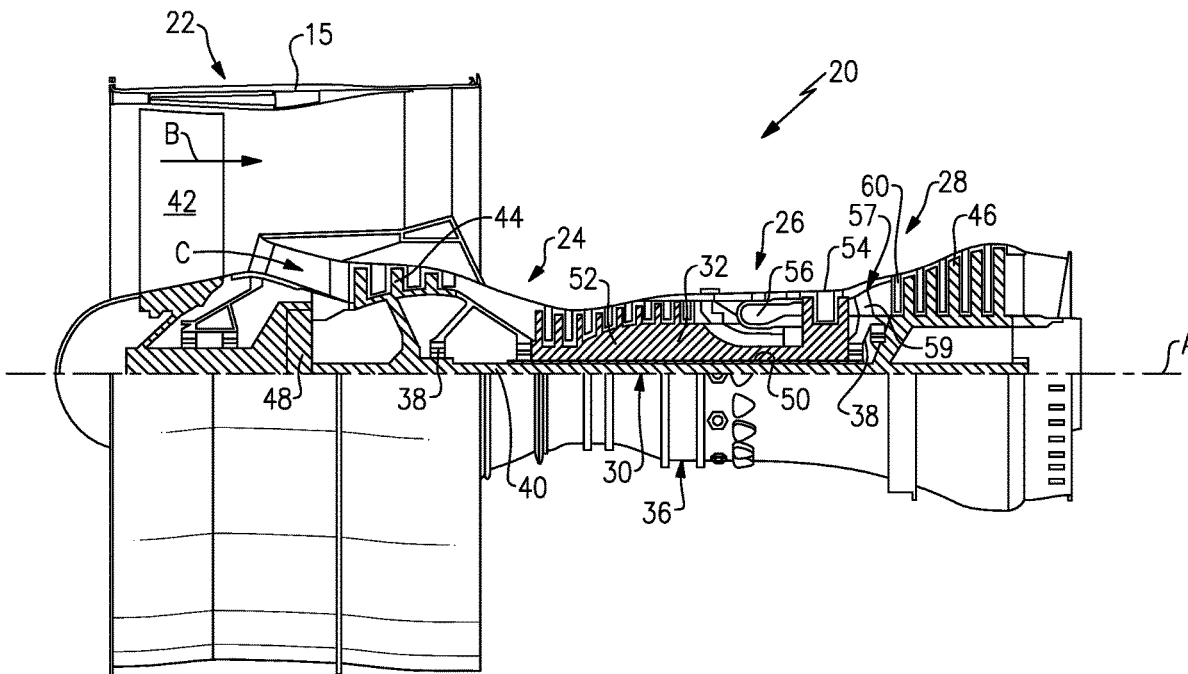


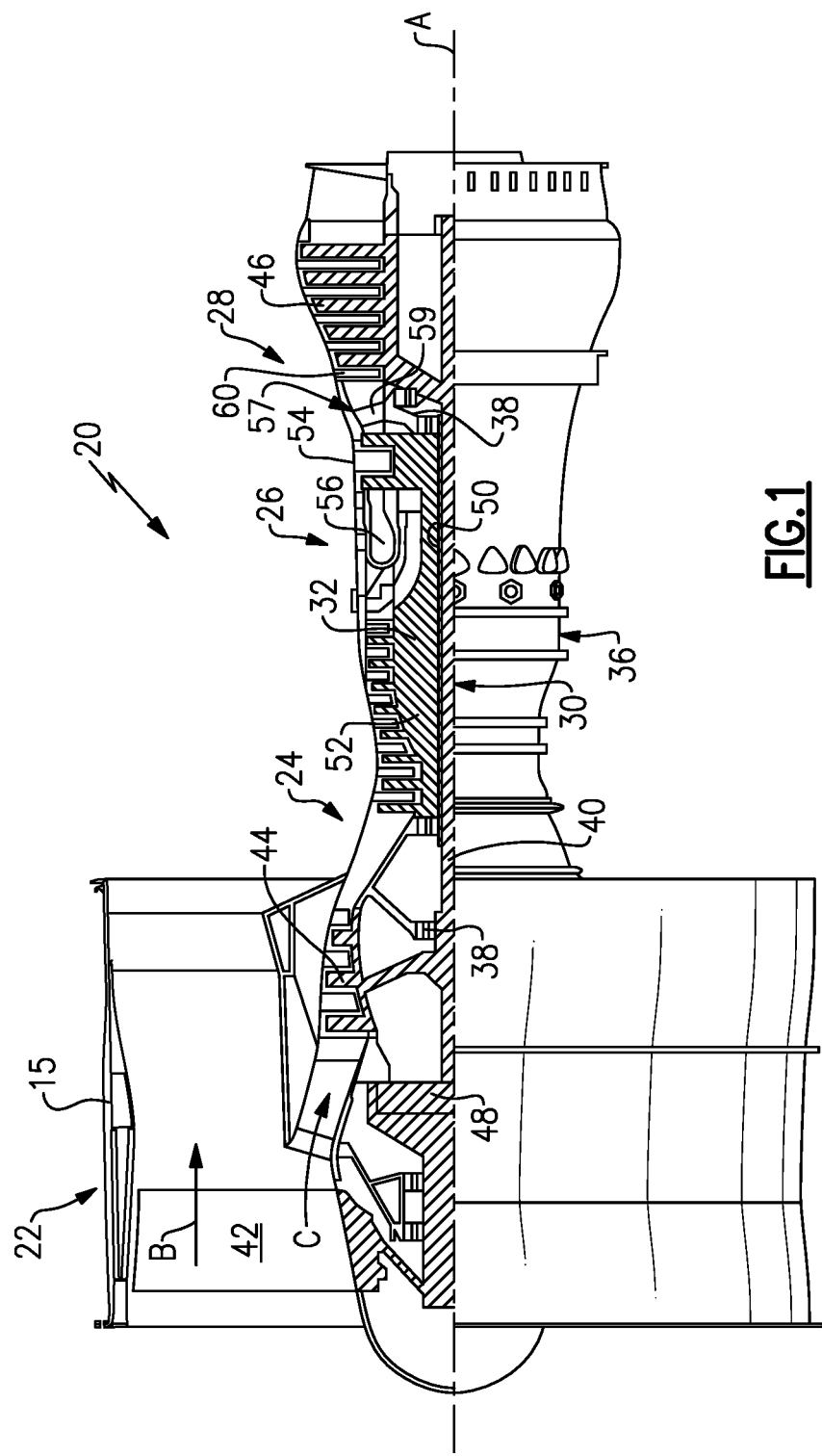


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(19) **United States**(12) **Patent Application Publication****Generale et al.**(10) **Pub. No.: US 2021/0108517 A1**(43) **Pub. Date: Apr. 15, 2021**(54) **CMC AIRFOIL WITH COOLING HOLES**(71) Applicant: **United Technologies Corporation**,
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2300/6033 (2013.01); **F05D 2220/30** (2013.01)(57) **ABSTRACT**

An airfoil includes a continuous airfoil piece that is formed of a laminated ceramic matrix composite. The continuous airfoil piece defines a platform and an airfoil section adjacent the platform. The airfoil section includes at least one internal passage, an airfoil wall that has an interior surface that borders the internal passage, an exterior, combustion gaspath surface, and a plurality of cooling holes. Each cooling hole spans between first and second hole ends. The first hole end opens at the interior surface to the internal passage and the second hole end opens at the exterior surface.





