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COOLING AIR DELIVERY SYSTEM AND METHODS THEREOF

Abstract

A gas turbine engine is provided. The gas turbine engine includes: a compressor section; a combustion section comprising an inner combustor casing defining in part an aft cavity with the compressor section and defining in part a diffuser cavity with an outer combustor casing; and a cooling system having: a compressor discharge pressure duct located outward of the outer combustor casing and positioned in fluid communication with the diffuser cavity, the compressor section, or both for receiving an airflow from the diffuser cavity, from the compressor section, or both; a heat exchanger in thermal communication with the compressor discharge pressure duct for reducing a temperature of the airflow; a cooling duct located inward of the outer combustor casing and in fluid communication with the compressor discharge pressure duct and the aft cavity for receiving the airflow and providing at least a portion of the airflow to the aft cavity.

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Background/Summary

CROSS-REFERENCE TO RELATED APPLICATIONS [0001] This application is a continuation application of U.S. application Ser. No. 17/869,060 filed Jul. 20, 2022, which is hereby incorporated by reference in its entirety.

FIELD
[0002] The present disclosure generally relates to a cooling air delivery system of a gas turbine engine.
BACKGROUND
[0003] A gas turbine engine generally includes a turbomachine and a rotor assembly. Gas turbine engines, such as turbofan engines, may be used for aircraft propulsion. In the case of a turbofan, the rotor assembly may be configured as a fan assembly.
[0004] In gas turbine engines, thermal management systems are incorporated to cool certain components and prevent damage due to overheating. In existing thermal management systems, air ducts passing through a combustion section of the gas turbine engine may be provided to provide a flow of cooling air to a turbine section of the gas turbine engine.

Description

BRIEF DESCRIPTION OF THE DRAWINGS
[0005] A full and enabling disclosure of the present disclosure, including the best mode thereof, directed to one of ordinary skill in the art, is set forth in the specification, which makes reference to the appended figures, in which:
[0006] FIG. **1** is a cross-sectional view of a gas turbine engine in accordance with an exemplary aspect of the present disclosure.
[0007] FIG. **2** is a cross-sectional view of a compressor section and a combustion section of the exemplary gas turbine that shows a cooling system in accordance with an exemplary aspect of the present disclosure.
[0008] FIG. **3**A is a schematic, cross-sectional view of the cooling system of FIG. **2**.
[0009] FIG. **3**B is a schematic, cross-sectional view of a cooling system in accordance with another exemplary aspect of the present disclosure.
[0010] FIG. **4** is a close-up view of a portion of the compressor section, combustion section, and cooling system of FIG. **2**.
[0011] FIG. **5** is a flowchart of a method in accordance with an exemplary aspect of the present disclosure.
DETAILED DESCRIPTION
[0012] Reference will now be made in detail to present embodiments of the disclosure, one or more examples of which are illustrated in the accompanying drawings. The detailed description uses numerical and letter designations to refer to features in the drawings. Like or similar designations in the drawings and description have been used to refer to like or similar parts of the disclosure.
[0013] The word “exemplary” is used herein to mean “serving as an example, instance, or illustration.” Any implementation described herein as “exemplary” is not necessarily to be

construed as preferred or advantageous over other implementations. Additionally, unless specifically identified otherwise, all embodiments described herein should be considered exemplary.
[0014] As used herein, the terms “first”, “second”, and “third” may be used interchangeably to distinguish one component from another and are not intended to signify location or importance of the individual components.
[0015] The terms “forward” and “aft” refer to relative positions within a gas turbine engine or vehicle, and refer to the normal operational attitude of the gas turbine engine or vehicle. For example, with regard to a gas turbine engine, forward refers to a position closer to an engine inlet and aft refers to a position closer to an engine nozzle or exhaust.
[0016] The terms “upstream” and “downstream” refer to the relative direction with respect to fluid flow in a fluid pathway. For example, “upstream” refers to the direction from which the fluid flows, and “downstream” refers to the direction to which the fluid flows.
[0017] The terms “coupled” refers to both direct coupling as well as indirect coupling through one or more intermediate components or features, unless otherwise specified herein.
[0018] The singular forms “a”, “an”, and “the” include plural references unless the context clearly dictates otherwise.
[0019] Here and throughout the specification and claims, range limitations are combined and interchanged, such ranges are identified and include all the sub-ranges contained therein unless context or language indicates otherwise. For example, all ranges disclosed herein are inclusive of the endpoints, and the endpoints are independently combinable with each other.
[0020] The terms “low” and “high”, or their respective comparative degrees (e.g., -er, where applicable), when used with a compressor, a turbine, a shaft, or spool components, etc. each refer to relative speeds within an engine unless otherwise specified. For example, a “low turbine” or “low speed turbine” defines a component configured to operate at a rotational speed, such as a maximum allowable rotational speed, lower than a “high turbine” or “high speed turbine” at the engine.
[0021] The term “turbomachine” refers to a machine including one or more compressors, a heat generating section (e.g., a combustion section), and one or more turbines that together generate a torque output.
[0022] The term “gas turbine engine” refers to an engine having a turbomachine as all or a portion of its power source. Example gas turbine engines include gas turbine engines, turboprop engines, turbojet engines, turboshaft engines, etc.
[0023] The term “combustion section” refers to any heat addition system for a turbomachine. For example, the term combustion section may refer to a section including one or more of a deflagrative combustion assembly, a rotating detonation combustion assembly, a pulse detonation combustion assembly, or other appropriate heat addition assembly. In certain example embodiments, the combustion section may include an annular combustor, a can combustor, a cannular combustor, a trapped vortex combustor (TVC), or other appropriate combustion system, or combinations thereof.
[0024] The present disclosure is generally related to a cooling system for a gas turbine engine with a compressor section, a combustion section, and a turbine section. The cooling system may include air ducts that provide a flow of cooling air to one or more sections of the gas turbine engine to keep internal components of the gas turbine engine in an improved working environment. Accordingly, the inventors of the present disclosure have found that improvements to the cooling system would be beneficial.
[0025] The disclosure presents a cooling system that includes a compressor discharge pressure duct located outward of an outer combustor casing and in fluid communication with the diffuser cavity for receiving an airflow from the diffuser cavity; a heat exchanger in thermal communication with the compressor discharge pressure duct for reducing a temperature of the airflow; and a cooling duct located inward of the outer combustor casing and in fluid communication with the compressor

discharge pressure duct for receiving the airflow. The cooling duct is further in fluid communication with an aft cavity for providing at least a portion of the airflow to the aft cavity. The aft cavity is defined in part by an inner combustor casing and a compressor section (e.g., a high-pressure compressor) of the gas turbine engine. In such a manner, the cooling system may provide cooled airflow to the aft cavity during operation of the gas turbine engine. The cooling system may further provide a second portion of the airflow to a turbine section of the gas turbine engine (e.g., a forward wheelspace cavity) for cooling, e.g., a stage 1 nozzle, high-pressure turbine rotor blades, etc.

