US009869183B2

(12) **United States Patent**
**Suciu et al.**

(10) **Patent No.: US 9,869,183 B2**
(45) **Date of Patent: Jan. 16, 2018**

(54) **THERMAL BARRIER COATING INSIDE COOLING CHANNELS**

(71) Applicant: **United Technologies Corporation**, Hartford, CT (US)

(72) Inventors: **Gabriel L. Suciu**, Glastonbury, CT (US); **Brian D. Merry**, Andover, CT (US); **James D. Hill**, Tolland, CT (US); **Wesley K. Lord**, South Glastonbury, CT (US)

(73) Assignee: **UNITED TECHNOLOGIES CORPORATION**, Farmington, CT (US)

( * ) Notice: Subject to any disclaimer, the term of this patent is extended or adjusted under 35 U.S.C. 154(b) by 330 days.

(21) Appl. No.: **14/799,994**

(22) Filed: **Jul. 15, 2015**

(65) **Prior Publication Data**

US 2016/0032731 A1 Feb. 4, 2016

**Related U.S. Application Data**

(60) Provisional application No. 62/032,039, filed on Aug. 1, 2014.

(51) **Int. Cl.**

***F01D 5/08*** (2006.01)
***F01D 5/28*** (2006.01)

(52) **U.S. Cl.**

CPC ............. ***F01D 5/082*** (2013.01); ***F01D 5/084*** (2013.01); ***F01D 5/085*** (2013.01); ***F01D 5/288*** (2013.01); *Y02T 50/672* (2013.01); *Y02T 50/676* (2013.01)

(58) **Field of Classification Search**

CPC ... F01D 5/022; F01D 5/06; F01D 5/08; F01D 5/081; F01D 5/082; F01D 5/084; F01D 5/085; F01D 25/12; F05D 2300/20; F05D 2300/30; F05D 2230/30; F05D 2230/90; F05D 2240/20; F05D 2240/24

See application file for complete search history.

(56) **References Cited**

U.S. PATENT DOCUMENTS

| | | | | |
|---|---|---|---|---|
| 5,735,671 A * | 4/1998 | Brauer | F01D 5/081 | 415/177 |
| 6,408,610 B1 * | 6/2002 | Caldwell | C23C 14/046 | 29/889.2 |

(Continued)

FOREIGN PATENT DOCUMENTS

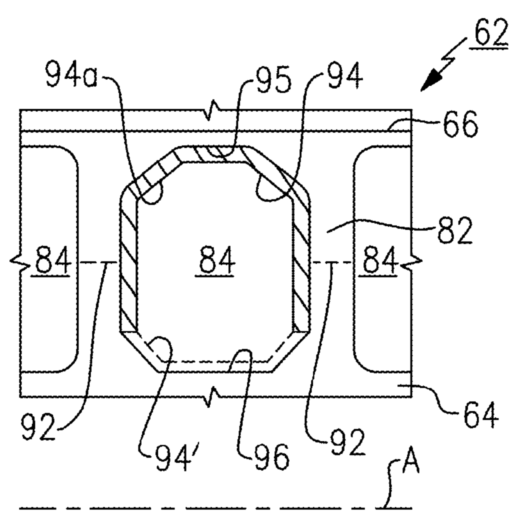
| | | |
|---|---|---|
| EP | 1802784 | 2/2012 |
| WO | 2014052288 | 4/2014 |

*Primary Examiner* — Nathaniel Wiehe
*Assistant Examiner* — Elton Wong
(74) *Attorney, Agent, or Firm* — Carlson, Gaskey & Olds, P.C.

(57) **ABSTRACT**

A rotor for a gas turbine engine according to an exemplary aspect of the present disclosure includes, among other things, a rotor disk rotatable about an axis and a gas path wall coupled to and radially outward of the rotor disk. The gas path wall bounds a radially inward portion of a gas path. A plurality of rotor spokes are radially intermediate the rotor disk and the gas path wall. The plurality of rotor spokes is circumferentially spaced to define a plurality of cooling channels intermediate the rotor disk and the gas path wall. A thermal barrier coating is disposed on a surface of at least one of the plurality of cooling channels. A method of cooling a rotor assembly is also disclosed.

**20 Claims, 5 Drawing Sheets**



(56) **References Cited**

U.S. PATENT DOCUMENTS

| | | | | |
|---|---|---|---|---|
| 8,784,061 | B2 * | 7/2014 | Are | F01D 5/066 415/175 |
| 8,845,288 | B2 * | 9/2014 | Tholath | F01D 5/081 416/219 R |
| 2006/0078417 | A1 | 4/2006 | Benton | |
| 2006/0110254 | A1 | 5/2006 | Itzel et al. | |
| 2007/0116574 | A1 * | 5/2007 | Itzel | F01D 5/082 416/193 A |
| 2007/0248457 | A1 | 10/2007 | Dodd et al. | |
| 2009/0075023 | A1 | 3/2009 | Cosack et al. | |
| 2011/0280716 | A1 | 11/2011 | Konitzer et al. | |
| 2013/0101436 | A1 * | 4/2013 | Colson | F04D 29/329 416/97 R |
| 2013/0108445 | A1 * | 5/2013 | Suciu | F01D 5/026 416/95 |
| 2013/0108468 | A1 | 5/2013 | Suciu et al. | |
| 2015/0322795 | A1 * | 11/2015 | Thomas | F01D 5/188 427/8 |

