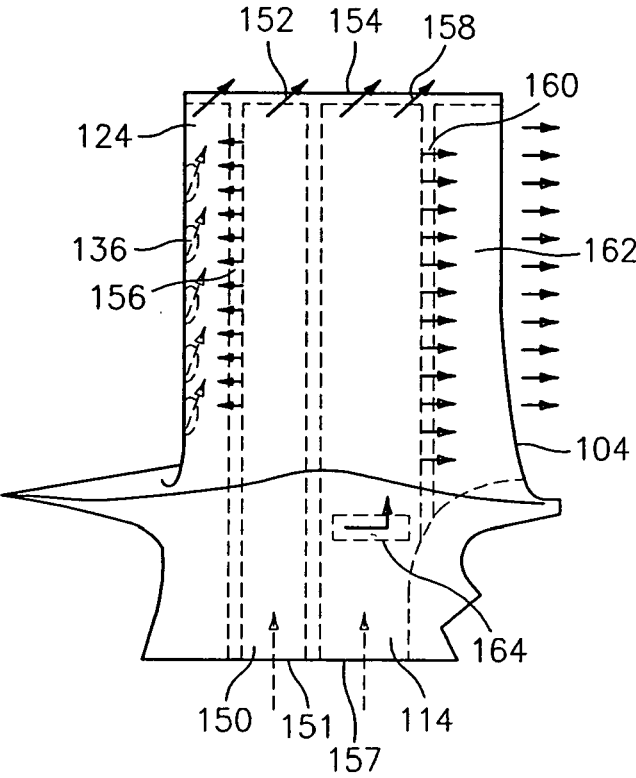


FIG. 7



European Patent
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Application Number
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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			

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**ANNEX TO THE EUROPEAN SEARCH REPORT
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This annex lists the patent family members relating to the patent documents cited in the above-mentioned European search report.
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