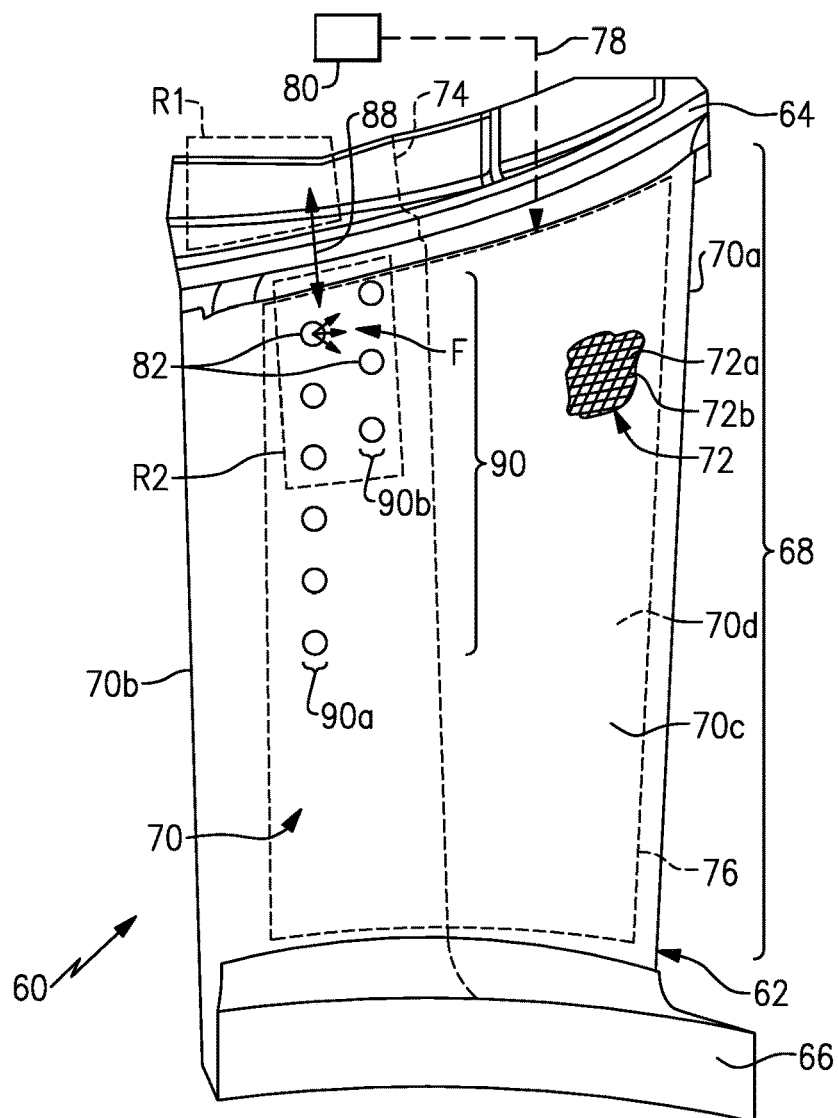


FIG. 2

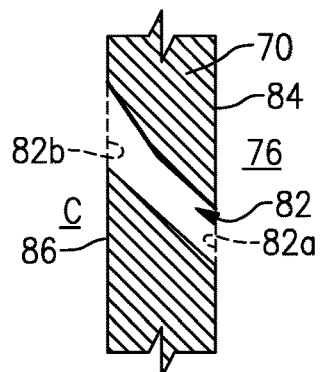


FIG. 3

CMC AIRFOIL WITH COOLING HOLES

BACKGROUND

[0001] A gas turbine engine typically includes a fan section, a compressor section, a combustor section, and a turbine section. Air entering the compressor section is compressed and delivered into the combustion section where it is mixed with fuel and ignited to generate a high-speed exhaust gas flow. The high-speed exhaust gas flow expands through the turbine section to drive the compressor and the fan section.

[0002] The compressor section can include rotors that carry airfoils to compress the air entering the compressor section. A shaft may be coupled to the rotors to rotate the airfoils.

SUMMARY

[0003] An airfoil according to an example of the present disclosure includes a continuous airfoil piece formed of a laminated ceramic matrix composite. The continuous airfoil piece defines a platform and an airfoil section that extends adjacent the platform. The airfoil section has at least one internal passage and an airfoil wall. The airfoil wall has an interior surface that borders the at least one internal passage, an exterior combustion gaspath surface, and a plurality of cooling holes. Each cooling hole spans between first and second hole ends. The first hole end opens at the interior surface to the at least one internal passage and the second hole end opens at the exterior surface.

[0004] In a further embodiment of any of the foregoing embodiments, the cooling holes are in an aft 50% of the airfoil section.

[0005] In a further embodiment of any of the foregoing embodiments, the cooling holes are in a radially outer 50% of the airfoil section.

[0006] In a further embodiment of any of the foregoing embodiments, the cooling holes are in an aft 50% of the airfoil section and a radially outer 50% of the airfoil section.

[0007] In a further embodiment of any of the foregoing embodiments, the cooling holes are arranged as a group in first and second radial rows, and the first and second radial rows are radially staggered.

[0008] In a further embodiment of any of the foregoing embodiments, the laminated ceramic matrix composite includes a silicon carbide matrix and silicon carbide fibers disposed in the silicon carbide matrix.

[0009] In a further embodiment of any of the foregoing embodiments, the cooling holes are at a location on the airfoil section that coincides with a peak temperature location.

[0010] A gas turbine engine according to an example of the present disclosure includes a source of cooling air, a combustor providing combustion gases, and a turbine section receiving the combustion gases. The turbine section has an airfoil that has a continuous airfoil piece formed of a laminated ceramic matrix composite. The continuous airfoil piece defines a platform and an airfoil section adjacent the platform. The airfoil section has at least one internal passage receiving the cooling air and an airfoil wall. The airfoil wall has an interior surface bordering the at least one internal passage, an exterior combustion gaspath surface, and a plurality of cooling holes. Each cooling hole spans between first and second hole ends. The first hole end opens at the

interior surface to the at least one internal passage and the second hole end opens at the exterior surface. The cooling holes discharge the cooling air from the at least one passage. A thermal gradient is established by the cooling air discharged from the cooling holes to be within a temperature difference of no more than 300° C. between a low temperature region of the airfoil piece and a high temperature region of the airfoil section.