* cited by examiner

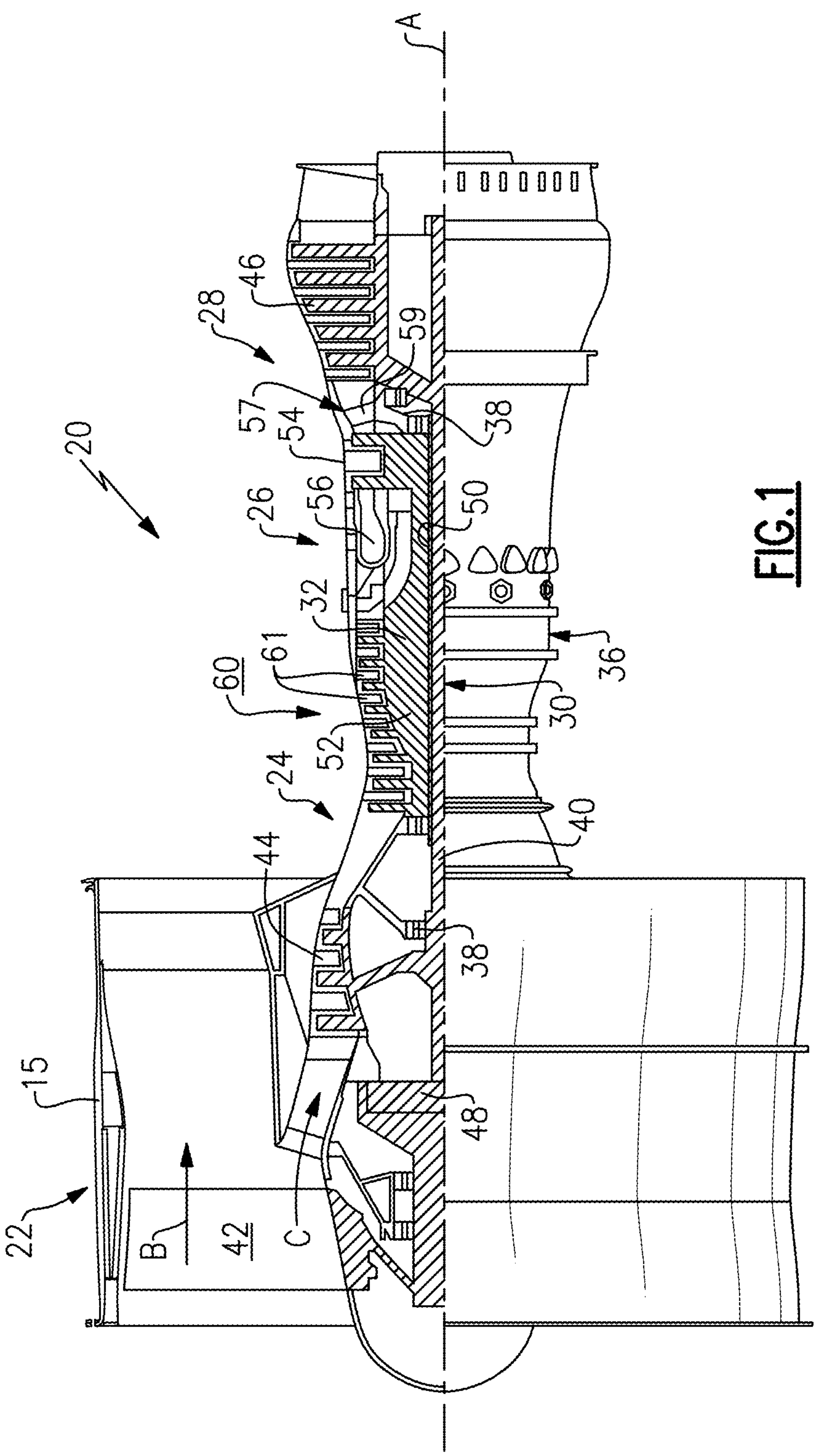

A
20
28
46
59
57
54
38
26
56
50
32
60
61
36
30
52
24
40
44
38
15
22
B
42
C
48

FIG.1

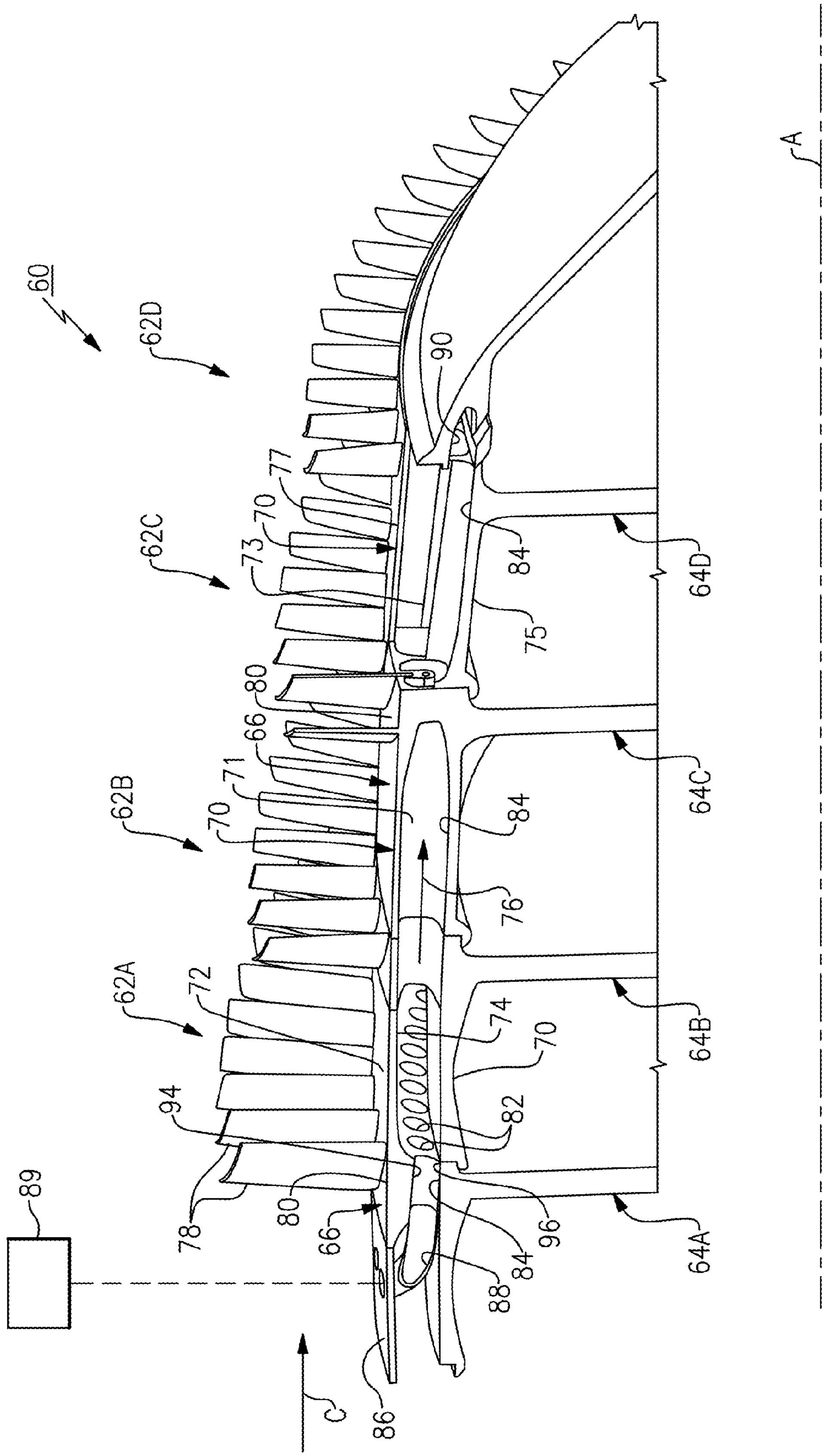

60
62D
62C
62B
62A
A
90
77
77
70
73
84
64D
75
80
80
66
71
70
84
64C
76
72
74
70
64B
94
82
82
89
78
80
66
84
96
88
84
64A
C
86