[0026] As will be appreciated, it may generally be desirable to increase an overall pressure ratio (OPR, which is a ratio of a pressure at a forward end of the compressor to a pressure at an aft end of the compressor) of a gas turbine engine, e.g., in order to improve an overall efficiency of the gas turbine engine. With the increase in OPR, a compressor exit temperature may also increase, particularly at high power operating conditions (e.g., takeoff, climb, etc.). Inclusion of a cooling system of the present disclosure may allow for an increased OPR by providing the cooled airflow to, e.g., the aft cavity to cool the hotter components of the compressor section.

[0027] Further aspects of the present disclosure may allow for modulation of the airflow provided, e.g., to the aft cavity, such that the airflow is only provided when needed, improving an efficiency of the gas turbine engine.

[0028] Referring now to the drawings, wherein identical numerals indicate the same elements throughout the figures, FIG. **1** is a schematic, cross-sectional view of a propulsion system **10** in accordance with an exemplary embodiment of the present disclosure. More particularly, for the embodiment of FIG. **1**, propulsion system **10** includes a gas turbine engine **12**. In one example, gas turbine engine **12** may be a turbofan engine, such as a high-bypass gas turbine jet engine. As shown in FIG. **1**, gas turbine engine **12** defines an axial direction A (extending parallel to longitudinal centerline **14** provided for reference) and a radial direction R. In general, gas turbine engine **12** includes a fan section **16** and a turbomachine **18** disposed downstream from fan section **16**.

[0029] The exemplary turbomachine **18** depicted generally includes an outer casing **20** that defines an annular inlet **22** and is substantially tubular. Outer casing **20** encases, in serial flow order/relationship, a compressor section **23** including a booster or low-pressure compressor **24** and a high-pressure compressor **26**; a combustion section **28**; and a turbine section **29** including a high-pressure turbine **30** and a low-pressure turbine **32**. A high-pressure shaft **34** drivingly connects high-pressure turbine **30** to high-pressure compressor **26**. A low-pressure shaft **36** drivingly connects low-pressure turbine **32** to low-pressure compressor **24**.

[0030] For the embodiment depicted, fan section **16** includes a variable pitch fan **38** having a plurality of fan blades **40** coupled to a disk **42** in a spaced apart manner. As depicted, fan blades **40** extend outwardly from disk **42** generally along radial direction R. Each fan blade **40** is rotatable relative to disk **42** about a pitch axis P by virtue of fan blades **40** being operatively coupled to a suitable actuation member **44** configured to collectively vary the pitch of fan blades **40**, e.g., in unison. Fan blades **40**, disk **42**, and actuation member **44** are together rotatable about longitudinal centerline **14** by low-pressure shaft **36** across a power gear box **46**. Power gear box **46** includes a plurality of gears for stepping down the rotational speed of low-pressure shaft **36** to a more efficient rotational fan speed.

[0031] Referring still to the exemplary embodiment of FIG. **1**, disk **42** is covered by a rotatable front hub **48** aerodynamically contoured to promote an airflow through the plurality of fan blades **40**. Additionally, fan section **16** includes an annular fan casing or outer nacelle **50** that circumferentially surrounds variable pitch fan **38** and/or at least a portion of turbomachine **18**. It should be appreciated that in some embodiments, nacelle **50** is configured to be supported relative to turbomachine **18** by a plurality of circumferentially spaced outlet guide vanes **52**. Moreover, a downstream section **54** of nacelle **50** extends over an outer portion of turbomachine **18** so as to define a bypass airflow passage **56** therebetween.

[0032] During operation of gas turbine engine **12**, a volume of air **58** enters gas turbine engine **12** through an associated inlet **60** of nacelle **50** and/or fan section **16**. As the volume of air **58** passes across fan blades **40**, a first portion of air **58** as indicated by arrows **62** is directed or routed into bypass airflow passage **56** and a second portion of air **58** as indicated by arrow **64** is directed or routed into low-pressure compressor **24**. The ratio between first portion of air **62** and second portion of air **64** is commonly known as a bypass ratio. The pressure of second portion of air **64** is then increased as it is routed through high-pressure compressor **26** and into combustion section **28**, where it is mixed with fuel and burned to provide combustion gases **66**. Subsequently, combustion gases **66** are routed through high-pressure turbine **30** and low-pressure turbine **32**, where a portion of thermal and/or kinetic energy from combustion gases **66** is extracted.

[0033] Combustion gases **66** are then routed through combustion section **28** of turbomachine **18** to provide propulsive thrust. Simultaneously, the pressure of the first portion of air **62** is substantially increased as first portion of air **62** is routed through bypass airflow passage **56** before it is exhausted from fan nozzle exhaust section **68** of gas turbine engine **12**, also providing propulsive thrust.

[0034] It should be appreciated, however, that gas turbine engine **12** depicted in FIG. **1** is by way of example only, and that in other exemplary embodiments, aspects of the present disclosure may additionally, or alternatively, be applied to any other suitable gas turbine engine. For example, in other exemplary embodiments, gas turbine engine **12** may instead be any other suitable aeronautical gas turbine engine, such as a turbojet engine, turboshaft engine, turboprop engine, etc. Additionally, in still other exemplary embodiments, gas turbine engine **12** may include or be operably connected to any other suitable accessory systems.

[0035] Referring now to FIG. **2**, FIG. **2** is a cross-sectional view of a portion of turbomachine **18** and shows high-pressure compressor **26**, combustion section **28**, and high-pressure turbine **30**. The turbomachine **18** is shown with a forward direction to the left and an aft direction to the right (not labeled).