[0011] In a further embodiment of any of the foregoing embodiments, the low temperature region is at a non-gaspath region of the platform and the high temperature region is at a portion of the exterior combustion gaspath surface of the airfoil wall.

[0012] In a further embodiment of any of the foregoing embodiments, the cooling holes are in a radially outer 50% of the airfoil section.

[0013] In a further embodiment of any of the foregoing embodiments, the cooling holes are in an aft 50% of the airfoil section.

[0014] In a further embodiment of any of the foregoing embodiments, the laminated ceramic matrix composite includes a silicon carbide matrix and silicon carbide fibers disposed in the silicon carbide matrix.

[0015] In a further embodiment of any of the foregoing embodiments, the cooling holes are arranged as a group in first and second radial rows, and the first and second radial rows are radially staggered.

[0016] A method for cooling an airfoil according to an example of the present disclosure includes providing an airfoil as in any of the foregoing embodiments, establishing a thermal gradient between a low temperature region of the airfoil piece and a high temperature region of the airfoil section to be within a temperature difference of no more than 300° C. by providing cooling air to the internal passage and then through the cooling holes. The cooling air discharges as a cooling film on the airfoil section. The portion has a low temperature region and a high temperature region.

[0017] In a further embodiment of any of the foregoing embodiments, the low temperature region is at a non-gaspath region of the platform and the high temperature region is at a portion of the exterior combustion gaspath surface of the airfoil wall.

[0018] In a further embodiment of any of the foregoing embodiments, the laminated ceramic matrix composite includes a silicon carbide matrix and silicon carbide fibers disposed in the silicon carbide matrix.

[0019] In a further embodiment of any of the foregoing embodiments, the cooling holes are in an aft 50% of the airfoil section.

[0020] In a further embodiment of any of the foregoing embodiments, the cooling holes are in a radially outer 50% of the airfoil section.

[0021] In a further embodiment of any of the foregoing embodiments, the cooling holes are arranged as a group in first and second radial rows, and the first and second radial rows are radially staggered.

BRIEF DESCRIPTION OF THE DRAWINGS

[0022] The various features and advantages of the present disclosure will become apparent to those skilled in the art from the following detailed description. The drawings that accompany the detailed description can be briefly described as follows.

[0023] FIG. 1 illustrates an example gas turbine engine.

[0024] FIG. 2 illustrates an airfoil of the engine.

[0025] FIG. 3 illustrates a sections view through a portion of the airfoil.

DETAILED DESCRIPTION

[0026] FIG. 1 schematically illustrates a gas turbine engine 20. The gas turbine engine 20 is disclosed herein as a two-spool turbofan that generally incorporates a fan section 22, a compressor section 24, a combustor section 26 and a turbine section 28. The fan section 22 drives air along a bypass flow path B in a bypass duct defined within a nacelle 15, and also drives air along a core flow path C for compression and communication into the combustor section 26 then expansion through the turbine section 28. Although depicted as a two-spool turbofan gas turbine engine in the disclosed non-limiting embodiment, it should be understood that the concepts described herein are not limited to use with two-spool turbofans as the teachings may be applied to other types of turbine engines including three-spool architectures.

[0027] The exemplary engine 20 generally includes a low speed spool 30 and a high speed spool 32 mounted for rotation about an engine central longitudinal axis A relative to an engine static structure 36 via several bearing systems 38. Terms such as “axial,” “radial,” “circumferential,” and variations of these terms are made with reference to the engine central axis A. It should be understood that various bearing systems 38 at various locations may alternatively or additionally be provided, and the location of bearing systems 38 may be varied as appropriate to the application.

[0028] The low speed spool 30 generally includes an inner shaft 40 that interconnects, a first (or low) pressure compressor 44 and a first (or low) pressure turbine 46. The inner shaft 40 is connected to the fan 42 through a speed change mechanism, which in exemplary gas turbine engine 20 is illustrated as a geared architecture 48 to drive a fan 42 at a lower speed than the low speed spool 30. The high speed spool 32 includes an outer shaft 50 that interconnects a second (or high) pressure compressor 52 and a second (or high) pressure turbine 54. A combustor 56 is arranged in exemplary gas turbine 20 between the high pressure compressor 52 and the high pressure turbine 54. A mid-turbine frame 57 of the engine static structure 36 may be arranged generally between the high pressure turbine 54 and the low pressure turbine 46. The mid-turbine frame 57 further supports bearing systems 38 in the turbine section 28. The inner shaft 40 and the outer shaft 50 are concentric and rotate via bearing systems 38 about the engine central longitudinal axis A which is collinear with their longitudinal axes.