FIG.2

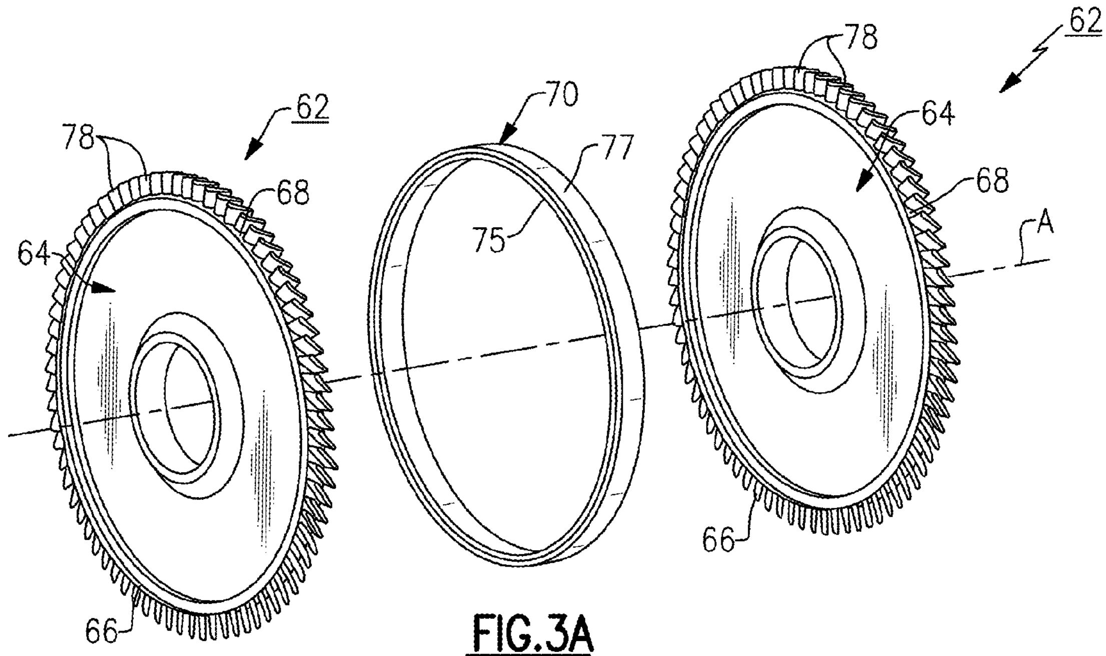



FIG.3A

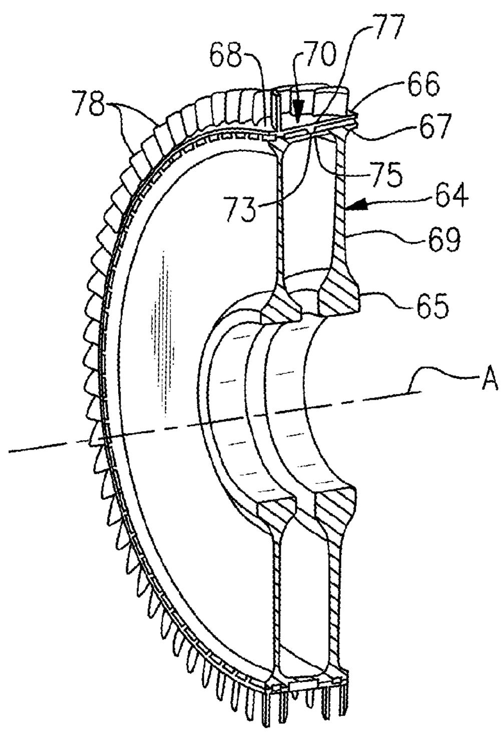



FIG.3B

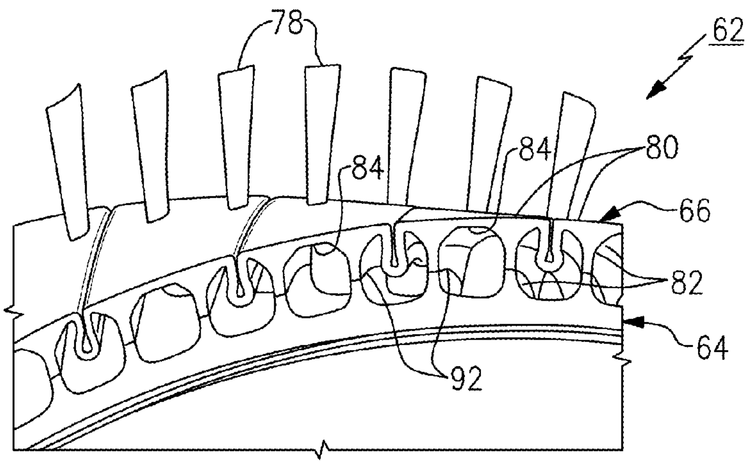



FIG.3C

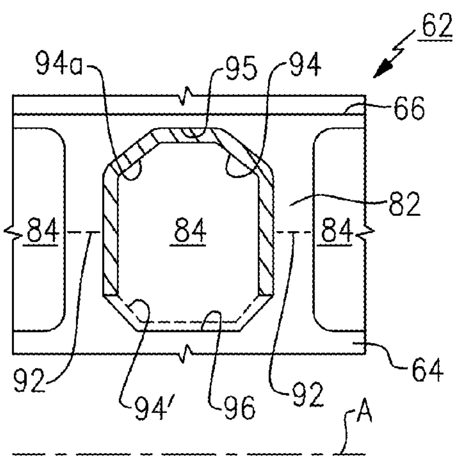



FIG.4A

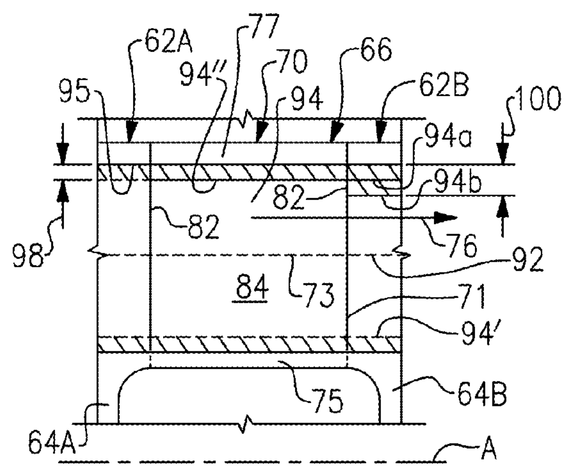



FIG.4B

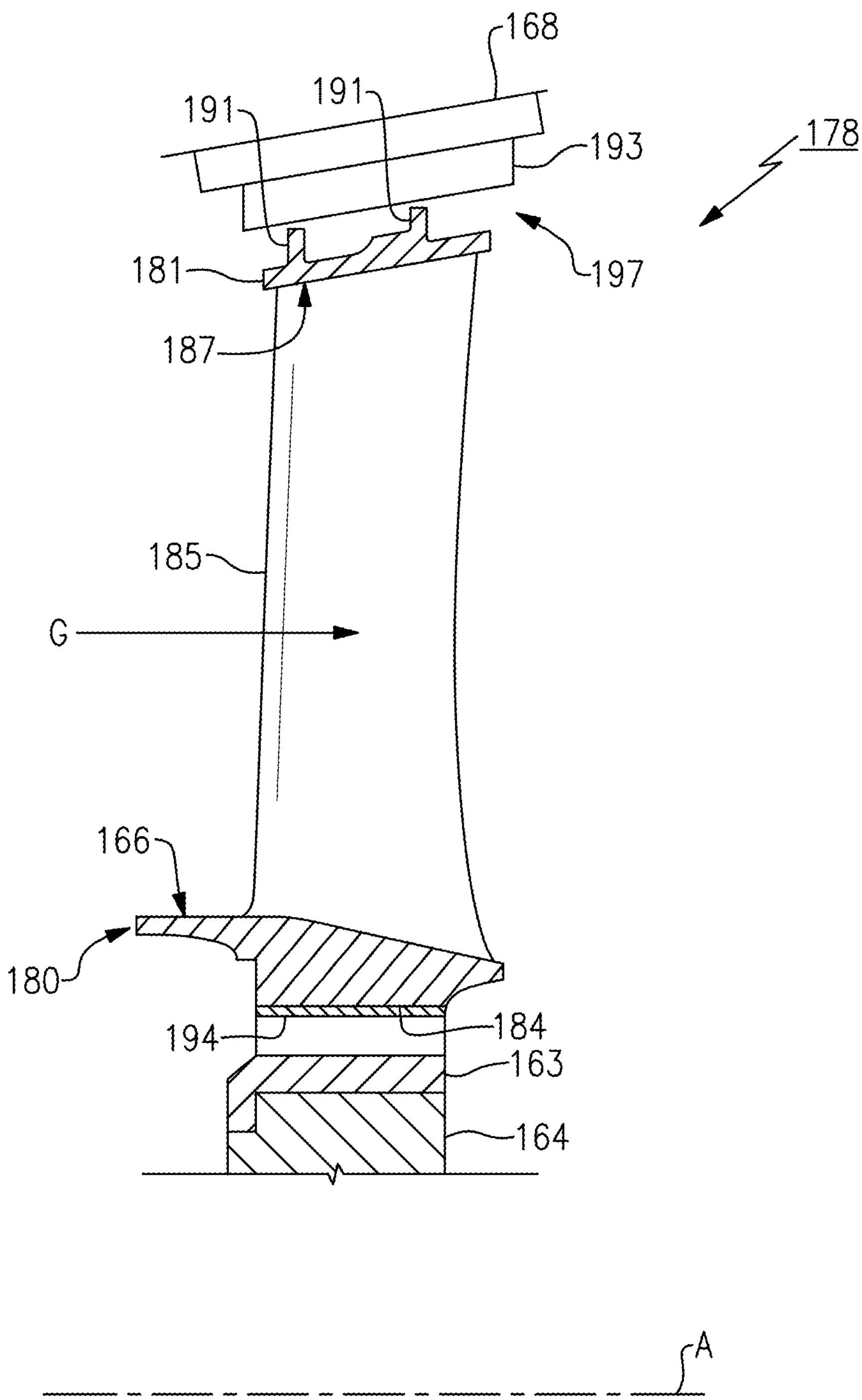



FIG.5

# THERMAL BARRIER COATING INSIDE COOLING CHANNELS

## CROSS-REFERENCE TO RELATED APPLICATION

The present disclosure claims priority to U.S. Provisional Patent Application No. 62/032,039 filed Aug. 1, 2014.

## BACKGROUND

The present disclosure relates to a gas turbine engine and, more particularly, to a cooling arrangement for a rotor assembly.

Typical gas turbine engines include a fan delivering air into a bypass duct as propulsion air. The fan also delivers air into a core flow path of a core engine where it is compressed in a compressor section. Compressed air is then delivered into a combustion section where it is mixed with fuel and ignited. Products of the combustion pass downstream over turbine rotors in a turbine section, driving them to rotate. Typically, but not necessarily, some of the core flow air is used to cool components in the turbine or other sections, as is known in the art.

At least one of the compressor and turbine sections typically includes a rotor assembly. The rotor assembly can operate in an environment in which significant pressure and temperature differentials exist across various portions of the rotor assembly. Some rotor assemblies include a secondary cooling flow path to provide cooling to portions of the rotor assembly.

