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GAS TURBINE ENGINE

Abstract

A gas turbine engine is provided. The gas turbine engine includes: a turbomachine having a compressor section, a combustion section, and a turbine section arranged in serial flow order, the compressor section having a high pressure compressor defining a high pressure compressor exit area (A.sub.HPCExit) in square inches; wherein the gas turbine engine defines a redline exhaust gas temperature (EGT) in degrees Celsius, a total sea level static thrust output (Fn.sub.Total) in pounds, and a corrected specific thrust, wherein the corrected specific thrust is greater than or equal to 42 and less than or equal to 90, the corrected specific determined as follows: Fn.sub.Total×EGT/(A.sub.HPCExit.sup.2×1000).

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Background/Summary

CROSS-REFERENCE TO RELATED APPLICATIONS [0001] This application is a continuation-in-part patent application of U.S. application Ser. No. 18/481,515 filed Oct. 5, 2023, which is a continuation-in-part application of U.S. application Ser. No. 17/978,629 filed Nov. 1, 2022. Each of these applications are hereby incorporated by reference in their entirety.

FIELD
[0002] The present disclosure relates to a gas turbine engine.
BACKGROUND
[0003] A gas turbine engine typically includes a fan and a turbomachine. The turbomachine generally includes an inlet, one or more compressors, a combustor, and at least one turbine. The compressors compress air which is channeled to the combustor where it is mixed with fuel. The mixture is then ignited for generating hot combustion gases. The combustion gases are channeled to the turbine(s) which extracts energy from the combustion gases for powering the compressor(s), as well as for producing useful work to propel an aircraft in flight. The turbomachine is mechanically coupled to the fan for driving the fan during operation.

Description

BRIEF DESCRIPTION OF THE DRAWINGS
[0004] 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 FIG.s, in which:
[0005] FIG. **1** is a schematic cross-sectional view of a three-stream engine in accordance with an exemplary embodiment of the present disclosure.
[0006] FIG. **2** is a close-up, schematic view of the exemplary three-stream engine of FIG. **1** with a cooled cooling air system in accordance with an exemplary embodiment of the present disclosure.
[0007] FIG. **3** is a close-up view of an aft-most stage of high pressure compressor rotor blades within the exemplary three-stream engine of FIG. **1**.
[0008] FIG. **4** is a close-up, schematic view of the exemplary three-stream engine of FIG. **1** showing the cooled cooling air system of FIG. **2**.
[0009] FIG. **5** is a schematic view of a thermal transport bus of the present disclosure.
[0010] FIG. **6** is a table depicting numerical values showing the relationships between various parameters in accordance with various example embodiments of the present disclosure.
[0011] FIG. **7** is a graph depicting a range of corrected specific thrust values and redline exhaust gas temperature values of gas turbine engines in accordance with various example embodiments of the present disclosure.
[0012] FIG. **8** is a schematic view of a ducted turbofan engine in accordance with an exemplary aspect of the present disclosure.
[0013] FIG. **9** is a schematic, close-up view of a gas turbine engine having a cooled cooling air system in accordance with another exemplary aspect of the present disclosure.
[0014] FIG. **10** is a schematic, close-up view of a gas turbine engine having a cooled cooling air

system in accordance with yet another exemplary aspect of the present disclosure.

[0015] FIG. **11** is a schematic, close-up view of a gas turbine engine having a cooled cooling air system in accordance with still another exemplary aspect of the present disclosure.

[0016] FIG. **12** is a schematic view of a turbofan engine in accordance with another exemplary aspect of the present disclosure.

[0017] FIG. **13** is a perspective view of a turbine disk, as viewed from the front or fan portion of the engine in the direction of gas flow, showing where the corrosion resistant coating can be located according to example embodiments of the present disclosure.

[0018] FIG. **14** is a cross-sectional representation of a corrosion resistant coating applied to a substrate component according to example embodiments of the present disclosure.

[0019] FIG. **15** is a flowchart of a method of coating an article according to example embodiments of the present disclosure.

[0020] FIG. **16** is a cross-sectional view of a portion of an exemplary turbine section of a gas turbine engine.

[0021] FIG. **17** is a cross-sectional view of an exemplary superalloy substrate having a surface coated with an exemplary coating according to embodiments.

[0022] FIG. **18** is a perspective view of an exemplary turbine disk, as viewed from the front or fan portion of the engine in the direction of gas flow, showing where a corrosion resistant coating may be desirably located.

DETAILED DESCRIPTION

[0023] 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.

[0024] The term “cooled cooling air system” is used herein to mean a system configured to provide a cooling airflow to one or more components exposed to a working gas flowpath of a turbomachine of a gas turbine engine at a location downstream of a combustor of the turbomachine and upstream of an exhaust nozzle of the turbomachine, the cooling airflow being in thermal communication with a heat exchanger for reducing a temperature of the cooling airflow at a location upstream of the one or more components.

[0025] The cooled cooling air systems contemplated by the present disclosure may include a thermal bus cooled cooling air system (see, e.g., FIGS. **4** and **5**) or a dedicated heat exchanger cooled cooling air system (i.e., a cooled cooling air system including a heat sink heat exchanger dedicated to the cooled cooling air system); a bypass heat exchanger cooled cooling air system having a heat sink heat exchanger thermally coupled to an airflow through a bypass passage (see, e.g., FIG. **9**); an air-to-air cooled cooling air system (i.e., a cooled cooling air system having a heat sink heat exchanger configured to transfer heat to an airflow; see, e.g., FIG. **9**); an oil-to-air cooled cooling air system (i.e., a cooled cooling air system having a heat sink heat exchanger configured to transfer heat to an oil flow); a fuel-to-air cooled cooling air system (i.e., a cooled cooling air system having a heat sink heat exchanger configured to transfer heat to a fuel flow, such as a Jet A fuel flow, a liquid hydrogen or hydrogen gas fuel flow, etc.; see, e.g., FIG. **4**); or a combination thereof.

[0026] In one or more of the exemplary cooled cooling air systems described herein, the cooled cooling air system may receive the cooling air from a downstream end of a high pressure compressor (i.e., a location closer to a last stage of the high pressure compressor), an upstream end of the high pressure compressor (i.e., a location closer to a first stage of the high pressure compressor), a downstream end of a low pressure compressor (i.e., a location closer to a last stage of the low pressure compressor), an upstream end of the low pressure compressor (i.e., a location closer to a first stage of the low pressure compressor), a location between compressors, a bypass passage, a combination thereof, or any other suitable airflow source.