[0029] The core airflow is compressed by the low pressure compressor 44 then the high pressure compressor 52, mixed and burned with fuel in the combustor 56, then expanded over the high pressure turbine 54 and low pressure turbine 46. The mid-turbine frame 57 includes airfoils 59 which are in the core airflow path C. The turbines 46, 54 rotationally drive the respective low speed spool 30 and high speed spool 32 in response to the expansion. It will be appreciated that each of the positions of the fan section 22, compressor section 24, combustor section 26, turbine section 28, and fan drive gear system 48 may be varied. For example, gear system 48 may be located aft of the low pressure compressor, or aft of the combustor section 26 or even aft of turbine section 28, and fan 42 may be positioned forward or aft of the location of gear system 48.

[0030] The engine 20 in one example is a high-bypass geared aircraft engine. In a further example, the engine 20 bypass ratio is greater than about six (6), with an example embodiment being greater than about ten (10), the geared architecture 48 is an epicyclic gear train, such as a planetary

gear system or other gear system, with a gear reduction ratio of greater than about 2.3 and the low pressure turbine 46 has a pressure ratio that is greater than about five. In one disclosed embodiment, the engine 20 bypass ratio is greater than about ten (10:1), the fan diameter is significantly larger than that of the low pressure compressor 44, and the low pressure turbine 46 has a pressure ratio that is greater than about five 5:1. Low pressure turbine 46 pressure ratio is pressure measured prior to inlet of low pressure turbine 46 as related to the pressure at the outlet of the low pressure turbine 46 prior to an exhaust nozzle. The geared architecture 48 may be an epicycle gear train, such as a planetary gear system or other gear system, with a gear reduction ratio of greater than about 2.3:1 and less than about 5:1. It should be understood, however, that the above parameters are only exemplary of one embodiment of a geared architecture engine and that the present invention is applicable to other gas turbine engines including direct drive turbofans.

[0031] A significant amount of thrust is provided by the bypass flow B due to the high bypass ratio. The fan section 22 of the engine 20 is designed for a particular flight condition—typically cruise at about 0.8 Mach and about 35,000 feet (10,668 meters). The flight condition of 0.8 Mach and 35,000 ft (10,668 meters), with the engine at its best fuel consumption—also known as “bucket cruise Thrust Specific Fuel Consumption (TSFC)” —is the industry standard parameter of lbf of fuel being burned divided by lbf of thrust the engine produces at that minimum point. “Low fan pressure ratio” is the pressure ratio across the fan blade alone, without a Fan Exit Guide Vane (“FEGV”) system. The low fan pressure ratio as disclosed herein according to one non-limiting embodiment is less than about 1.45. “Low corrected fan tip speed” is the actual fan tip speed in ft/sec divided by an industry standard temperature correction of $[(T_{\text{ram}}/R)/(518.7^{\circ}\text{R})]^{0.5}$. The “Low corrected fan tip speed” as disclosed herein according to one non-limiting embodiment is less than about 1150 ft/second (350.5 meters/second).

[0032] FIG. 2 illustrates a representative airfoil 60 from the turbine section 28 of the engine 20. In this example, the airfoil 60 is a vane arc segment, although it is to be understood that the examples herein may also be applied to blades. A plurality of vane arc segments are situated in a circumferential row about the engine central axis A. The airfoil 60 is comprised of a continuous airfoil piece 62. The continuous airfoil piece 62 includes several sections, including first (outer) and second (inner) platforms 64/66 and an airfoil section 68 that extends between the first and second platforms 64/66. If the airfoil 60 were a blade, there would only be one platform. The airfoil section 68 is comprised of an airfoil wall 70 that defines a leading end 70a, a trailing end 70b, and pressure and suction sides 70c/70d.

[0033] The continuous airfoil piece 62 is formed of a laminated ceramic matrix composite 72, shown in a cutaway portion in FIG. 2. For example, the laminated ceramic matrix composite 72 includes a ceramic matrix 72a and ceramic fibers 72b disposed in the ceramic matrix 72a. In one example, the ceramic matrix is silicon carbide (SiC) and the ceramic fibers are silicon carbide (SiC).

[0034] The laminated ceramic matrix composite 72 is comprised of fiber plies, one of which is represented schematically at 74, that are arranged in a stacked configuration and formed to the desired geometry of the airfoil piece 62. For instance, the fiber plies 74 may be layers or tapes that are

laid-up one on top of the other to form the stacked configuration. The fiber plies **74** may be woven or unidirectional, for example. At least a portion of the fiber plies **74** are continuous through the first platform **64**, the airfoil section **68**, and the second platform **66**. In this regard, the word “continuous” in the phrase “continuous airfoil piece” refers to the continuous airfoil piece **62** having fiber plies **74** that are uninterrupted through the first platform **64**, the airfoil section **68**, and the second platform **66**.