[0036] As shown in FIG. **2**, high-pressure compressor **26** includes a diffuser nozzle **70** and defines an aft cavity **72**. Diffuser nozzle **70** is disposed at an aft end of high-pressure compressor **26** and straightens or re-directs a flow of air flowing from high-pressure compressor **26** to combustion section **28**. Aft cavity **72** is also disposed at an aft end of high-pressure compressor **26** and stores and supplies, e.g., cooling air for use by high-pressure compressor **26**. More specifically, for the embodiment shown, the turbomachine **18** includes the high-pressure shaft **34**, the high-pressure compressor **26** includes an aft cone **21** connecting the high-pressure shaft **34** to, e.g., rotor blades of the high-pressure compressor **26** (see FIG. **1**), and the combustion section **28** includes an inner combustor casing **75**. The inner combustor casing **75** and aft cone **21** together define at least in part the aft cavity **72**.

[0037] Combustion section **28** further includes an outer combustor casing **74** and a combustor **80**. The outer combustor casing **74** is generally outward of the combustor **80** along the radial direction R and the inner combustor casing **75** is generally inward of the combustor **80** along the radial direction R. The outer combustor casing **74** of combustion section **28** defines at least one fluid opening **76** and, at least in part, a chamber configured to house the combustor **80**, referred to herein as a diffuser cavity **78**. The gas turbine engine **12** further includes a cooling system **200** that includes a heat exchanger **140** and a compressor discharge pressure ("CDP") duct **82** for carrying compressor discharge air **210** to the heat exchanger **140**. Generally, "compressor discharge air" refers to pressurized air generated by the high-pressure compressor **26** during operation. The heat exchanger **140** may use a cooling fluid **212** to cool the compressor discharge air **210**. The cooling fluid **212** may be, e.g., a supercritical fluid, such as a supercritical CO.sub.2 ("sCO.sub.2"). However, in other embodiments any other suitable cooling fluid may be utilized.

[0038] Moreover, the at least one fluid opening **76** defined by the outer combustor casing **74** includes a first fluid opening **76**A and a second fluid opening **76**B. The first and second fluid openings **76**A, **76**B each extend through a portion of the outer combustor casing **74**. The CDP duct **82** extends generally from the first fluid opening **76**A to the second fluid opening **76**B.

[0039] In addition to the CDP duct **82**, the cooling system **200** also includes a cooling duct. More specifically, the cooling system **200** also includes a first cooling duct **220** and a second cooling duct **240**. The first cooling duct **220** and the second cooling duct **240** are in fluid communication with the CDP duct **82** through the at least one fluid opening **76**, and more specifically through the second fluid opening **76**B in the embodiment depicted.

[0040] Briefly, the CDP duct **82**, the first cooling duct **220**, and the second cooling duct **240** are pipes or conduits. However, in other embodiments, any suitable structure may be provided for the respective ducts to transport a fluid flow.

[0041] As noted, the CDP duct **82** is connected to at least a portion of the outer combustor casing **74** and is also in fluid communication with the first fluid opening **76**A in the outer combustor casing **74**. The first fluid opening **76** is configured to communicate a flow **84** of cooled air from the heat exchanger **140** through the outer combustor casing **74** into the first cooling duct **220** and/or the second cooling duct **240**. The second fluid opening **76**B may be a single opening or may be multiple openings.

[0042] More specifically, for the embodiment depicted, the first fluid opening **76**A is in fluid communication with the diffuser cavity **78** for receiving the compressor discharge air **210** from the diffuser cavity **78** and providing such airflow to and through the CDP duct **82**. The second fluid opening **76**B is in fluid communication with the CDP duct **82** and heat exchanger **140** for providing a flow **84** of cooled air from the heat exchanger **140** to the remaining portions of the cooling system **200** (e.g., the first cooling duct **220** and the second cooling duct **240**).

[0043] It will be appreciated, however, that in other exemplary embodiments, the CDP duct **82** may instead extend from the compressor section **23** to the heat exchanger **140**, and to the at least one fluid opening **76**. For example, a portion of CDP duct **82** may alternatively be configured to receive compressor bleed airflow from a location within the compressor section **23**.

[0044] Referring still to FIG. **2**, the first cooling duct **220** and/the second cooling duct **240** extend through a portion of diffuser cavity **78** and pass between high-pressure compressor **26** and the combustor **80**. The first cooling duct **220** and/or the second cooling duct **240** are configured to transport a first flow of cooled air **222** and/or a second flow of cooled air **242**, respectively, from the at least one fluid opening **76** and to one or more sections of the gas turbine engine **12**. In certain exemplary embodiments, the first cooling duct **220** and/or the second cooling duct **240** can deliver the first flow of cooled air **222** and/or the second flow of cooled air **242** to a portion of or to components of the gas turbine engine **12**, e.g., to components relating to the high-pressure turbine **30**. In additional embodiments, the first cooling duct **220** and/or the second cooling duct **240** may further include transporting the first flow of cooled air **222** and/or the second flow of cooled air **242** through the diffuser cavity **78** before or after transporting the first flow of cooled air **222** and/or the second flow of cooled air **242** to the one or more sections of the gas turbine engine **12**.

[0045] As used herein, the first flow of cooled air **222** and/or the second flow of cooled air **242**, may be denoted collectively as flow **84** of cooled air, such that the first flow of cooled air **222** may be referred to as a first portion of flow **84**, and the second flow of cooled air **242** may be referred to as a second portion of flow **84**. The flow **84** of cooled air, and subsequently, the first flow of cooled air **222** and the second flow of cooled air **242**, is a flow of cooled cooling air from a cooled cooling air heat exchanger, e.g., heat exchanger **140**, into the gas turbine engine **12**. It will be appreciated, however, that in other exemplary embodiments, flow **84** of cooled air can come from other sources of cooling air such as an ambient source, a bleed air source, a thermal management system of propulsion system **10**, or other air sources.

[0046] In this example, a first end **88** of the first cooling duct **220** is disposed proximate to a portion of the outer combustor casing **74** that surrounds the second fluid opening **76**B and a second end **90** of the first cooling duct **220** is positioned proximate the aft cavity **72**. Similar, a first end **92** of the second cooling duct **240** is positioned proximate the outer combustor casing **74** that surrounds the second fluid opening **76**B and a second end **94** is positioned proximate the HP turbine **30**. As used herein, the term “proximate” refers to being closer to one object than another object (e.g., closer to the outer combustor casing **74** than the HP turbine **30**).