## SUMMARY

A rotor for a gas turbine engine according to an example of the present disclosure includes a rotor disk rotatable about an axis. A gas path wall is coupled to and radially outward of the rotor disk. The gas path wall bounds a radially inward portion of a gas path. A plurality of rotor spokes are radially intermediate the rotor disk and the gas path wall. The plurality of rotor spokes are circumferentially spaced to define a plurality of cooling channels intermediate the rotor disk and the gas path wall. A thermal barrier coating is disposed on a surface of at least one of the plurality of cooling channels.

In a further embodiment of any of the foregoing embodiments, the thermal barrier coating is disposed on a surface of the gas path wall defining one of the plurality of cooling channels.

In a further embodiment of any of the foregoing embodiments, the thermal barrier coating is disposed on the rotor spoke.

In a further embodiment of any of the foregoing embodiments, the rotor disk and the gas path wall are attached at an interface along the rotor spoke, the thermal barrier coating extending radially outward from the interface.

In a further embodiment of any of the foregoing embodiments, the thermal barrier coating extends radially inward from the interface.

In a further embodiment of any of the foregoing embodiments, surfaces of the gas path wall bounding the gas path are free of the thermal barrier coating.

In a further embodiment of any of the foregoing embodiments, surfaces of the plurality of cooling channels facing radially outwards from the axis are free of the thermal barrier coating.