[0035] The airfoil section **68** includes at least one internal passage **76**. The configuration of the internal passage **76** is not particularly limited and may be a single central cavity, a sub-cavity, a serpentine passage, or the like. The internal passage **76** is connected by a cooling air line **78** to a source **80** of cooling air. For example, the source **80** of cooling air may be, but is not limited to, the compressor section **24**, in which case the cooling air line **78** is a compressor bleed line.

[0036] The airfoil wall **70** includes a plurality of cooling holes **82**. As shown in a representative view in FIG. **3**, the airfoil wall **70** has an interior surface **84** that borders the internal passage **76**, and an exterior surface **86**. The exterior surface **86** is a combustion gaspath surface in the core flow path **C**. The cooling holes **82** each span between first and second hole ends **82a/82b**. The first hole end **82a** opens at the interior surface **84** to the internal passage **76** and the second hole end **82b** opens at the exterior surface **86** to the core gas path **C**. For example, the cooling holes **82** may be diffusion cooling holes that are flared, or additional advanced shapes.

[0037] In general, vanes formed of superalloys employ a thermal management strategy that involves cooling the vane as much as possible, to avoid exceeding the temperature limit of the superalloy and to limit effects of creep and fatigue, without much regard to thermally induced internal stresses as the superalloy strength/toughness enables this to be a secondary concern. This paradigm dictates use of many cooling holes arranged at locations that are designed to maximize cooling effectiveness. Ceramic materials, however, have higher maximum use temperatures in comparison to metallic superalloys. Therefore, vanes formed of ceramic materials have no need to employ the same thermal management strategy that is used for superalloy vanes.

[0038] Additionally, ceramic materials have significantly lower thermal conductivity than superalloys and do not possess the same strength and ductility characteristics, making them more susceptible to distress from thermal gradients and the thermally induced stresses those cause. For instance, although a portion of a ceramic vane may be exposed to the high temperatures in the core gas path, another portion of the ceramic vane that is not in the core gas path may be at a lower temperature due to the low thermal conductivity of the ceramic. This, in turn, may generate high thermal gradients that cause thermally induced stresses. While the high strength and toughness of superalloys permits resistance to thermal stresses, laminated ceramic matrix composites by comparison are more prone to distress from thermal stress. Thermal stresses may cause distress at relatively weak locations, such as interlaminar interfaces between fiber plies where there are no fibers carrying load. Additionally, some laminated ceramic matrix composites, such as SiC/SiC composites in which the matrix is formed by chemical vapor infiltration, may have residual porosity that may be subject to distress from thermal stresses. Therefore, although maximum cooling may be desirable for superalloy vanes, maxi-

mized cooling of a ceramic vane, particularly laminated ceramic matrix composites and composites with residual porosity, may exacerbate thermal gradients and thus be counter-productive to meeting durability goals. In this regard, the cooling holes **82** of the airfoil **60** are configured to establish a desired thermal gradient rather than maximize cooling.

[0039] For example, during operation of the engine **20** the airfoil piece **62** has low and high temperature regions. Most typically, the high or highest temperature regions are those regions that are exposed in the core gaspath **C**, and the low or lowest temperature regions are those that are outside of the core flow path **C**. For instance, as shown in FIG. **2**, the airfoil piece **62** has a low temperature region **R1** and a high temperature region **R2**. In this example, the low temperature region **R1** is at a non-gaspath region of the first platform **64**. The low temperature region **R1** may include, for example, the radially outer portion of the first platform and features that are on the radially outer side of the first platform, such as flanges, rails, hooks, or the like.

[0040] The high temperature region **R2** is at a portion of the exterior combustion gaspath surface **86** of the airfoil wall **70**. As an example, the high temperature region **R2** may be exposed to temperatures of approximately 1400 F to 2800 F in the core flow path **C**. The temperature at the low temperature region **R1** by comparison may be approximately up to 1000 F colder. In one example, the high temperature region **R2** coincides with a peak temperature location across the airfoil section **68**. A peak temperature location is a location of the airfoil section **68** that is exposed to higher temperatures than surrounding areas of the airfoil section **68**. A peak temperature location may be determined, for example, by experiment or by computer simulation to generate a temperature profile which can be used to identify peak temperature locations. In some examples, the temperature profile may correspond to cruise conditions or maximum thrust conditions.