[0047] In the depicted embodiment, the high-pressure turbine **30** includes an inlet guide vane **100** and a first stage blade **102**. In FIG. **2**, a single inlet guide vane **100** and the first stage blade **102** are shown. However, it will be appreciated that gas turbine engine **12** includes a plurality of inlet guide vanes **100** and a plurality of first blades including the first stage blade **102** extending around a circumferential direction of longitudinal centerline **14** (see also, FIG. **3**A). Inlet guide vane **100** is a stationary airfoil for guiding or redirecting a flow of fluid passing across inlet guide vane **100**. Here, inlet guide vane **100** straightens a or changes a direction of a flow of combustion gasses passing from combustion section **28** to high-pressure turbine **30**. First stage blade **102** is an airfoil configured to rotate (e.g., with a rotor disk **108**) about longitudinal centerline **14**. For example, as combustion gasses are expelled from combustor **80**, the combustion gasses push against the first stage blade **102**, causing the first stage blade **102** to rotate about longitudinal centerline **14**.

[0048] In between inlet guide vane **100** and the first stage blade **102**, a forward wheelspace cavity **104** is formed. More specifically, in at least certain exemplary aspects, forward wheelspace cavity **104** is defined and formed by components corresponding to inlet guide vane **100** and the first stage blade **102**, such as a frame **106** supporting inlet guide vane **100** and rotor disk **108** to which first stage rotor blade is attached.

[0049] Further, the gas turbine engine **12** may include an assembly for providing flow **84** of cooled air from the second cooling duct **240** to the rotor disk **108** and first stage turbine blades **102**. More specifically, frame **106** includes a nozzle **103** configured to turn the flow of flow **84** of cooled air from the second cooling duct **240** to at least partially match a rotation of the rotor disk **108**. Further, rotor disk **108** includes a rotating seal **105** rotatable therewith. The air from nozzle **103** is then provided through an opening **107** defined in the rotating seal **105** to an inlet **109** to an internal cooling duct **111** defined within the rotor disk **108** (depicted in phantom). Flow **84** of cooled air may then be provided through the rotor disk **108** to the first stage turbine blades **102**.

[0050] Combustion section **28** also defines a forward shaft outer cavity **110**. Forward shaft outer cavity **110** is disposed generally inward along radial direction R from the second cooling duct **240**. Forward shaft outer cavity **110** is fluidly connected to high-pressure turbine **30** via forward wheelspace cavity **104**, and is further separated from the aft cavity **72** by the rotating seal **105**.

[0051] As already mentioned, the cooling system **200** discussed herein includes the first cooling duct **220** and the second cooling duct **240**. The first cooling duct **220** and the second cooling duct **240** may present as two separate circuits for flow **84** of cooled air that reduces a temperature within particular sections of the gas turbine engine **12**, e.g., high-pressure compressor section **26**, combustion section **28**, and/or high-pressure turbine section **30**. A reduction in temperature provided by the cooling system **200** may preserve the longevity of the components within the gas turbine engine **12** and help maintain efficiency in operation of the gas turbine engine **12**. Additionally, it will be appreciated that although the first cooling duct **220** and the second cooling duct **240** are shown directly adjacent, e.g., without a gap along the length of each duct in FIG. **2**, the first cooling duct **220** and the second cooling duct **240** may be separated by a gap for at least a part of the length of each duct, or alternatively may share a common duct for at least a portion of a length of the first duct **220** (e.g., splitting into separate ducts at a manifold).

[0052] Referring still to FIG. **2**, the first cooling duct **220** includes a first manifold **256** extending in a circumferential direction C and a first duct branch **258** extending from the first manifold **256** to the aft cavity **72** (or rather to the inner combustor casing **75**). Although a single first duct branch **258** is depicted, the first cooling duct **220** may include a plurality of first duct branches **258** spaced along the circumferential direction C (see, e.g., FIGS. **3**A and **3**B).

[0053] Similarly, the second cooling duct **240** includes a second manifold **260** extending in a circumferential direction C and a second duct branch **261** extending from the second manifold **260** towards the HP turbine. Although a single second duct branch **261** is depicted, the second cooling duct **240** may include a plurality of second duct branch **261** spaced along the circumferential direction C (see, e.g., FIGS. **3**A and **3**B).

[0054] Referring now briefly to FIG. **3**A, a cross-sectional, schematic view is provided of the cooling system **200** and diffuser cavity **78** of FIG. **2**, as taken through the first and second manifolds **256**, **260**. As shown, for the embodiment depicted, the first manifold **256** extends in the circumferential direction C, and more specifically, extends **360** degrees in the circumferential direction C about the longitudinal centerline **14** of the gas turbine engine **12**. The first cooling duct **220** includes a plurality of the first duct branches **258** (depicted schematically) extending from the first manifold **256** and spaced along the circumferential direction C. Similarly, the second manifold **260** extends in the circumferential direction C, and more specifically, extends **360** degrees in the circumferential direction C about the longitudinal centerline **14** of the gas turbine engine **12**. The second cooling duct **240** includes a plurality of the second duct branches **261** (depicted schematically) extending from the second manifold **260** and spaced along the circumferential direction C.

[0055] However, in other embodiments, the cooling system **200** may be configured in any other suitable manner. For example, referring now to FIG. **3**B, a cross-sectional, schematic view is provided of a cooling system **200** and diffuser cavity **78** in accordance with another exemplary embodiment of the present disclosure is provided, as taken through a first manifold **256** and a second manifold **260**. The exemplary cooling system **200** of FIG. **3**B is configured in a similar manner as the exemplary cooling system **200** of FIG. **3**A, and the same or similar numbers may refer to the same or similar parts.

[0056] However, for the embodiment of FIG. **3**B, the first manifold **256** of a first cooling duct **220** extends less than **360** degrees in a circumferential direction C about a longitudinal centerline **14** of the gas turbine engine **12**, and similarly the second manifold **260** of a second cooling duct **240** also extends less than **360** degrees in the circumferential direction C about the longitudinal centerline **14** of the gas turbine engine **12**. In particular, for the embodiment depicted, the first manifold **256** extends between **30** degrees and **270** degrees, such as between **90** degrees and **200** degrees, such as between **120** degrees and **180** degrees. Similarly, for the embodiment depicted, the second manifold **260** extends between **30** degrees and **270** degrees, such as between **90** degrees and **200** degrees, such as between **120** degrees and **180** degrees.

[0057] Referring now back to FIG. **2** and also to FIG. **4**, a close-up, cross-sectional view of a portion of the turbomachine **18** is shown including a portion of the cooling system **200** of FIG. **2**. In particular, FIG. **4** provides a close-up view of the first cooling duct **220**.