[0027] 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.

[0028] 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.

[0029] 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.

[0030] 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.

[0031] The terms "coupled," "fixed," "attached to," and the like refer to both direct coupling, fixing, or attaching, as well as indirect coupling, fixing, or attaching through one or more intermediate components or features, unless otherwise specified herein.

[0032] The singular forms "a", "an", and "the" include plural references unless the context clearly dictates otherwise.

[0033] The phrases "from X to Y" and "between X and Y" each refers to a range of values inclusive of the endpoints (i.e., refers to a range of values that includes both X and Y).

[0034] A "third stream" as used herein means a non-primary air stream capable of increasing fluid energy to produce a minority of total propulsion system thrust. A pressure ratio of the third stream may be higher than that of the primary propulsion stream (e.g., a bypass or propeller driven propulsion stream). The thrust may be produced through a dedicated nozzle or through mixing of an airflow through the third stream with a primary propulsion stream or a core air stream, e.g., into a common nozzle.

[0035] As used herein, the term "substantially free" is understood to mean completely free of said constituent, or inclusive of trace amounts of same. "Trace amounts" are those quantitative levels of chemical constituent that are barely detectable and provide no benefit to the functional or aesthetic properties of the subject composition. As used herein, the term "chromate" refers to chromate ion, dichromate ion, and any other form of hexavalent chromium.

[0036] In certain exemplary embodiments an operating temperature of the airflow through the third stream may be less than a maximum compressor discharge temperature for the engine, and more specifically may be less than 350 degrees Fahrenheit (such as less than 300 degrees Fahrenheit, such as less than 250 degrees Fahrenheit, such as less than 200 degrees Fahrenheit, and at least as great as an ambient temperature). In certain exemplary embodiments these operating temperatures may facilitate heat transfer to or from the airflow through the third stream and a separate fluid stream. Further, in certain exemplary embodiments, the airflow through the third stream may contribute less than 50% of the total engine thrust (and at least, e.g., 2% of the total engine thrust) at a takeoff condition, or more particularly while operating at a rated takeoff power at sea level, static flight speed, 86 degrees Fahrenheit ambient temperature operating conditions.

[0037] Furthermore in certain exemplary embodiments, aspects of the airflow through the third stream (e.g., airstream, mixing, or exhaust properties), and thereby the aforementioned exemplary percent contribution to total thrust, may passively adjust during engine operation or be modified purposefully through use of engine control features (such as fuel flow, electric machine power, variable stators, variable inlet guide vanes, valves, variable exhaust geometry, or fluidic features) to adjust or optimize overall system performance across a broad range of potential operating conditions.

[0038] The term "takeoff power level" refers to a power level of a gas turbine engine used during a takeoff operating mode of the gas turbine engine during a standard day operating condition.
[0039] The term "standard day operating condition" refers to ambient conditions of sea level altitude, 59 degrees Fahrenheit, and 60 percent relative humidity.
[0040] The term "propulsive efficiency" refers to an efficiency with which the energy contained in an engine's fuel is converted into kinetic energy for the vehicle incorporating the engine, to accelerate it, or to replace losses due to aerodynamic drag or gravity.
[0041] The term redline exhaust gas temperature (referred to herein as "redline EGT") refers to a maximum permitted takeoff temperature documented in a Federal Aviation Administration ("FAA")-type certificate data sheet. For example, in certain exemplary embodiments, the term redline EGT may refer to a maximum permitted takeoff temperature of an airflow after a first stage stator downstream of an HP turbine of an engine that the engine is rated to withstand. For example, with reference to the exemplary engine **100** discussed below with reference to FIG. **2**, the term redline EGT refers to a maximum permitted takeoff temperature of an airflow after the first stator **208** downstream of the last stage of rotor blades **206** of the HP turbine **132** (at location **215** into the first of the plurality of LP turbine rotor blades **210**). In embodiments wherein the engine is configured as a three spool engine (as compared to the two spool engine of FIG. **2**; see FIG. **12**), the term redline EGT refers to a maximum permitted takeoff temperature of an airflow after the first stator downstream of the last stage of rotor blades of the intermediate speed turbine (see intermediate speed turbine **516** of the engine **500** of FIG. **12**). The term redline EGT is sometimes also referred to as an indicated turbine exhaust gas temperature or indicated turbine temperature.
[0042] Generally, a turbofan engine includes a fan and a turbomachine, with the turbomachine rotating the fan to generate thrust. The turbomachine includes a compressor section, a combustion section, a turbine section, and an exhaust section and defines a working gas flowpath therethrough. A relatively small amount of thrust may also be generated by an airflow exiting the working gas flowpath of the turbomachine through the exhaust section. In addition, certain turbofan engines may further include a third stream that contributes to a total thrust output of the turbofan engine, potentially allowing for a reduction in size of a core of the turbomachine for a given total turbofan engine thrust output.
[0043] Conventional turbofan engine design practice has limited a compressor pressure ratio based at least in part on the gas temperatures at the exit stage of a high pressure compressor. These relatively high temperatures at the exit of the high pressure compressor may also be avoided when they result in prohibitively high temperatures at an inlet to the turbine section, as well as when they result in prohibitively high exhaust gas temperatures through the exhaust section. For a desired turbofan engine thrust output produced from an increased pressure ratio across the high pressure compressor, there is an increase in the gas temperature at the compressor exit, at a combustor inlet, at the turbine section inlet, and through an exhaust section of the turbofan engine.
[0044] The inventors have recognized that there are generally three approaches to making a gas turbine engine capable of operating at higher temperatures while providing a net benefit to engine performance: reducing the temperature of a gas used to cool core components, utilizing materials capable of withstanding higher operating temperature conditions, or a combination thereof.
[0045] Referring to the case of an engine that utilizes cooled cooling air for operating at higher temperatures, the inventors of the present disclosure discovered, unexpectedly, that the costs associated with achieving a higher compression by reducing gas temperatures used to cool core components to accommodate higher core gas temperatures may indeed produce a net benefit, contrary to prior expectations in the art. The inventors discovered during the course of designing several engine architectures of varying thrust classes and mission requirements (including the engines illustrated and described in detail herein) a relationship exists among the exhaust gas passing through the exhaust section, the desired maximum thrust for the engine, and the size of the exit stage of the high pressure compressor, whereby including this technology produces a net

benefit. Previously it was thought that the cost for including a technology to reduce the temperature of gas intended for cooling compressor and turbine components was too prohibitive, as compared to the benefits of increasing the core temperatures.