[0041] As discussed above, such thermal gradients in laminated ceramic matrix composites may cause thermal stress and distress at interlaminar interfaces. To reduce the thermal gradient, and thus facilitate a reduction in thermal stresses in the laminated ceramic matrix composite **72**, cooling air is provided from the source **80**, through the air line **78**, and into the internal passage **76**. The internal passage **76** feeds the cooling air to the cooling holes **82**. The cooling holes **82** discharge the cooling air at the exterior surface **86**, where the cooling air serves to form a cooling air film, as generally represented at **F** (FIG. **2**), across at least a portion of the exterior surface **86**.

[0042] The cooling air film **F** cools the airfoil wall **70** in the immediate vicinity of the cooling holes **82**, as well areas that are closely adjacent the cooling holes **82**. For instance, in the illustrated example, the cooling holes **82** are in close proximity to the first platform and may thus serve to reduce the temperature in the high temperature region **R1** as well as the adjacent portion of the first platform **64** and fillet area between the first platform **64** and the airfoil section **68**. The region **R2**, fillet area, and first platform **64**, being formed of the laminated ceramic matrix composite **72**, do not require cooling to lower the temperature below a maximum use temperature. Rather, the cooling holes **82** are designed to establish a thermal gradient, as represented generally at **88**, between the regions **R1/R2** to be within a temperature difference of no more than 300° C. and in further examples

no more than 250° C., 200° C., 150° C., or 100° C. In the illustrated example, this means that the difference in temperature across the portion of the airfoil piece **62** from region **R2** to region **R1** (e.g., inclusive of the fillet area in this example) is no more than 150° C.

[0043] In the illustrated example, the cooling holes **82** are located in the aft 50% of the length of the airfoil section **68** and in the radially outer 50% of the height (span) of the airfoil section (from the second platform **66** to the first platform **64**). Collectively, the aft 50% and the radially outer 50% may be referred to as the 50%/50% zone. In particular, the 50%/50% zone of the airfoil section **68** may be exposed to the highest temperatures in the temperature profile and may thus be the location where there is likely to be the greatest thermal gradient relative to the cooler, adjacent first platform **64**. By utilizing the cooling holes in the 50%/50% zone the thermal gradient between regions **R1** and **R2** may be limited to a temperature difference of no more than 150° C.

[0044] The cooling holes **82** may also be arranged as a group, as represented generally at **90**. In this example, the group **90** includes a first radial row **90a** of the cooling holes **82** and a second radial row **90b** of the cooling holes **82** that may be radially staggered from the first radial row **90a**. A “row” includes cooling holes **82** that lie on a common straight line, in this case in the radial direction. As an example, the breakout points of the cooling holes **82** at the surface of the airfoil section **68** are all on a common straight line for each respective row **90a/90b**. Alternatively, at least a portion of the cooling holes **82** lie on a common straight line for each respective row **90a/90b**. The staggering of the rows **90a/90b** may facilitate augmenting the film effectiveness due to the super-positioning of ejected film with considerations for the minimum permissible distance between cooling holes. Additionally, the first row **90a** may be radially longer and contain a higher number of cooling holes **82** than the second row **90b**. Such a configuration may be used to provide a higher number of the cooling holes **82** at a location that coincides with the highest temperatures in the region **R2**, while other locations where the temperature is not quite as high may have fewer of the cooling holes **82**.

[0045] As will be appreciated from this disclosure, the size, number, and location of the cooling holes **82** can be configured in cooperation with other factors, such as the temperature and flow rate of the cooling air and the temperatures of the regions **R1/R2**, to attain the temperature difference no more than 150° C. It is to be further appreciated that the cooling holes **82** and examples herein may also be applied between other or additional low and high temperature regions of the airfoil **60** that exceed target thermal gradients, such as but not limited to, between radially adjacent low and high temperature regions on the airfoil section **68**, between axially adjacent low and high temperature regions on the airfoil section **68**, or between low and high temperature regions on the airfoil section **68** and second platform **66**.

[0046] Although a combination of features is shown in the illustrated examples, not all of them need to be combined to realize the benefits of various embodiments of this disclosure. In other words, a system designed according to an embodiment of this disclosure will not necessarily include all of the features shown in any one of the Figures or all of the portions schematically shown in the Figures. Moreover,

selected features of one example embodiment may be combined with selected features of other example embodiments. [0047] The preceding description is exemplary rather than limiting in nature. Variations and modifications to the disclosed examples may become apparent to those skilled in the art that do not necessarily depart from this disclosure. The scope of legal protection given to this disclosure can only be determined by studying the following claims.