[0058] In the embodiment depicted, the first cooling duct **220** is configured to transport the first flow of cooled air **222** to the high-pressure compressor **26**. More specifically, the first cooling duct **220** is configured to transport the first flow of cooled air **222** to the aft cavity **72**, where the aft cavity **72** is disposed at an aft end of the high-pressure compressor **26** (see FIG. **2**), as described above. As the high-pressure compressor **26** may experience relatively extreme operating conditions (e.g., high temperatures and high-pressures, coupled with high rotational speeds), being able to reduce the temperature within the high-pressure compressor **26** or of one or more components supporting the high-pressure compressor **26** would improve efficiency and longevity of the gas turbine engine **12**.

[0059] Additionally, in some exemplary embodiments, the first cooling duct **220** may be configured to provide the first flow of cooled air **222** to the aft cavity **72**. The first flow of cooled air **222** travels along the aft cone **21** and up to an aft-most stage of HP compressor rotor blades **101** of the high-pressure compressor **26** (see FIG. **2**). The first flow of cooled air **222** may be provided through a bore of the aft-most stage and into the HP compressor rotor blades **101** (e.g., in a similar manner as the airflow travels into the first stage blade **102**, described above with reference to FIG.

2). In such a manner, the first flow of cooled air **222** may reduce the temperature of the HP compressor rotor blades **101** during operation of the gas turbine engine **12** at an engine condition (e.g., as compared to when the cooling system **200** is not operating at the same engine condition).

[0060] Additionally, for the embodiment depicted the second cooling duct **240** is in fluid communication with the high-pressure turbine **30** of the turbine section **29** for cooling the high-pressure turbine **30** of the turbine section **29**. As mentioned above, the high-pressure turbine **30** includes an inlet guide vane **100** and a first stage blade **102**, with the inlet guide vane **100** and the first stage blade **102** defining a forward wheelspace cavity **104**. In some embodiments, the second cooling duct **240** is configured to transport the second flow of cooled air **242** to the forward wheelspace cavity **104** to cool the forward wheelspace cavity **104** and/or the high-pressure turbine **30**. As mentioned above regarding the high-pressure compressor **26**, cooling the high-pressure turbine **30** may also help improve efficiency and longevity of the gas turbine engine **12**.

[0061] Additionally, a portion of the first flow of cooled air **222** flows (and, in some embodiments, leaks) from the aft cavity **72** into the forward wheelspace **104**. In particular, in the embodiment depicted, the gas turbine engine **12** includes a seal **264** positioned between the inner combustor casing **75** and the high-pressure shaft **34**. A portion of the cooled air **222** may flow through the seal **265** to the forward shaft outer cavity **110**, and through the forward shaft outer cavity **110** to the high-pressure turbine **30**.

[0062] However, it will be appreciated that the first cooling duct **220** and the second cooling duct **240** may be rerouted to transport the first flow of cooled air **222** and/or the second flow of cooled air **242** to other parts of the gas turbine engine **12**, such as other sections of the gas turbine engine **12** that may be subject to extreme operating conditions. For example, the cooling system **200** may be used to cool a last stage rotor blade, e.g., the compressor rotor blade **101** in FIG. **2**. In other embodiments, the cooling system **200** may additionally and/or alternatively be used to cool other components of the gas turbine engine **12**.

[0063] The range of cooling of the cooling system **200** depends at least in part on the material of the section in which the cool air is being delivered to. For example, the first flow of cooled air **222** is being transported to the high-pressure compressor **26**, and in particular to the aft cavity **72**, in the embodiment depicted. The high-pressure compressor **26** includes components exposed to a working gas flowpath through the turbomachine **16** (e.g., the aft-most stage of high-pressure compressor rotor blades **101**; see FIG. **2**) formed of a material, the material having a material temperature limit. “Material temperature limit” as used herein may refer to the highest temperatures that a material can withstand without damage (e.g., inelastic damage under normal loads, such as creep; changes in chemical structures; premature degradation). In the embodiment depicted, the cooling system **200** is configured to provide the first flow of cooled air **222** through the first cooling duct **220** at a temperature (in degrees F.) 85% or less of the material temperature limit, such as 70% or less, such as 55% or less, such as 40% or less, such as 30% or less. The range of cooling of the cooling system **200** may additionally depend at least in part on the percentage of the flow **84** that can be transported through the cooling system **200**. For example, as noted, the component may be the aft-most stage of HP compressor rotor blades, and the material temperature limit may be ranging from 1200 degrees F. to 1400 degrees F.

[0064] Referring still to FIGS. **2** and **4**, it will be appreciated that in some exemplary embodiments, at least one of the first cooling duct **220** and the second cooling duct **240** is modulated, e.g., adjustable. For example, the cooling system **200** generally includes at least one sensor **245** for sensing compressor exit temperature is included in the cooling system **200** and a controller **248** operably connected to the sensor **245** (see FIG. **2**). The sensor **245** may be configured to sense data indicative of a temperature and/or a pressure within the compressor section **23**, a rotational speed of one or more aspects of the compressor section **23**, etc. In particular, the sensor **245** may be configured to sense data indicative of one or more operative conditions of the gas turbine engine **12**. It will be appreciated that, in additional or alternative embodiments, the at least one sensor **245** may include a plurality of sensors, such as two or more sensors, such as three or more sensors, and/or such as four or more sensors. In addition, the plurality of sensors may be located throughout the interior of the gas turbine engine **12** in order to sense data indicative of the temperature in specific sections of the gas turbine engine **12**.

[0065] Also, as shown in FIG. **4**, the cooling system **200** further includes at least one valve **250**. The at least one valve **250** may operate based on the operative conditions sensed by the sensor **245**. The controller **248** may also be operably coupled to the at least one sensor **245** and the at least one valve **250**.

[0066] In the embodiment depicted, the valve **250** is positioned in the first duct branch **258** of the first cooling duct **220**, and one or more seals **252**A, **252**B are provided where the first duct branch **258** couples with the inner combustor casing **75**. In such a manner, the cooling system **200** may include a plurality of valve **250**, with each valve **250** of the plurality of valves **250** in a respective first duct branch **258**.

[0067] Notably, in other exemplary embodiments, however, the valve **250** may be located upstream of the first duct manifold **256**.

[0068] As noted, the exemplary controller **248** depicted in FIGS. **2** and **4** is configured to receive the data sensed from the at least one sensor **245** and, e.g., may make control decisions for the cooling system **200** based on the received data. In one or more exemplary embodiments, the controller **248** depicted in FIG. **2** may be a stand-alone controller **248** for the cooling system **200**, or alternatively, may be integrated into one or more of a controller for the gas turbine engine **12** with which the cooling system **200** is integrated, a controller for an aircraft including the gas turbine engine **12** with which the cooling system **200** is integrated, etc.