[0046] For example, the inventors of the present disclosure found that a cooled cooling air system may be included while maintaining or even increasing the maximum turbofan engine thrust output, based on this discovery. The cooled cooling air system may receive an airflow from the compressor section, reduce a temperature of the airflow using a heat exchanger, and provide the cooled airflow to one or more components of the turbine section, such as a first stage of high pressure turbine rotor blades. In such a manner, a first stage of high pressure turbine rotor blades may be capable of withstanding increased temperatures by using the cooled cooling air, while providing a net benefit to the turbofan engine, i.e., while taking into consideration the costs associated with accommodations made for the system used to cool the cooling air.

[0047] The inventors reached this conclusion after evaluating potentially negative impacts to engine performance brought on by introduction of a cooled cooling air system. For example, a cooled cooling air system may generally include a duct extending through a diffusion cavity between a compressor exit and a combustor within the combustion section, such that increasing the cooling capacity may concomitantly increase a size of the duct and thus increase a drag or blockage of an airflow through the diffusion cavity, potentially creating problems related to, e.g., combustor aerodynamics. Similarly, a dedicated or shared heat exchanger of the cooled cooling air system may be positioned in a bypass passage of the turbofan engine, which may create an aerodynamic drag or may increase a size of the shared heat exchanger and increase aerodynamic drag. Size and weight increases associated with maintaining certain risk tolerances were also taken into consideration. For example, a cooled cooling air system must be accompanied with adequate safeguards in the event of a burst pipe condition, which safeguards result in further increases in the overall size, complexity, and weight of the system.

[0048] With a goal of arriving at an improved turbofan engine capable of operating at higher temperatures at the compressor exit and turbine inlet, the inventors have proceeded in the manner of designing turbofan engines having an overall pressure ratio, total thrust output, redline exhaust gas temperature, and the supporting technology characteristics; checking the propulsive efficiency and qualitative turbofan engine characteristics of the designed turbofan engine; redesigning the turbofan engine to have higher or lower compression ratios based on the impact on other aspects of the architecture, total thrust output, redline exhaust gas temperature, and supporting technology characteristics; rechecking the propulsive efficiency and qualitative turbofan engine characteristics of the redesigned turbofan engine; etc. during the design of several different types of turbofan engines, including the turbofan engines described below with reference to FIGS. **1** and **4** through **8** through **11**, which will now be discussed in greater detail.

[0049] Referring now to FIG. **1**, a schematic cross-sectional view of an engine **100** is provided according to an example embodiment of the present disclosure. Particularly, FIG. **1** provides a turbofan engine having a rotor assembly with a single stage of unducted rotor blades. In such a manner, the rotor assembly may be referred to herein as an “unducted fan,” or the entire engine **100** may be referred to as an “unducted turbofan engine.” In addition, the engine **100** of FIG. **1** includes a third stream extending from a location downstream of a ducted mid-fan to a bypass passage over the turbomachine, as will be explained in more detail below.

[0050] For reference, the engine **100** defines an axial direction A, a radial direction R, and a circumferential direction C. Moreover, the engine **100** defines an axial centerline or longitudinal axis **112** that extends along the axial direction A. In general, the axial direction A extends parallel to the longitudinal axis **112**, the radial direction R extends outward from and inward to the longitudinal axis **112** in a direction orthogonal to the axial direction A, and the circumferential direction extends three hundred sixty degrees (360°) around the longitudinal axis **112**. The engine **100** extends between a forward end **114** and an aft end **116**, e.g., along the axial direction A.

[0051] The engine **100** includes a turbomachine **120** and a rotor assembly, also referred to a fan section **150**, positioned upstream thereof. Generally, the turbomachine **120** includes, in serial flow order, a compressor section, a combustion section **130**, a turbine section, and an exhaust section. Particularly, as shown in FIG. **1**, the turbomachine **120** includes a core cowl **122** that defines an annular core inlet **124**. The core cowl **122** further encloses at least in part a low pressure system and a high pressure system. For example, the core cowl **122** depicted encloses and supports at least in part a booster or low pressure ("LP") compressor **126** for pressurizing the air that enters the turbomachine **120** through core inlet **124**. A high pressure ("HP"), multi-stage, axial-flow compressor **128** receives pressurized air from the LP compressor **126** and further increases the pressure of the air. The pressurized air stream flows downstream to a combustor of the combustion section **130** where fuel is injected into the pressurized air stream and ignited to raise the temperature and energy level of the pressurized air.

[0052] It will be appreciated that as used herein, the terms "high/low speed" and "high/low pressure" are used with respect to the high pressure/high speed system and low pressure/low speed system interchangeably. Further, it will be appreciated that the terms "high" and "low" are used in this same context to distinguish the two systems, and are not meant to imply any absolute speed and/or pressure values.

[0053] The high energy combustion products flow from the combustion section **130** downstream to a high pressure turbine **132**. The high pressure turbine **132** drives the high pressure compressor **128** through a high pressure shaft **136**. In this regard, the high pressure turbine **132** is drivingly coupled with the high pressure compressor **128**. As will be appreciated, the high pressure compressor **128**, the combustion section **130**, and the high pressure turbine **132** may collectively be referred to as the "core" of the engine **100**. The high energy combustion products then flow to a low pressure turbine **134**. The low pressure turbine **134** drives the low pressure compressor **126** and components of the fan section **150** through a low pressure shaft **138**. In this regard, the low pressure turbine **134** is drivingly coupled with the low pressure compressor **126** and components of the fan section **150**. The LP shaft **138** is coaxial with the HP shaft **136** in this example embodiment. After driving each of the turbines **132**, **134**, the combustion products exit the turbomachine **120** through a turbomachine exhaust nozzle **140**.