What is claimed is:

1. An airfoil comprising:

a continuous airfoil piece formed of a laminated ceramic matrix composite, the continuous airfoil piece defining a platform and an airfoil section extending adjacent the platform, the airfoil section including,

at least one internal passage, and

an airfoil wall having an interior surface bordering the at least one internal passage, an exterior, combustion gaspath surface, and a plurality of cooling holes, each said cooling hole spanning between first and second hole ends, the first hole end opening at the interior surface to the at least one internal passage and the second hole end opening at the exterior surface.

2. The vane arc segment as recited in claim 1, wherein the cooling holes are in an aft 50% of the airfoil section.

3. The vane arc segment as recited in claim 1, wherein the cooling holes are in a radially outer 50% of the airfoil section.

4. The vane arc segment as recited in claim 1, wherein the cooling holes are in an aft 50% of the airfoil section and a radially outer 50% of the airfoil section.

5. The vane arc segment as recited in claim 1, wherein the cooling holes are arranged as a group in first and second radial rows, and the first and second radial rows are radially staggered.

6. The vane arc segment as recited in claim 1, wherein the laminated ceramic matrix composite includes a silicon carbide matrix and silicon carbide fibers disposed in the silicon carbide matrix.

7. The vane arc segment as recited in claim 1, wherein the cooling holes are at a location on the airfoil section that coincides with a peak temperature location.

8. A gas turbine engine comprising:

a source of cooling air;

a combustor providing combustion gases;

a turbine section receiving the combustion gases, the turbine section having an airfoil including a continuous airfoil piece formed of a laminated ceramic matrix composite, the continuous airfoil piece defining a platform and an airfoil section adjacent the platform, the airfoil section including,

at least one internal passage receiving the cooling air, and

an airfoil wall having an interior surface bordering the at least one internal passage, an exterior, combustion gaspath surface, and a plurality of cooling holes, each said cooling hole spanning between first and second hole ends, the first hole end opening at the interior surface to the at least one internal passage and the second hole end opening at the exterior surface, the cooling holes discharging the cooling air from the at least one passage; and

a thermal gradient established by the cooling air discharged from the cooling holes to be within a temperature difference of no more than 300° C. between a low

temperature region of the airfoil piece and a high temperature region of the airfoil section.

9. The gas turbine engine as recited in claim 8, wherein the low temperature region is at a non-gaspath region of the platform and the high temperature region is at a portion of the exterior combustion gaspath surface of the airfoil wall.

10. The gas turbine engine as recited in claim 9, wherein the cooling holes are in a radially outer 50% of the airfoil section.

11. The gas turbine engine as recited in claim 10, wherein the cooling holes are in an aft 50% of the airfoil section.

12. The gas turbine engine as recited in claim 11, wherein the laminated ceramic matrix composite includes a silicon carbide matrix and silicon carbide fibers disposed in the silicon carbide matrix.

13. The gas turbine engine as recited in claim 12, wherein the cooling holes are arranged as a group in first and second radial rows, and the first and second radial rows are radially staggered.

14. A method for cooling an airfoil, the method comprising:

providing an airfoil that includes a continuous airfoil piece formed of a laminated ceramic matrix composite, wherein the continuous airfoil piece defines a platform and an airfoil section adjacent the platform, the airfoil section includes at least one internal passage and an airfoil wall that has an interior surface borders the at least one internal passage, an exterior, combustion gaspath surface, and a plurality of cooling holes, each said cooling hole spans between first and second hole

ends, the first hole end opens at the interior surface to the at least one internal passage and the second hole end opens at the exterior surface; and

establishing a thermal gradient between a low temperature region of the airfoil piece and a high temperature region of the airfoil section to be within a temperature difference of no more than 300° C. by providing cooling air to the internal passage and then through the cooling holes, the cooling air discharging as a cooling film on the airfoil section, wherein the portion includes a low temperature region and a high temperature region.

15. The method as recited in claim 14, wherein the low temperature region is at a non-gaspath region of the platform and the high temperature region is at a portion of the exterior combustion gaspath surface of the airfoil wall.

16. The method as recited in claim 15, wherein the laminated ceramic matrix composite includes a silicon carbide matrix and silicon carbide fibers disposed in the silicon carbide matrix.

17. The method as recited in claim 16, wherein the cooling holes are in an aft 50% of the airfoil section.

18. The method as recited in claim 17, wherein the cooling holes are in a radially outer 50% of the airfoil section.

19. The method as recited in claim 18, wherein the cooling holes are arranged as a group in first and second radial rows, and the first and second radial rows are radially staggered.

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