[0069] Referring particularly to the operation of the controller **248**, in at least certain embodiments, the controller **248** can include one or more computing device(s) **144**. The computing device(s) **144** can include one or more processor(s) **144**A and one or more memory device(s) **144**B. The one or more processor(s) **144**A can include any suitable processing device, such as a microprocessor, microcontroller, integrated circuit, logic device, and/or other suitable processing device. The one or more memory device(s) **144**B can include one or more computer-readable media, including, but not limited to, non-transitory computer-readable media, RAM, ROM, hard drives, flash drives, and/or other memory devices.

[0070] The one or more memory device(s) **144**B can store information accessible by the one or more processor(s) **144**A, including computer-readable instructions **144**C that can be executed by the one or more processor(s) **144**A. The instructions **144**C can be any set of instructions that when executed by the one or more processor(s) **144**A, cause the one or more processor(s) **144**A to perform operations. In some embodiments, the instructions **144**C can be executed by the one or more processor(s) **144**A to cause the one or more processor(s) **144**A to perform operations, such as any of the operations and functions for which the controller **248** and/or the computing device(s) **144** are configured, the operations for operating the cooling system **200** (e.g., method **300**), as described herein, and/or any other operations or functions of the one or more computing device(s) **144**. The instructions **144**C can be software written in any suitable programming language or can be implemented in hardware. Additionally, and/or alternatively, the instructions **144**C can be executed in logically and/or virtually separate threads on processor(s) **144**A. The memory device(s) **144**B can further store data **144**D that can be accessed by the processor(s) **144**A. For example, the data **144**D can include data indicative of power flows, data indicative of engine/aircraft operating conditions, and/or any other data and/or information described herein.

[0071] The computing device(s) **144** can also include a network interface **144**E used to communicate, for example, with the other components of the cooling system **200** (e.g., via a communication network). The network interface **144**E can include any suitable components for interfacing with one or more network(s), including for example, transmitters, receivers, ports, controllers, antennas, and/or other suitable components. One or more devices can be configured to receive one or more commands from the computing device(s) **144** or provide one or more commands to the computing device(s) **144**.

[0072] The network interface **144**E can include any suitable components for interfacing with one or more network(s), including for example, transmitters, receivers, ports, controllers, antennas, and/or other suitable components. The technology discussed herein makes reference to computer-based systems and actions taken by and information sent to and from computer-based systems. One of ordinary skill in the art will recognize that the inherent flexibility of computer-based systems allows for a great variety of possible configurations, combinations, and divisions of tasks and functionality between and among components. For instance, processes discussed herein can be implemented using a single computing device or multiple computing devices working in combination. Databases, memory, instructions, and applications can be implemented on a single system or distributed across multiple systems. Distributed components can operate sequentially or in parallel.

[0073] The operative conditions sensed by the at least one sensor **245** may include at least one of the following: a high operating temperature condition; a high-pressure condition; a supersonic cruise condition; and a takeoff condition or a climb condition. The high operating temperature condition may refer to an internal temperature of a section of the gas turbine engine **12** where the internal temperature extremely high, e.g., when the temperature is over **1000** degrees Fahrenheit, over **1200** degrees Fahrenheit, over **1300** degrees Fahrenheit, over **1302** degrees Fahrenheit, and/or over **1306** degrees Fahrenheit. Additionally, if the gas turbine engine **12** is in the supersonic cruise mode for a significant amount of time, the valve **250** may be activated to cool the sections of the gas turbine engine **12** in need of cooling. Further, when the gas turbine engine **12** is in takeoff mode, e.g., where lots of air may be needed to provide enough thrust for the propulsion system **10**, there may be a built up in pressure and/or increased temperatures within the high-pressure compressor section **26**, where cooling air would be helpful.

[0074] Referring now to FIG. **5**, a flowchart of a method **300** of managing thermal energy in the gas turbine engine **12** is shown. The method **300** generally includes, at **310**, receiving an airflow from a diffuser cavity defined by an outer combustor casing of the gas turbine engine; at **320**, cooling the airflow received from the diffuser cavity with a heat exchanger; at **330**, providing the airflow from the heat exchanger to a cooling duct located inward of the outer combustor casing; and at **340**, providing at least a portion of the airflow from the cooling duct to an aft cavity defined at least in part by an inner combustor casing of the gas turbine engine and a compressor section of the gas turbine engine.

[0075] In some embodiments, the first section may be the high-pressure compressor section **26**. In other words, delivering the first flow of cooling air **222** to the first section of the gas turbine engine **12** via the first cooling duct **220** includes delivering the first flow of cooling air **222** to the high-pressure compressor section **26**. More particularly, the aft cavity **72** of the high-pressure compressor section **26**.

[0076] In additional and/or alternative embodiments, the second section of the gas turbine engine **12** may be the high-pressure turbine section **30**. Delivering the second flow of cooling air **242** to a second section of the gas turbine engine **12** may refer to providing a portion of cooling air **242** to the high-pressure turbine section **30**. More particularly, the second cooling duct **240** may transport the second flow of cooling air **242** to the forward wheelspace cavity **104** of the high-pressure turbine section **30**.

[0077] Additionally, the method **300** may further include receiving data indicative of an operative condition of the gas turbine engine. Providing at least a portion of the airflow from the cooling duct to the aft cavity (e.g., at **330**), may further include providing at least a portion of the airflow from the cooling duct to the aft cavity in response to receiving data indicative of the operative condition of the gas turbine engine. The operative condition may include at least one of the following: a high operating temperature condition; a high-pressure condition; a supersonic cruise condition; and a