In a further embodiment of any of the foregoing embodiments, a radially innermost surface of each of the plurality of cooling channels is free of the thermal barrier coating.

A further embodiment of any of the foregoing embodiments includes a plurality of airfoils extending radially outward from the rotor disk. Each of the plurality of airfoils includes a platform forming at least a portion of the gas path wall.

A further embodiment of any of the foregoing embodiments includes a spacer extending axially from the rotor disk. The spacer includes a plurality of spacer spokes circumferentially aligned with the plurality of rotor spokes such that the plurality of cooling channels extend axially from the rotor disk.

A section for a gas turbine engine according to an example of the present disclosure includes an outer gas path wall bounding a radially outward portion of a gas path. A plurality of rotor stages each include a rotor disk rotatable about an axis, and an inner gas path wall coupled to and radially outward of the rotor disk. The inner gas path wall bounds a radially inward portion of the gas path. A plurality of airfoils extends radially outward from the rotor disk. Each of the plurality of airfoils includes a platform forming at least a portion of the inner gas path wall. A plurality of rotor spokes extends radially between the rotor disk and each platform. The plurality of rotor spokes are circumferentially spaced to define a plurality of cooling channels between the rotor disk and each platform. A thermal barrier coating is disposed on a surface of at least one of the plurality of cooling channels.

In a further embodiment of any of the foregoing embodiments, the plurality of rotor stages includes a first rotor stage and a second rotor stage. The thermal barrier coating extends axially along the plurality of cooling channels between at least the first rotor stage and the second rotor stage.

In a further embodiment of any of the foregoing embodiments, the thermal barrier coating is disposed on surfaces of the plurality of cooling channels facing radially inwards and is defined by the first rotor stage and the second rotor stage.

In a further embodiment of any of the foregoing embodiments, a thickness of the thermal barrier coating disposed on the plurality of cooling channels of the first rotor stage is different from a thickness of the thermal barrier coating disposed on the plurality of cooling channels of the second rotor stage.

In a further embodiment of any of the foregoing embodiments, surfaces of the gas path radially between the inner gas path wall and the outer gas path wall are free of the thermal barrier coating.

In a further embodiment of any of the foregoing embodiments, each of the plurality of airfoils includes a shroud extending radially outward from an airfoil section, the shroud forming at least a portion of the outer gas path wall.

A further embodiment of any of the foregoing embodiments includes a spacer extending axially from the rotor disk. The spacer includes a plurality of spacer spokes circumferentially aligned with the plurality of rotor spokes such that the plurality of cooling channels extend axially from the rotor disk.

In a further embodiment of any of the foregoing embodiments, the plurality of cooling channels is connected to receive cooling air from an upstream compressor stage.

A method of cooling a rotor assembly according to an example of the present disclosure includes the steps of providing a rotor disk rotatable about an axis, providing a gas path wall coupled to and radially outward of the rotor disk, the gas path wall bounding a gas path, providing a plurality of spokes intermediate the rotor disk and the gas path wall, the plurality of rotor spokes being circumferentially spaced such that there is a plurality of cooling channels intermediate the rotor disk and the gas path wall, and controlling thermal conductivity through the gas path wall into the plurality of cooling channels by using a thermal barrier coating disposed on at least a portion of the plurality of cooling channels.

A further embodiment of any of the foregoing embodiments includes the step of permitting thermal conductivity between the rotor disk and the plurality of cooling channels.

Although the different examples have the specific components shown in the illustrations, embodiments of this disclosure are not limited to those particular combinations. It is possible to use some of the components or features from one of the examples in combination with features or components from another one of the examples.

The various features and advantages of this 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.

## BRIEF DESCRIPTION OF THE DRAWINGS

Various features will become apparent to those skilled in the art from the following detailed description of the disclosed non-limiting embodiment. The drawings that accompany the detailed description can be briefly described as follows:

FIG. **1** is a schematic cross-sectional view of a gas turbine engine;

FIG. **2** is a partial perspective view of a rotor assembly;

FIG. **3**A is an exploded view of a portion of the rotor assembly of FIG. **2**;

FIG. **3**B is a partial perspective view of the rotor assembly of FIG. **3**A;

FIG. **3**C is an axial view of a portion of the rotor assembly of FIG. **3**A;

FIG. **4**A is a schematic cross section view of a thermal barrier coating arrangement; and

FIG. **4**B is a schematic axial view of the thermal barrier coating arrangement of FIG. **4**A.

FIG. **5** is a side view of a second embodiment of an airfoil having a thermal barrier coating arrangement.

## DETAILED DESCRIPTION

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**. Alternative engines might include an augmentor section (not shown) among other systems or features. The fan section **22** drives air along a bypass flow path B in a bypass duct defined within a nacelle **15**, while the compressor section **24** 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 may be applied to other types of turbine engines including three-spool architectures and ground-based engines that do not have a fan.

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**. 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.

The low speed spool **30** generally includes an inner shaft **40** that interconnects a fan **42**, 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 the 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** is 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.

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 combustor section **26** or even aft of turbine section **28**, and fan section **22** may be positioned forward or aft of the location of gear system **48**.

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. 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.

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. The flight condition of 0.8 Mach and 35,000 ft, with the engine at its best fuel consumption—also known as "bucket cruise Thrust Specific Fuel Consumption ('TSFC')"—is the industry standard parameter of lbm 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 [(Tram ° R)/(518.7° 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.

FIG. **2** illustrates an isolated schematic perspective view of a rotor assembly **60** of the gas turbine engine **20**. In this disclosure, like reference numerals designate like elements where appropriate and reference numerals with the addition of one-hundred or multiples thereof designate modified elements that are understood to incorporate the same features and benefits of the corresponding original elements. In this example, the rotor assembly **60** is located in the compressor section **24**, such as the high pressure compressor **52**. In other examples, the rotor assembly **60** is located in the turbine section **28**. However, other parts of the gas turbine engine **20** may alternatively or additionally benefit from these examples. Other systems may also benefit from the teachings herein, including ground based power generation and marine based systems.

The rotor assembly **60** includes one or more rotor stages **62**. In some examples, the rotor assembly **60** includes a first rotor stage **62**A, a second rotor stage **62**B, a third rotor stage **62**C and a fourth rotor stage **62**D. However, other quantities of rotor stages are contemplated herein. Each of the rotor stages **62** includes vanes **61** (shown in FIG. **1**) within a portion of the core flow path C.