[0054] Accordingly, the turbomachine **120** defines a working gas flowpath or core duct **142** that extends between the core inlet **124** and the turbomachine exhaust nozzle **140**. The working gas flowpath **142** is an annular duct positioned generally inward of the core cowl **122** along the radial direction R. The working gas flowpath **142** (e.g., the working gas flowpath through the turbomachine **120**) may be referred to as a second stream.

[0055] The fan section **150** includes a fan **152**, which is the primary fan in this example embodiment. For the depicted embodiment of FIG. **1**, the fan **152** is an open rotor or unducted fan **152**. In such a manner, the engine **100** may be referred to as an open rotor engine.

[0056] As depicted, the fan **152** includes an array of fan blades **154** (only one shown in FIG. **1**). The fan blades **154** are rotatable, e.g., about the longitudinal axis **112**. As noted above, the fan **152** is drivingly coupled with the low pressure turbine **134** via the LP shaft **138**. For the embodiments shown in FIG. **1**, the fan **152** is coupled with the LP shaft **138** via a speed reduction gearbox **155**, e.g., in an indirect-drive or geared-drive configuration.

[0057] Moreover, the array of fan blades **154** can be arranged in equal spacing around the longitudinal axis **112**. Each fan blade **154** has a root and a tip and a span defined therebetween, and further defines a central blade axis **156**. For this embodiment, each fan blade **154** of the fan **152** is rotatable about its respective central blade axis **156**, e.g., in unison with one another. One or more actuators **158** are provided to facilitate such rotation and therefore may be used to change a pitch of the fan blades **154** about their respective central blades' axes **156**.

[0058] The fan section **150** further includes a fan guide vane array **160** that includes fan guide vanes **162** (only one shown in FIG. **1**) disposed around the longitudinal axis **112**. For this embodiment, the fan guide vanes **162** are not rotatable about the longitudinal axis **112**. Each fan guide vane **162** has a root and a tip and a span defined therebetween. The fan guide vanes **162** may be unshrouded as shown in FIG. **1** or, alternatively, may be shrouded, e.g., by an annular shroud spaced outward from the tips of the fan guide vanes **162** along the radial direction R or attached to the fan guide vanes **162**.

[0059] Each fan guide vane **162** defines a central blade axis **164**. For this embodiment, each fan guide vane **162** of the fan guide vane array **160** is rotatable about its respective central blade axis **164**, e.g., in unison with one another. One or more actuators **166** are provided to facilitate such rotation and therefore may be used to change a pitch of the fan guide vane **162** about its respective central blade axis **164**. However, in other embodiments, each fan guide vane **162** may be fixed or unable to be pitched about its central blade axis **164**. The fan guide vanes **162** are mounted to a fan cowl **170**. Notably, the engine **100** defines a bypass passage **194** over the fan cowl **170** and core cowl **122**.

[0060] As shown in FIG. **1**, in addition to the fan **152**, which is unducted, a ducted fan **184** is included aft of the fan **152**, such that the engine **100** includes both a ducted and an unducted fan which both serve to generate thrust through the movement of air without passage through at least a portion of the turbomachine **120** (e.g., without passage through the HP compressor **128** and combustion section for the embodiment depicted). The ducted fan **184** is rotatable about the same axis (e.g., the longitudinal axis **112**) as the fan **152**. The ducted fan **184** is, for the embodiment depicted, driven by the low pressure turbine **134** (e.g. coupled to the LP shaft **138**). In the embodiment depicted, as noted above, the fan **152** may be referred to as the primary fan, and the ducted fan **184** may be referred to as a secondary fan. It will be appreciated that these terms “primary” and “secondary” are terms of convenience, and do not imply any particular importance, power, or the like.

[0061] The ducted fan **184** includes a plurality of fan blades (not separately labeled in FIG. **1**) arranged in a single stage, such that the ducted fan **184** may be referred to as a single stage fan. The fan blades of the ducted fan **184** can be arranged in equal spacing around the longitudinal axis **112**. Each blade of the ducted fan **184** has a root and a tip and a span defined therebetween.

[0062] The fan cowl **170** annularly encases at least a portion of the core cowl **122** and is generally positioned outward of at least a portion of the core cowl **122** along the radial direction R. Particularly, a downstream section of the fan cowl **170** extends over a forward portion of the core cowl **122** to define a fan duct flowpath, or simply a fan duct **172**. According to this embodiment, the fan duct flowpath or fan duct **172** may be understood as forming at least a portion of the third stream of the engine **100**.

[0063] Incoming air may enter through the fan duct **172** through a fan duct inlet **176** and may exit through a fan exhaust nozzle **178** to produce propulsive thrust. The fan duct **172** is an annular duct positioned generally outward of the working gas flowpath **142** along the radial direction R. The fan cowl **170** and the core cowl **122** are connected together and supported by a plurality of substantially radially-extending, circumferentially-spaced stationary struts **174** (only one shown in FIG. **1**). The stationary struts **174** may each be aerodynamically contoured to direct air flowing thereby. Other struts in addition to the stationary struts **174** may be used to connect and support the fan cowl **170** and/or core cowl **122**. In many embodiments, the fan duct **172** and the working gas flowpath **142** may at least partially co-extend (generally axially) on opposite sides (e.g., opposite radial sides) of the core cowl **122**. For example, the fan duct **172** and the working gas flowpath **142** may each extend directly from a leading edge **144** of the core cowl **122** and may partially co-extend generally axially on opposite radial sides of the core cowl **122**.

[0064] The engine **100** also defines or includes an inlet duct **180**. The inlet duct **180** extends between an engine inlet **182** and the core inlet **124**/fan duct inlet **176**. The engine inlet **182** is defined generally at the forward end of the fan cowl **170** and is positioned between the fan **152** and the fan guide vane array **160** along the axial direction A. The inlet duct **180** is an annular duct that is positioned inward of the fan cowl **170** along the radial direction R. Air flowing downstream along the inlet duct **180** is split, not necessarily evenly, into the working gas flowpath **142** and the fan duct **172** by the leading edge **144** of the core cowl **122**. The inlet duct **180** is wider than the working gas flowpath **142** along the radial direction R. The inlet duct **180** is also wider than the fan duct **172** along the radial direction R. The secondary fan **184** is positioned at least partially in the inlet duct **180**.