takeoff condition.
[0078] In some embodiments, receiving data indicative of the operative condition includes receiving data indicative of the gas turbine engine being in the operative condition. Further, providing at least the portion of the airflow from the cooling duct to the aft cavity in response to receiving data indicative of the operative condition of the gas turbine engine may include increasing the portion of the airflow provided to the aft cavity in response to receiving the data indicative of the gas turbine engine being in the operative condition.
[0079] Additionally, receiving data indicative of the operative condition may include receiving data indicative of the gas turbine engine no longer being in the operative condition. Providing at least the portion of the airflow from the cooling duct to the aft cavity in response to receiving data indicative of the operative condition of the gas turbine engine may include decreasing the portion of the airflow provided to the aft cavity in response to receiving the data indicative of the gas turbine engine no longer being in the operative condition.
[0080] Additionally, the method **300** may further include flowing compressor discharge air **210** from the high-pressure compressor **26** of the gas turbine engine **12**. Flowing the compressor discharge air **210** from the high-pressure compressor **26** of the gas turbine engine **12** may occur prior to providing the first flow of cooled air **222** to the first cooling duct **220** and/or prior to providing the second flow of cooled air **242** to the second cooling duct **240**. In some embodiments, flowing compressor discharge air **210** from the high-pressure compressor **26** of the gas turbine engine **12** may be triggered by the at least one sensor **245** sensing one or more operative conditions. As mentioned above, one or more operative conditions comprise at least one of the following: high operating temperature; high-pressure; supersonic cruise; and takeoff.
[0081] The method **300** may also include cooling the compressor discharge air **210** using a heat exchanger **140** prior to at least one of providing the first flow of cooled air **222** to the first cooling duct **220** and providing the second flow of cooled air **242** to the second cooling duct **240**. In at least some embodiments, cooling the compressor discharge air **210** using the heat exchanger **140** occurs after flowing the compressor discharge air **210** from the high-pressure compressor **26**. However, it will be appreciated in some embodiments, cooling the compressor discharge air **210** using the heat exchanger **140** may occur simultaneously as flowing the compressor discharge air **210** from the high-pressure compressor **26**, e.g., where the heat exchanger **140** cools the compressor discharge air **210** while at least some of the compressor discharge air downstream is still flowing out from the compressor section **23** into the heat exchanger **140**.
[0082] This written description uses examples to disclose the present disclosure, including the best mode, and also to enable any person skilled in the art to practice the disclosure, including making and using any devices or systems and performing any incorporated methods. The patentable scope of the disclosure is defined by the claims, and may include other examples that occur to those skilled in the art. Such other examples are intended to be within the scope of the claims if they include structural elements that do not differ from the literal language of the claims, or if they include equivalent structural elements with insubstantial differences from the literal languages of the claims.
[0083] Further aspects are provided by the subject matter of the following clauses:
[0084] A gas turbine engine comprising: a compressor section comprising a high-pressure compressor; a combustion section comprising an outer combustor casing and an inner combustor casing, the inner combustor casing defining in part an aft cavity with the compressor section and defining in part a diffuser cavity with the outer combustor casing; and a cooling system for cooling at least part of the gas turbine engine, the cooling system comprising: a compressor discharge pressure duct located outward of the outer combustor casing and positioned in fluid communication with the diffuser cavity, the compressor section, or both for receiving an airflow from the diffuser cavity, from the compressor section, or both; a heat exchanger in thermal communication with the compressor discharge pressure duct for reducing a temperature of the airflow; a cooling duct

located inward of the outer combustor casing and in fluid communication with the compressor discharge pressure duct and the aft cavity for receiving the airflow and providing at least a portion of the airflow to the aft cavity.
[0085] The gas turbine engine of one or more of the preceding clauses, wherein the cooling duct is a first cooling duct, and wherein the gas turbine engine further comprises: a turbine section, wherein the cooling system further comprises a second cooling duct in fluid communication with the compressor discharge pressure duct and further in fluid communication with the turbine section.
[0086] The gas turbine engine of one or more of the preceding clauses, wherein the high-pressure compressor comprises a material, wherein the material defines a material temperature limit in degrees Fahrenheit, and wherein cooling system is configured to provide the airflow to the diffuser cavity at a temperature in degrees Fahrenheit less than or equal to 85% of material temperature limit.
[0087] The gas turbine engine of one or more of the preceding clauses, wherein the turbine section comprises a high-pressure turbine, wherein the second cooling duct is in fluid communication with the high-pressure turbine of the turbine section.
[0088] The gas turbine engine of one or more of the preceding clauses, wherein the high-pressure turbine comprises an inlet guide vane and a first stage blade, wherein the inlet guide vane and the first stage blade define a forward wheelspace cavity, and wherein the gas turbine engine defines a purge air flowpath from the aft cavity to the forward wheelspace cavity.
[0089] The gas turbine engine of one or more of the preceding clauses, the cooling system further comprising: at least one sensor for sensing data indicative of an operative condition of the gas turbine engine; and at least one valve in fluid communication with the cooling duct, wherein the at least one valve is configured to operate based on the operative condition.
[0090] The gas turbine engine of one or more of the preceding clauses, wherein the operative condition comprises at least one of: a high operating temperature condition; a high-pressure condition; a supersonic cruise condition; a takeoff condition; or a climb condition.
[0091] The gas turbine engine of one or more of the preceding clauses, wherein the operative condition is the high operating temperature condition, and wherein the high operating temperature comprises an operating condition wherein a compressor exit temperature is higher than 1000 degrees Fahrenheit.
[0092] The gas turbine engine of one or more of the preceding clauses, wherein the cooling system further comprises a controller operably coupled to the at least one sensor for receiving the data indicative of the operative condition and the at least one valve for actuating the at least one valve, wherein the controller is configured to move the at least one valve to an open position when the gas turbine engine is in the operative condition and is further configured to move the at least one valve to a closed position when the gas turbine engine is not in the operative condition.
[0093] The gas turbine engine of one or more of the preceding clauses, wherein the cooling duct is a first cooling duct, wherein the cooling system further comprises a second cooling duct in fluid communication with the compressor discharge pressure duct and a manifold fluidly connected to at least one of the first cooling duct and the second cooling duct, the manifold extending a total of 360 degrees and comprising at least one segment.
[0094] The gas turbine engine of one or more of the preceding clauses, wherein the manifold comprises two segments, wherein each of the two segments extends 180 degrees.
[0095] The gas turbine engine of one or more of the preceding clauses, wherein the heat exchanger is positioned outside of the outer combustor casing.
[0096] The gas turbine engine of one or more of the preceding clauses, wherein the heat exchanger is operated by a supercritical fluid.
[0097] The gas turbine engine of one or more of the preceding clauses, wherein the compressor discharge pressure duct is in fluid communication with the diffuser cavity for receiving the airflow.
[0098] A method of cooling one or more sections in a gas turbine engine, the method comprising:

receiving an airflow from a diffuser cavity defined by an outer combustor casing of the gas turbine engine; cooling the airflow received from the diffuser cavity with a heat exchanger; providing the airflow from the heat exchanger to a cooling duct located inward of the outer combustor casing; and providing at least a portion of the airflow from the cooling duct to an aft cavity defined at least in part by an inner combustor casing of the gas turbine engine and a compressor section of the gas turbine engine.
[0099] The method of one or more of the preceding clauses, further comprising: receiving data indicative of an operative condition of the gas turbine engine, wherein providing at least a portion of the airflow from the cooling duct to the aft cavity comprises providing at least a portion of the airflow from the cooling duct to the aft cavity in response to receiving data indicative of the operative condition of the gas turbine engine.
[0100] The method of one or more of the preceding clauses, wherein the operative condition comprises at least one of: a high operating temperature condition; a high-pressure condition; a supersonic cruise condition; or a takeoff condition.
[0101] The method of one or more of the preceding clauses, wherein receiving data indicative of the operative condition comprises receiving data indicative of the gas turbine engine being in the operative condition, and wherein providing at least the portion of the airflow from the cooling duct to the aft cavity in response to receiving data indicative of the operative condition of the gas turbine engine comprises increasing the portion of the airflow provided to the aft cavity in response to receiving the data indicative of the gas turbine engine being in the operative condition.
[0102] The method of one or more of the preceding clauses, wherein receiving data indicative of the operative condition comprises receiving data indicative of the gas turbine engine no longer being in the operative condition, and wherein providing at least the portion of the airflow from the cooling duct to the aft cavity in response to receiving data indicative of the operative condition of the gas turbine engine comprises decreasing the portion of the airflow provided to the aft cavity in response to receiving the data indicative of the gas turbine engine no longer being in the operative condition.
[0103] The method of one or more of the preceding clauses, further comprising: providing at least a portion of the airflow from the cooling duct to a turbine section of the gas turbine engine.

Claims

1. A gas turbine engine comprising: a compressor section comprising a compressor; a combustion section comprising an outer combustor casing and an inner combustor casing, the inner combustor casing defining in part an aft cavity with the compressor section and defining in part a diffuser cavity with the outer combustor casing; and a cooling system for cooling at least part of the gas turbine engine, the cooling system comprising: a sensor for sensing data indicative of an operative condition of the gas turbine engine; a compressor discharge pressure duct positioned in fluid communication with the diffuser cavity, the compressor section, or both for receiving an airflow from the diffuser cavity, from the compressor section, or both; a heat exchanger in thermal communication with the compressor discharge pressure duct; a cooling duct located inward of the outer combustor casing and in fluid communication with the compressor discharge pressure duct and the aft cavity; and a valve in fluid communication with the cooling duct configured to operate based on the operative condition.
2. The gas turbine engine of claim 1, wherein the compressor comprises a material, wherein the material defines a material temperature limit in degrees Fahrenheit, and wherein the cooling system is configured to provide another portion of the airflow to the aft cavity at a temperature in degrees Fahrenheit less than or equal to 85% of the material temperature limit when the gas turbine engine is operated at a rated speed during standard day operating conditions.
3. The gas turbine engine of claim 1, wherein the operative condition comprises at least one of: a

high operating temperature condition; a high-pressure condition; a supersonic cruise condition; a takeoff condition; or a climb condition.

4. The gas turbine engine of claim 3, wherein the operative condition is the high operating temperature condition, and wherein the high operating temperature comprises an operating condition wherein a compressor exit temperature is higher than 1000 degrees Fahrenheit.

5. The gas turbine engine of claim 1, further comprising a controller, the controller comprising one or more computing devices in operable communication with the sensor and the valve, the one or more computing devices of the controller being configured to receive the data indicative of the operative condition from the sensor and control the valve to provide a portion of the airflow from the cooling duct to the aft cavity in response to receiving the data indicative of the operative condition of the gas turbine engine.

6. The gas turbine engine of claim 1, wherein the cooling system further comprises a controller, the controller comprising one or more computing devices operably coupled to the at least one sensor, for receiving the data indicative of the operative condition, and to the valve, for actuating the valve, wherein the one or more computing devices of the controller are configured to move the valve to an open position when the gas turbine engine is in the operative condition and are further configured to move the valve to a closed position when the gas turbine engine is not in the operative condition.

7. The gas turbine engine of claim 1, wherein the cooling duct includes a pipe and a manifold extending in a circumferential direction from the pipe about a longitudinal centerline of the gas turbine engine, the pipe and the manifold extending through the diffuser cavity downstream from the heat exchanger.

8. The gas turbine engine of claim 7, wherein the manifold is divided into two segments, wherein each of the two segments extends 180 degrees about the longitudinal centerline of the gas turbine engine.

9. The gas turbine engine of claim 1, wherein the heat exchanger is positioned outside of the outer combustor casing.

10. The gas turbine engine of claim 1, wherein the heat exchanger is fluidly coupled to the diffuser cavity via a compressor discharge pressure duct.

11. The gas turbine engine of claim 1, wherein the heat exchanger includes supercritical carbon dioxide as a cooling fluid.

12. The gas turbine engine of claim 1, wherein the cooling duct is a first cooling duct for receiving the airflow and providing a first portion of the airflow to the aft cavity, the first cooling duct including a manifold extending in a circumferential direction of the gas turbine engine, and wherein the gas turbine engine further comprises a second cooling duct located inward of the outer combustor casing and in fluid communication with the compressor discharge pressure duct for receiving the airflow and providing a second portion of the airflow.

13. The gas turbine engine of claim 12, wherein the valve is located downstream of the manifold and in fluid communication with at least one of the first cooling duct or the second cooling duct.

14. The gas turbine engine of claim 12, wherein the first cooling duct fluidly couples the heat exchanger to the aft cavity.

15. The gas turbine engine of claim 1, wherein the heat exchanger is fluidly coupled to the diffuser cavity via a first fluid opening.

16. The gas turbine engine of claim 1, wherein the compressor is in fluid communication with the diffuser cavity to provide a flow of compressed air from the compressor to the diffuser cavity.

17. The gas turbine engine of claim 1, further comprising a high-pressure shaft disposed radially inward from the inner combustor casing, wherein the high-pressure shaft and the inner combustor casing at least partially define a forward shaft outer cavity.

18. The gas turbine engine of claim 17, wherein a seal extends from the high-pressure shaft to the inner combustor casing to separate the aft cavity from the forward shaft outer cavity.

19. The gas turbine engine of claim 18, wherein the aft cavity is defined at least in part by the inner combustor casing, the seal, and the compressor.

20. The gas turbine engine of claim 1, further comprising a turbine section, wherein the cooling duct provides a portion of the airflow to the turbine section.