Referring also to FIGS. **3**A and **3**B, each rotor stage **62** includes at least one rotor disc **64**. Each rotor disc **64** includes a hub **65**, arranged concentrically about the axis A, a rim **67**, and a web **69** extending between the hub **65** and the rim **67**. The rims **67** are configured to receive upstream and downstream rims **67** with the last rim **67** joining to the rear hub **65** or outer shaft **50**, for rotation about the axis A.

Each of the rotor stages **62** also includes a core gas path wall **66** radially outward of the rotor disc **64**. The core gas path wall **66** includes a radially outer surface **72** bounding a flow path, such as the core flow path C, and also an opposed radially inner surface **74** bounding a cooling flow path **76** (shown in FIG. **2**). The core gas path wall **66** is configured to minimize fluid communication between the core flow path C and the cooling flow path **76**.

Each rotor disc **64** includes airfoils **78** extending radially outward from each of the rotor discs **64**. Each airfoil **78** includes a platform **80** forming at least a portion of the core gas path wall **66**. In some examples, the airfoils **78** are rotatable blades. In another example, the airfoils **78** are static vanes.

FIG. **3**C illustrates an isolated axial view of a portion of one of the rotor stages **62**. Each of the rotor stages **62** can include one or more rotor spokes **82** intermediate the rotor disc **64** and the core gas path wall **66**. The rotor spokes **82** have a relatively lesser circumferentially thickness than the rotor disc **64** and the core gas path wall **66** to minimize thermal conductivity between the rotor disc **64** and the core gas path wall **66**.

In some examples, the rotor stage **62** is a hybrid dual alloy integrally bladed rotor (IBR) in which the core gas path wall **66** and/or airfoils **78** are manufactured of one type of material and the rotor disk **64** is manufactured of different material. Bi-metal construction provides material capability to separately address different temperature requirements. For example, the airfoils **78** can be manufactured of a single crystal nickel alloy that are transient liquid phase bonded with the rotor disk **64** which is manufactured of a different material such as an extruded billet nickel alloy. Alternatively, or in addition to the different materials, the core gas path wall **66** and/or airfoils **78** may be subject to a first type of heat treat and the rotor disk **64** to a different heat treat. That is, the Bi-metal construction as defined herein includes different chemical compositions as well as different treatments of the same chemical compositions such as that provided by differential heat treatment. The rotor disc **64** and the core gas path wall **66** can be attached at an interface **92** along the rotor spoke **82**. Of course, the rotor disc **64** and core gas path wall **66** can be formed of the same materials, and may also be mechanically attached in another conventional manner.

The rotor spokes **82** are circumferentially spaced about the axis A such that there is a plurality of cooling channels **84** radially intermediate the rotor disc **64** and a radially inner surface of core gas path wall **66**. The cooling channels **84** define a portion of the cooling flow path **76** (shown in FIG. **2**). The cooling channels **84** can have various geometries such as a polygon, a generally circular configuration, or a generally rectangular configuration. However, other geometries of the cooling channels **84** are contemplated and can be selected based on the cooling demands and geometries of the rotor stages **62**.

Referring back to FIGS. **3**A and **3**B, each rotor stage **62** can include at least one spacer **70** having a generally annular geometry and extending axially from at least one rotor disc **64** to bound the core flow path C (shown in FIG. **2**). However, each spacer **70** can have different geometries to bound the core flow path C. The rotor discs **64** and spacers **70** are arranged in a stacked configuration to define a portion of the rotor assembly **60**. Each spacer **70** can be integrally formed with the rotor disc **64** and/or core gas path wall **66**. In other examples, each spacer **70** is a separate component and can include a plurality of sections extending circumferentially around the axis A to define an array.

Each of the spacers **70** can include a plurality of spacer spokes **71** which are circumferentially aligned with the plurality of rotor spokes **82**, such that the plurality of cooling channels **84** extend axially from the rotor disc **64**. Each of the spacers **70** can also define a spacer interface **73** in which an inner portion **75** and an outer portion **77** are attached at the spacer interface **73** using conventional techniques such as those described above in attaching the rotor disc **64** and the core gas path wall **66**.

The cooling channels **84** are connected to receive coolant provided to an inlet **86** (FIG. **2**), through an inlet duct **88**, by a coolant source **89**. In some examples, the coolant source **89** is bleed air from an upstream compressor stage, such as a stage of the low pressure compressor **44**. In other examples, the coolant source **89** is ambient air from the bypass flow path B. The coolant provided to the cooling flow path **76** receives heat rejected from select portions of the rotor assembly **60**, which is then discharged via an outlet **90**. The relatively warm air communicated to the outlet **90** can be provided to another portion of the engine **20**, such as a downstream combustor **56** or a portion of the turbine **28**, or can be discharged overboard the aircraft.

Referring to FIG. **4**A, a thermal barrier coating ("TBC") **94** is disposed on a surface of the cooling flow path **76** (shown in FIG. **2**) to minimize heat transfer between the core gas path wall **66** and the cooling flow path **76**. The thermal barrier coating **94** can be disposed on a surface of at least one of the cooling channels **84** utilizing various techniques. For example, the thermal barrier coating **94** can be disposed on a surface of the cooling channels **84** by air plasma spraying, or chemical vapor deposition. However, other techniques for disposing the thermal barrier coating **94** are contemplated. The thermal barrier coating **99** can be made of various materials such as ceramics, alumina, or zirconia, although other materials or composites are also contemplated. The material of the thermal barrier coating **94** can be selected according to the operating conditions of the rotor assembly **60**, and in some examples, to the operating conditions of each rotor stage **62**. Minimizing heat transfer between the core gas path wall **66** and the cooling flow path **76** can reduce the amount of cooling requirements of the rotor assembly **60** by selectively delivering relatively cooler fluid in the cooling flow path **76** to select portions of the rotor assembly **60**, including the radially inward portions of the rotor stages **62** such as each rotor disc **64**.

The thermal barrier coating **94** can be disposed on various surfaces defining the cooling flow path **76** to provide the general benefits described herein. In some examples, the thermal barrier coating **94** is disposed on at least a radially outermost surface **95** of at least one of the cooling channels **84** to minimize thermal conductivity between the core gas path wall **66** and the cooling flow path **76**. The further examples, the thermal barrier coating **94** extends radially outward from at least one of the interfaces **73**, **92**, and can also extend radially inward from at least one of the interfaces **73**, **92**. In yet other examples, the thermal barrier coating **94** is disposed on each surface of the cooling flow path **76** facing radially inward and being defined by one of the rotor stages **62**. This arrangement advantageously provides cooling to the radially inward portions of the rotor stages **62**, while minimizing thermal conductivity between the core gas path wall **66** and the cooling flow path **76**. Another benefit of disposing the thermal barrier coating **94** on at least a radially outermost surface **95** is that, as the rotor disc **64** rotates about the axis A, the centrifugal loading on the thermal barrier coating **94** minimizes spallation or liberation of the thermal barrier coating **94** from the cooling channel **84**. This permits the engine designer to select a thermal barrier coating having relatively good thermal characteristics without undue consideration of the tensile characteristics of the thermal barrier coating to oppose the centrifugal loading.