[0065] Notably, for the embodiment depicted, the engine **100** includes one or more features to increase an efficiency of a third stream thrust, Fn.sub.3S (e.g., a thrust generated by an airflow through the fan duct **172** exiting through the fan exhaust nozzle **178**, generated at least in part by the ducted fan **184**). In particular, the engine **100** further includes an array of inlet guide vanes **186** positioned in the inlet duct **180** upstream of the ducted fan **184** and downstream of the engine inlet **182**. The array of inlet guide vanes **186** are arranged around the longitudinal axis **112**. For this embodiment, the inlet guide vanes **186** are not rotatable about the longitudinal axis **112**. Each inlet guide vane **186** defines a central blade axis (not labeled for clarity), and is rotatable about its respective central blade axis, e.g., in unison with one another. In such a manner, the inlet guide vanes **186** may be considered a variable geometry component. One or more actuators **188** are provided to facilitate such rotation and therefore may be used to change a pitch of the inlet guide vanes **186** about their respective central blade axes. However, in other embodiments, each inlet guide vane **186** may be fixed or unable to be pitched about its central blade axis.

[0066] Further, located downstream of the ducted fan **184** and upstream of the fan duct inlet **176**, the engine **100** includes an array of outlet guide vanes **190**. As with the array of inlet guide vanes **186**, the array of outlet guide vanes **190** are not rotatable about the longitudinal axis **112**. However, for the embodiment depicted, unlike the array of inlet guide vanes **186**, the array of outlet guide vanes **190** are configured as fixed-pitch outlet guide vanes.

[0067] Further, it will be appreciated that for the embodiment depicted, the fan exhaust nozzle **178** of the fan duct **172** is further configured as a variable geometry exhaust nozzle. In such a manner, the engine **100** includes one or more actuators **192** for modulating the variable geometry exhaust nozzle. For example, the variable geometry exhaust nozzle may be configured to vary a total cross-sectional area (e.g., an area of the nozzle in a plane perpendicular to the longitudinal axis **112**) to modulate an amount of thrust generated based on one or more engine operating conditions (e.g., temperature, pressure, mass flowrate, etc. of an airflow through the fan duct **172**). A fixed geometry exhaust nozzle may also be adopted.

[0068] The combination of the array of inlet guide vanes **186** located upstream of the ducted fan **184**, the array of outlet guide vanes **190** located downstream of the ducted fan **184**, and the fan exhaust nozzle **178** may result in a more efficient generation of third stream thrust, Fn.sub.3S, during one or more engine operating conditions. Further, by introducing a variability in the geometry of the inlet guide vanes **186** and the fan exhaust nozzle **178**, the engine **100** may be capable of generating more efficient third stream thrust, Fn.sub.3S, across a relatively wide array of engine operating conditions, including takeoff and climb as well as cruise.

[0069] Moreover, referring still to FIG. **1**, in exemplary embodiments, air passing through the fan duct **172** may be relatively cooler (e.g., lower temperature) than one or more fluids utilized in the turbomachine **120**. In this way, one or more heat exchangers **196** may be positioned in thermal communication with the fan duct **172**. For example, one or more heat exchangers **196** may be disposed within the fan duct **172** and utilized to cool one or more fluids from the core engine with the air passing through the fan duct **172**, as a resource for removing heat from a fluid, e.g., compressor bleed air, oil or fuel.

[0070] Although not depicted, the heat exchanger **196** may be an annular heat exchanger extending substantially 360 degrees in the fan duct **172** (e.g., at least 300 degrees, such as at least 330 degrees). In such a manner, the heat exchanger **196** may effectively utilize the air passing through the fan duct **172** to cool one or more systems of the engine **100** (e.g., a cooled cooling air system (described below), lubrication oil systems, compressor bleed air, electrical components, etc.). The heat exchanger **196** uses the air passing through duct **172** as a heat sink and correspondingly increases the temperature of the air downstream of the heat exchanger **196** and exiting the fan exhaust nozzle **178**.

[0071] As will be appreciated, the engine **100** defines a total sea level static thrust output Fn.sub.Total, corrected to standard day conditions, which is generally equal to a maximum total engine thrust. It will be appreciated that “sea level static thrust corrected to standard day conditions” refers to an amount of thrust an engine is capable of producing while at rest relative to the earth and the surrounding air during standard day operating conditions.

[0072] The total sea level static thrust output Fn.sub.Total may generally be equal to a sum of: a fan stream thrust Fn.sub.Fan (i.e., an amount of thrust generated by the fan **152** through the bypass passage **194**), the third stream thrust Fn.sub.3S (i.e., an amount of thrust generated through the fan duct **172**), and a turbomachine thrust Fn.sub.TM (i.e., an amount of thrust generated by an airflow through the turbomachine exhaust nozzle **140**), each during the static, sea level, standard day conditions. The engine **100** may define a total sea level static thrust output Fn.sub.Total greater than or equal to 15,000 pounds. For example, it will be appreciated that the engine **100** may be configured to generate at least 25,000 pounds and less than 80,000 pounds, such as between 25,000 and 50,000 pounds, such as between 35,000 and 45,000 pounds of thrust during a takeoff operating power, corrected to standard day sea level conditions.

[0073] As will be appreciated, the engine **100** defines a redline exhaust gas temperature (referred to herein as “EGT”), which is defined above, and for the embodiment of FIG. **1** refers to a maximum permitted takeoff temperature of an airflow after the first stator **208** downstream of the last stage of rotor blades **206** of the HP turbine **132** (at location **215** into the first of the plurality of LP turbine rotor blades **210**; see FIG. **2**).

[0074] Referring now to FIG. **2**, a close-up, simplified, schematic view of a portion of the engine **100** of FIG. **1** is provided. The engine **100**, as noted above includes the turbomachine **120** having the LP compressor **126**, the HP compressor **128**, the combustion section **130**, the HP turbine **132**, and the LP turbine **134**. The LP compressor **126** includes a plurality of stages of LP compressor rotor blades **198** and a plurality of stages of LP compressor stator vanes **200** alternatingly spaced with the plurality of stages of LP compressor rotor blades **198**. Similarly, the HP compressor **128** includes a plurality of stages of HP compressor rotor blades **202** and a plurality of stages of HP compressor stator vanes **204** alternatingly spaced with the plurality of stages of HP compressor rotor blades **202**. Moreover, within the turbine section, the HP turbine **132** includes at least one stage of HP turbine rotor blades **206** and at least one stage of HP turbine stator vanes **208**, and the LP turbine **134** includes a plurality of stages of LP turbine rotor blades **210** and a plurality of stages of LP turbine stator vanes **212** alternatingly spaced with the plurality of stages of LP turbine rotor blades **210**. With reference to the HP turbine **132**, the HP turbine **132** includes at least a first stage **214** of HP turbine rotor blades **206**.

[0075] Referring particularly to the HP compressor **128**, the plurality of stages of HP compressor rotor blades **202** includes an aftmost stage **216** of HP compressor rotor blades **202**. Referring briefly to FIG. **3**, a close-up view of an HP compressor rotor blade **202** in the aftmost stage **216** of HP compressor rotor blades **202** is provided. As will be appreciated, the HP compressor rotor blade **202** includes a trailing edge **218** and the aftmost stage **216** of HP compressor rotor blades **202** includes a rotor **220** having a base **222** to which the HP compressor rotor blade **202** is coupled. The base **222** includes a flowpath surface **224** defining in part the working gas flow path **142** through the HP compressor **128**. Moreover, the HP compressor **128** includes a shroud or liner **226** located outward of the HP compressor rotor blade **202** along the radial direction R. The shroud or liner **226** also includes a flowpath surface **228** defining in part the working gas flow path **142** through the HP compressor **128**.

[0076] The engine **100** (FIG. **3**) defines a reference plane **230** intersecting with an aft-most point of the trailing edge **218** of the HP compressor rotor blade **202** depicted, the reference plane **230** being orthogonal to the axial direction A. Further, the HP compressor **128** defines a high pressure compressor exit area (A.sub.HPCExit) within the reference plane **230**. More specifically, the HP compressor **128** defines an inner radius (R.sub.INNER) extending along the radial direction R within the reference plane **230** from the longitudinal axis **112** to the flowpath surface **224** of the base **222** of the rotor **220** of the aftmost stage **216** of HP compressor rotor blades **202**, as well as an outer radius (ROUTER) extending along the radial direction R within the reference plane **230** from the longitudinal axis **112** to the flowpath surface **228** of the shroud or liner **226**. The HP compressor **128** exit area is defined according to Expression (1):

[00001] $A_{HPCExit} = (R_{OUTER^2} - R_{INNER^2})$. Expression(1)

[0077] The inventors of the present disclosure have found that for a given total thrust output (Fn.sub.Total), a decrease in size of the high pressure compressor exit area (A.sub.HPCExit) may generally relate in an increase in a compressor exit temperature (i.e., a temperature of the airflow through the working gas flowpath **142** at the reference plane **230**), a turbine inlet temperature (i.e., a temperature of the airflow through the working gas flowpath **142** provided to the first stage **214** of HP turbine rotor blades **206**; see FIG. **2**), and the redline exhaust gas temperature (EGT). In particular, the inventors of the present disclosure have found that the high pressure compressor exit area (A.sub.HPCExit) may generally be used as an indicator of the above temperatures to be achieved by the engine **100** during operation for a given total thrust output (Fn.sub.Total) of the engine **100**.

[0078] Referring back to FIG. **2**, the exemplary engine **100** depicted includes one or more technologies to accommodate the relatively small high pressure compressor exit area (A.sub.HPCExit) for the total thrust output (Fn.sub.Total) of the engine **100**. In particular, for the embodiment depicted, the exemplary engine **100** includes a cooled cooling air system **250**. The exemplary cooled cooling air system **250** is in fluid communication with the HP compressor **128** and the first stage **214** of HP turbine rotor blades **206**. More specifically, for the embodiment depicted, the cooled cooling air system **250** includes a duct assembly **252** and a cooled cooling air (CCA) heat exchanger **254**. The duct assembly **252** is in fluid communication with the HP compressor **128** for receiving an airflow from the HP compressor **128** and providing such airflow to the first stage **214** of HP turbine rotor blades **206** during operation of the engine **100**. The CCA heat exchanger **254** is in thermal communication with the airflow through the duct assembly **252** for reducing a temperature of the airflow through the duct assembly **252** upstream of the first stage **214** of HP turbine rotor blades **206**.

[0079] Briefly, as will be explained in more detail below, the engine **100** depicted further includes a thermal transport bus **300**, with the CCA heat exchanger **254** of the cooled cooling air system **250** in thermal communication with, or integrated into, the thermal transport bus **300**. For the embodiment depicted, the engine **100** further includes the heat exchanger **196** in the fan duct **172** in thermal communication with, or integrated into, the thermal transport bus **300**, such that heat from the CCA heat exchanger **254** of the cooled cooling air system **250** may be transferred to the heat exchanger **196** in the fan duct **172** using the thermal transport bus **300**.

[0080] Referring now to FIG. **4**, a close-up, schematic view of the turbomachine **120** of the engine **100** of FIG. **2**, including the cooled cooling air system **250**, is provided.

[0081] As is shown, the turbine section includes a compressor casing **256**, and the combustion section **130** of the turbomachine **120** generally includes an outer combustor casing **258**, an inner combustor casing **260**, and a combustor **262**. The combustor **262** generally includes an outer combustion chamber liner **264** and an inner combustion chamber liner **266**, together defining at least in part a combustion chamber **268**. The combustor **262** further includes a fuel nozzle **270** configured to provide a mixture of fuel and air to the combustion chamber **268** to generate combustion gases.