In some examples, the radially innermost surface **96** of each of the cooling channels **84** is free of the thermal barrier coating **94**, and in further examples, surfaces of the cooling flow path **76** facing radially outwards from the axis A is also free of the thermal barrier coating **94**. In yet further examples, surfaces of the cooling flow path **76** facing radially outwards from the axis A, such as the radially innermost surface **96** of each of the cooling channels **84**, are free of any thermal barrier coating, including the thermal barrier coating **94** or another thermal barrier coating. In some examples, the surfaces of the cooling flow path **76** facing radially outwards from the axis A are free of any coating. This minimizes the risk of spallation or liberation of the thermal barrier coating **94**, wherein the thermal barrier coating **94** chips off or otherwise disjoins from the radially outer surface **72** of the gas path wall **66** and is communicated through the core flow path C as debris. Debris can undesirably result in surface erosion or other degradation of downstream components in communication with the core flow path C and can disrupt operation of the engine **20**. Disposing a thermal barrier coating on the radially outer surface **72** of the gas path wall **66** may also result in contact with adjacent components such as a stationary vane **61** disposed in the core flow path C due to a rub out condition during transient conditions of the engine **20**. Of course, the thermal barrier coating **94** can be disposed on substantially each surface of the cooling channels **84** while obtaining the general benefits described herein. For example, a thermal barrier coating **94'** is disposed on each surface of a cross-section of at least one of the cooling channels **84** to minimize thermal conductivity between the cooling flow path **76** and the rotor disc **64**. The arrangement of thermal barrier coating **94'** may be desirable to deliver coolant to an adjacent or downstream portion of one of the cooling channels **84** based upon design parameters. However, disposing the thermal barrier coating **94'** on the radially innermost surface **96** of the cooling channel **84** may cause a portion of the thermal barrier coating **94'** to spall due to the centrifugal load observed by rotation of the rotor disc **64** about the axis A during operation of the engine **20**.

Referring to FIG. **4**B, which is a cross-sectional view along axis A of a portion of the rotor assembly **60**, the thermal barrier coating **94** can have different axial arrangements with respect to the rotor assembly **60**. The rotor assembly **60** can be configured such that the thermal barrier coating **94*a*** extends axially along the cooling flow path **76** between at least two rotor stages **62**, such as the first rotor stage **62**A and the second rotor stage **62**B illustrated in FIG. **4**B. The thermal barrier coating **94*a*** can extend axially along at least one of the spacers **70**. The thermal barrier coating **94** can also be disposed on surfaces of the inlet duct **88** and/or the outlet **90** (shown in FIG. **2**). In other examples, an upstream one of the rotor stages **62**, such as rotor stage **62**A, can be free of any thermal barrier coating **94"**, while the thermal barrier coating **94"** can be disposed on a surface of one of the cooling channels **84** of a downstream one of the rotor stages **62**, such as rotor stage **62**B. This arrangement permits the coolant to receive some heat rejected from the core gas path wall **66** of the upstream one of the rotor stages **62** depending on design parameters.

A thickness of thermal barrier coating **94** can also vary axially along the cooling flow path **76**. As illustrated by FIG. **4**B, a thickness **98** of the thermal barrier coating **94** disposed on a surface of the cooling flow path **76** defined by the rotor stage **62**A is different than a thickness **100** of the thermal barrier coating **94** disposed on a surface of the cooling flow path **76** defined by the second rotor stage **62**B. The thickness **100** can be greater than the thickness **98** to minimize thermal conductivity between the core gas path wall **66** and the cooling flow path **76** according to the temperature characteristics and cooling requirements of each of the rotor stages **62**.

The thermal barrier coating **94** can be disposed onto a surface of the cooling flow path **76** as one or more distinct layers. For example, the thermal barrier coating **94** can be formed by at least two layers **94*a***, **94*b***, although more than two layers are contemplated. The layers **94*a***, **94*b*** can include different materials depending on the thermal characteristics of the respective rotor stage **62**, for example. In other examples, the thermal barrier coating **94** is disposed on the cooling channel **84** as a single layer having a substantially uniform thickness. It is contemplated that the thermal barrier coating **94** can be arranged to have a different thickness and can be made of different materials in the radial, circumferential and/or axial directions.

FIG. **5** illustrates a highly schematic view of a second embodiment of an airfoil **178** having a thermal barrier coating arrangement. The airfoil **178** includes an airfoil section **185** and a root section **163** attached to the rotor disc **164** utilizing various techniques known in the art. The airfoil section **185** extends radially between an inner platform **180** forming at least a portion of a core gas path wall **166** and a shroud **181**. The shroud **181** forms at least a portion of an outer gas path wall **187**. The airfoil **178** is configured to rotate about the engine central longitudinal axis A.

The shroud **181** can include one or more sealing elements **191**, such as knife edges, configured to cooperate with a sealing feature **193**. The sealing feature **193** can be an abradable honeycomb structure, for example, to define a seal **197**. In other examples, the sealing elements **191** and the sealing feature **193** can be arranged to define a labyrinth seal. The sealing feature **193** can be a component attached to, or formed in, a casing **168** or another static structure. The casing **168** can be provided by the turbine section **28**, for example. The seal **197** is configured to account for different relative thermal characteristics of the airfoil **178**, the rotor disk **164**, and the casing **168**. Accordingly, the seal **197** minimizes a leakage flow of the combustion products from a gas path G and into radial space between the sealing feature **193** and the shroud **181**. It should be appreciated that other seal arrangements including a shroud are contemplated.

At least one cooling channel **184** is formed in the root section **163** of the airfoil **178**. A thermal barrier coating **194** is disposed on various surfaces of the cooling channel **184** utilizing any of the arrangements disclosed herein. It should also be appreciated that the airfoil **178** arrangement can be utilized in combination with any of the rotor arrangements disclosed herein, such that at least one rotor stage of the engine **20** can include the airfoil **178**. Cooling of the rotor assembly **60** operates as follows. Coolant is provided to the rotor assembly **60** at the inlet **86** and through the inlet duct **88**. The coolant is then communicated from the inlet duct **88** to the cooling flow path **76** defined by the plurality of cooling channels **84**. The coolant accepts heat communicated from portions of the rotor disc **64**. Rather, the arrangement of thermal barrier coating **94** permits thermal conductivity between the core flow path C and the core gas path wall **66**. Thermal conductivity through the core gas path wall **66** into the plurality of cooling channels **84** is controlled by using the thermal barrier coating **94**. Thermal conductivity is permitted between the rotor disc **64** and the cooling flow path **76**, such that the coolant accepts heat rejected from at least one of the rotor disks **64** as the coolant is communicated between the inlet **86** and the outlet **90** of the rotor assembly **60**. The relatively warmer coolant is communicated to the outlet **90**, which can be distributed to another portion of the gas turbine engine **20**, such as a downstream combustor section **56** or the turbine section **28**, or can be discharged overboard.

Although particular step sequences are shown, described, and claimed, it should be understood that steps may be performed in any order, separated or combined unless otherwise indicated and will still benefit from the present disclosure.

It should be understood that relative positional terms such as "forward," "aft," "upper," "lower," "above," "below," and the like are with reference to the normal operational attitude of the vehicle and should not be considered otherwise limiting.

The foregoing description is exemplary rather than defined by the limitations within. Various non-limiting embodiments are disclosed herein, however, one of ordinary skill in the art would recognize that various modifications and variations in light of the above teachings will fall within the scope of the appended claims. It is therefore to be understood that within the scope of the appended claims, the disclosure may be practiced other than as specifically described. For that reason the appended claims should be studied to determine true scope and content.

What is claimed is:

**1**. A rotor for a gas turbine engine comprising:

a rotor disk rotatable about an axis;

a gas path wall coupled to and radially outward of the rotor disk, the gas path wall bounding a radially inward portion of a gas path;

a plurality of rotor spokes radially intermediate the rotor disk and the gas path wall, the plurality of rotor spokes being circumferentially spaced to define a plurality of cooling channels intermediate the rotor disk and the gas path wall; and

a thermal barrier coating disposed on a surface of at least one of the plurality of cooling channels, wherein surfaces of the gas path wall bounding the gas path are free of the thermal barrier coating.

**2**. The rotor as recited in claim **1**, wherein the thermal barrier coating is disposed on a surface of the gas path wall defining one of the plurality of cooling channels.

**3**. The rotor as recited in claim **2**, wherein the thermal barrier coating is disposed on the rotor spoke.

**4**. The rotor as recited in claim **1**, wherein the rotor disk and the gas path wall are attached at an interface along the rotor spoke, the thermal barrier coating extending radially outward from the interface.

**5**. The rotor as recited in claim **4**, wherein the thermal barrier coating extends radially inward from the interface.

**6**. The rotor as recited in claim **1**, wherein surfaces of the plurality of cooling channels facing radially outwards from the axis are free of the thermal barrier coating.

**7**. The rotor as recited in claim **1**, wherein a radially innermost surface of each of the plurality of cooling channels is free of the thermal barrier coating.

**8**. The rotor as recited in claim **1**, further comprising a plurality of airfoils extending radially outward from the rotor disk, each of the plurality of airfoils including a platform forming at least a portion of the gas path wall.

**9**. The rotor as recited in claim **1**, further comprising a spacer extending axially from the rotor disk, the spacer including a plurality of spacer spokes circumferentially aligned with the plurality of rotor spokes such that the plurality of cooling channels extend axially from the rotor disk.

**10**. The rotor as recited in claim **1**, wherein the rotor disk is compressor rotor disk.

**11**. A section for a gas turbine engine comprising:

an outer gas path wall bounding a radially outward portion of a gas path;

a plurality of rotor stages each comprising:

a rotor disk rotatable about an axis;

an inner gas path wall coupled to and radially outward of the rotor disk, the inner gas path wall bounding a radially inward portion of the gas path;

a plurality of airfoils extending radially outward from the rotor disk, each of the plurality of airfoils including a platform forming at least a portion of the inner gas path wall;

a plurality of rotor spokes extending radially between the rotor disk and each platform, the plurality of rotor spokes being circumferentially spaced to define a plurality of cooling channels between the rotor disk and each platform;

a thermal barrier coating disposed on a surface of at least one of the plurality of cooling channels;

wherein the plurality of rotor stages includes a first rotor stage and a second rotor stage, the thermal barrier coating extending axially along the plurality of cooling channels between at least the first rotor stage and the second rotor stage; and

wherein the thermal barrier coating is disposed on surfaces of the plurality of cooling channels facing radially inwards and being defined by the first rotor stage and the second rotor stage.

**12**. The section as recited in claim **11**, wherein a thickness of the thermal barrier coating disposed on the plurality of cooling channels of the first rotor stage is different from a thickness of the thermal barrier coating disposed on the plurality of cooling channels of the second rotor stage.

**13**. The section as recited in claim **11**, wherein surfaces of the gas path radially between the inner gas path wall and the outer gas path wall are free of the thermal barrier coating.

**14**. The section as recited in claim **11**, wherein each of the plurality of airfoils includes a shroud extending radially outward from an airfoil section, the shroud forming at least a portion of the outer gas path wall.

**15**. The section as recited in claim **11**, further comprising a spacer extending axially from the rotor disk, the spacer including a plurality of spacer spokes circumferentially aligned with the plurality of rotor spokes such that the plurality of cooling channels extend axially from the rotor disk.

**16**. The rotor assembly as recited in claim **11**, wherein the plurality of cooling channels is connected to receive cooling air from an upstream compressor stage.

**17**. The section as recited in claim **11**, wherein the plurality of rotor stages are compressor rotor stages.

**18**. The section as recited in claim **11**, wherein surfaces of the plurality of cooling channels facing radially outwards from the axis are free of the thermal barrier coating.

**19**. A method of cooling a rotor assembly comprising the steps of:

providing a rotor disk rotatable about an axis;

providing a gas path wall coupled to and radially outward of the rotor disk, the gas path wall bounding a gas path;

providing a plurality of spokes intermediate the rotor disk and the gas path wall, the plurality of rotor spokes being circumferentially spaced such that there is a plurality of cooling channels intermediate the rotor disk and the gas path wall;

controlling thermal conductivity through the gas path wall into the plurality of cooling channels by using a thermal barrier coating disposed on at least a portion of the plurality of cooling channels; and

wherein surfaces of the plurality of cooling channels facing radially outwards from the axis are free of the thermal barrier coating.

**20**. The method as recited in claim **19**, comprising the step of permitting thermal conductivity between the rotor disk and the plurality of cooling channels.

* * * * *