[0082] The engine **100** further includes a fuel delivery system **272** including at least a fuel line **274** in fluid communication with the fuel nozzle **270** for providing fuel to the fuel nozzle **270**.
[0083] The turbomachine **120** includes a diffuser nozzle **276** located downstream of the aftmost stage **216** of HP compressor rotor blades **202** of the HP compressor **128**, within the working gas flowpath **142**. In the embodiment depicted, the diffuser nozzle **276** is coupled to, or integrated with the inner combustor casing **260**, the outer combustor casing **258**, or both. The diffuser nozzle **276** is configured to receive compressed airflow from the HP compressor **128** and straighten such compressed air prior to such compressed air being provided to the combustion section **130**. The combustion section **130** defines a diffusion cavity **278** downstream of the diffuser nozzle **276** and upstream of the combustion chamber **268**.
[0084] As noted above, the exemplary engine **100** further includes the cooled cooling air system **250**. The cooled cooling air system **250** includes the duct assembly **252** and the CCA heat exchanger **254**. More specifically, the duct assembly **252** includes a first duct **280** in fluid communication with the HP compressor **128** and the CCA heat exchanger **254**. The first duct **280** more specifically extends from the HP compressor **128**, through the compressor casing **256**, to the CCA heat exchanger **254**. For the embodiment depicted, the first duct **280** is in fluid communication with the HP compressor **128** at a location in between the last two stages of HP compressor rotor blades **202**. In such a manner, the first duct **280** is configured to receive a cooling airflow from the HP compressor **128** and to provide the cooling airflow to the CCA heat exchanger **254**.
[0085] It will be appreciated, however, that in other embodiments, the first duct **280** may additionally or alternatively be in fluid communication with the HP compressor **128** at any other suitable location, such as at any other location closer to a downstream end of the HP compressor **128** than an upstream end of the HP compressor **128**, or alternatively at a location closer to the upstream end of the HP compressor **128** than the downstream end of the HP compressor **128**.
[0086] The duct assembly **252** further includes a second duct **282** extending from the CCA heat exchanger **254** to the outer combustor casing **258** and a third duct **284** extending from the outer combustor casing **258** inwardly generally along the radial direction R. The CCA heat exchanger **254** may be configured to receive the cooling airflow and to extract heat from the cooling airflow to reduce a temperature of the cooling airflow. The second duct **282** may be configured to receive cooling airflow from the CCA heat exchanger **254** and provide the cooling airflow to the third duct **284**. The third duct **284** extends through the diffusion cavity generally along the radial direction R.
[0087] Moreover, for the embodiment depicted, the duct assembly **252** further includes a manifold **286** in fluid communication with the third duct **284** and a fourth duct **288**. The manifold **286** extends generally along the circumferential direction C of the engine **100**, and the fourth duct **288** is more specifically a plurality of fourth ducts **288** extending from the manifold **286** at various locations along the circumferential direction C forward generally along the axial direction A towards the turbine section. In such a manner, the duct assembly **252** of the cooled cooling air system **250** may be configured to provide cooling airflow to the turbine section at a variety of locations along the circumferential direction C.
[0088] Notably, referring still to FIG. **4**, the combustion section **130** includes an inner stator assembly **290** located at a downstream end of the inner combustion chamber liner **266**, and coupled to the inner combustor casing **260**. The inner stator assembly **290** includes a nozzle **292**. The fourth duct **288**, or rather, the plurality of fourth ducts **288**, are configured to provide the cooling airflow to the nozzle **292**. The nozzle **292** may include a plurality of vanes spaced along the circumferential direction C configured to impart a circumferential swirl to the cooling airflow provided through the plurality of fourth ducts **288** to assist with such airflow being provided to the first stage **214** of HP turbine rotor blades **206**.
[0089] In particular, for the embodiment depicted, the HP turbine **132** further includes a first stage HP turbine rotor **294**, with the plurality of HP turbine rotor blades **206** of the first stage **214**

coupled to the first stage HP turbine rotor **294**. The first stage HP turbine rotor **294** defines an internal cavity **296** configured to receive the cooling airflow from the nozzle **292** and provide the cooling airflow to the plurality of HP turbine rotor blades **206** of the first stage **214**. In such a manner, the cooled cooling air system **250** may provide cooling airflow to the HP turbine rotor blades **206** to reduce a temperature of the plurality HP turbine rotor blades **206** at the first stage **214** during operation of the engine **100**.

[0090] For example, in certain exemplary aspects, the cooled cooling air system **250** may be configured to provide a temperature reduction of the cooling airflow equal to at least 15% of the EGT and up to 45% of the EGT. Further, in certain exemplary aspects, the cooled cooling air system **250** may be configured to receive between 2.5% and 35% of an airflow through the working gas flowpath **142** at an inlet to the HP compressor **128**, such as between 3% and 20%, such as between 4% and 15%.

[0091] In addition, as briefly mentioned above, the cooled cooling air system **250** may utilize the thermal transport bus **300** to reject heat from the cooling air extracted from the compressor section of the turbomachine **120**. In particular, for the embodiment shown the CCA heat exchanger **254** is in thermal communication with or integrated into the thermal transport bus **300**. Notably, the thermal transport bus **300** further includes a fuel heat exchanger **302** in thermal communication with the fuel line **274**. In such a manner, the thermal transport bus **300** may extract heat from the cooling air extracted from the compressor section through the cooled cooling air system **250** and provide such heat to a fuel flow through the fuel line **274** upstream of the fuel nozzle **270**.

[0092] For the embodiment depicted, the thermal transport bus **300** includes a conduit having a flow of thermal transport fluid therethrough. More specifically, referring now briefly to FIG. **5**, a schematic view of a thermal transport bus **300** as may be utilized with the exemplary engine **100** described above with reference to FIGS. **1** through **4** is provided.

[0093] The thermal transport bus **300** includes an intermediary heat exchange fluid flowing therethrough and is formed of one or more suitable fluid conduits **304**. The heat exchange fluid may be an incompressible fluid having a high temperature operating range. Additionally, or alternatively, the heat exchange fluid may be a single phase fluid, or alternatively, may be a phase change fluid. In certain exemplary embodiments, the heat exchange fluid may be a supercritical fluid, such as a supercritical CO.sub.2.

[0094] The exemplary thermal transport bus **300** includes a pump **306** in fluid communication with the heat exchange fluid in the thermal transport bus **300** for generating a flow of the heat exchange fluid in/through the thermal transport bus **300**.

[0095] Moreover, the exemplary thermal transport bus **300** includes one or more heat source exchangers **308** in thermal communication with the heat exchange fluid in the thermal transport bus **300**. Specifically, the thermal transport bus **300** depicted includes a plurality of heat source exchangers **308**. The plurality of heat source exchangers **308** are configured to transfer heat from one or more of the accessory systems of an engine within which the thermal transport bus **300** is installed (e.g., engine **100** of FIGS. **1** through **4**) to the heat exchange fluid in the thermal transport bus **300**. For example, in certain exemplary embodiments, the plurality of heat source exchangers **308** may include one or more of: a CCA heat source exchanger (such as CCA heat exchanger **254** in FIGS. **2** and **4**); a main lubrication system heat source exchanger for transferring heat from a main lubrication system; an advanced clearance control (ACC) system heat source exchanger for transferring heat from an ACC system; a generator lubrication system heat source exchanger for transferring heat from the generator lubrication system; an environmental control system (ECS) heat exchanger for transferring heat from an ECS; an electronics cooling system heat exchanger for transferring heat from the electronics cooling system; a vapor compression system heat source exchanger; an air cycle system heat source exchanger; and an auxiliary system(s) heat source exchanger.

[0096] For the embodiment depicted, there are three heat source exchangers **308**. The heat source exchangers **308** are each arranged in series flow along the thermal transport bus **300**. However, in other exemplary embodiments, any other suitable number of heat source exchangers **308** may be included and one or more of the heat source exchangers **308** may be arranged in parallel flow along the thermal transport bus **300** (in addition to, or in the alternative to the serial flow arrangement depicted). For example, in other embodiments there may be a single heat source exchanger **308** in thermal communication with the heat exchange fluid in the thermal transport bus **300**, or alternatively, there may be at least two heat source exchangers **308**, at least four heat source exchangers **308**, at least five heat source exchangers **308**, or at least six heat source exchangers **308**, and up to twenty heat source exchangers **308** in thermal communication with heat exchange fluid in the thermal transport bus **300**.

[0097] Additionally, the exemplary thermal transport bus **300** of FIG. **5** further includes one or more heat sink exchangers **310** permanently or selectively in thermal communication with the heat exchange fluid in the thermal transport bus **300**. The one or more heat sink exchangers **310** are located downstream of the plurality of heat source exchangers **308** and are configured for transferring heat from the heat exchange fluid in the thermal transport bus **300**, e.g., to atmosphere, to fuel, to a fan stream, etc. For example, in certain embodiments the one or more heat sink exchangers **310** may include at least one of a RAM heat sink exchanger, a fuel heat sink exchanger, a fan stream heat sink exchanger, a bleed air heat sink exchanger, an engine intercooler heat sink exchanger, a bypass passage heat sink exchanger, or a cold air output heat sink exchanger of an air cycle system. The fuel heat sink exchanger is a “fluid to heat exchange fluid” heat exchanger wherein heat from the heat exchange fluid is transferred to a stream of liquid fuel (see, e.g., fuel heat exchanger **302** of the engine **100** of FIG. **4**). Moreover, the fan stream heat sink exchanger is generally an “air to heat exchange fluid” heat exchanger which transfers heat from the heat exchange fluid to an airflow through the fan stream (see, e.g., heat exchanger **196** of FIGS. **1** and **2**). Further, the bleed air heat sink exchanger is generally an “air to heat exchange fluid” heat exchanger which flows, e.g., bleed air from the LP compressor **126** over the heat exchange fluid to remove heat from the heat exchange fluid.

[0098] For the embodiment of FIG. **5**, the one or more heat sink exchangers **310** of the thermal transport bus **300** depicted includes a plurality of individual heat sink exchangers **310**. More particularly, for the embodiment of FIG. **5**, the one or more heat sink exchangers **310** include three heat sink exchangers **310** arranged in series. The three heat sink exchangers **310** are configured as a bypass passage heat sink exchanger, a fuel heat sink exchanger, and a fan stream heat sink exchanger. However, in other exemplary embodiments, the one or more heat sink exchangers **310** may include any other suitable number and/or type of heat sink exchangers **310**. For example, in other exemplary embodiments, a single heat sink exchanger **310** may be provided, at least two heat sink exchangers **310** may be provided, at least four heat sink exchangers **310** may be provided, at least five heat sink exchangers **310** may be provided, or up to twenty heat sink exchangers **310** may be provided. Additionally, in still other exemplary embodiments, two or more of the one or more heat sink exchangers **310** may alternatively be arranged in parallel flow with one another.

[0099] Referring still to the exemplary embodiment depicted in FIG. **5**, one or more of the plurality of heat sink exchangers **310** and one or more of the plurality of heat source exchangers **308** are selectively in thermal communication with the heat exchange fluid in the thermal transport bus **300**. More particularly, the thermal transport bus **300** depicted includes a plurality of bypass lines **312** for selectively bypassing each heat source exchanger **308** and each heat sink exchanger **310** in the plurality of heat sink exchangers **310**. Each bypass line **312** extends between an upstream juncture **314** and a downstream juncture **316**—the upstream juncture **314** located just upstream of a respective heat source exchanger **308** or heat sink exchanger **310**, and the downstream juncture **316** located just downstream of the respective heat source exchanger **308** or heat sink exchanger **310**.

[0100] Additionally, each bypass line **312** meets at the respective upstream juncture **314** with the thermal transport bus **300** via a three-way valve **318**. The three-way valves **318** each include an inlet fluidly connected with the thermal transport bus **300**, a first outlet fluidly connected with the thermal transport bus **300**, and a second outlet fluidly connected with the bypass line **312**. The three-way valves **318** may each be a variable throughput three-way valve, such that the three-way valves **318** may vary a throughput from the inlet to the first and/or second outlets. For example, the three-way valves **318** may be configured for providing anywhere between zero percent (0%) and one hundred percent (100%) of the heat exchange fluid from the inlet to the first outlet, and similarly, the three-way valves **318** may be configured for providing anywhere between zero percent (0%) and one hundred percent (100%) of the heat exchange fluid from the inlet to the second outlet.

[0101] Notably, the three-way valves **318** may be in operable communication with a controller of an engine including the thermal transport bus **300** (e.g., engine **100** of FIGS. **1** through **4**).

[0102] Further, each bypass line **312** also meets at the respective downstream juncture **316** with the thermal transport bus **300**. Between each heat source exchanger **308** or heat sink exchanger **310** and downstream juncture **316**, the thermal transport bus **300** includes a check valve **320** for ensuring a proper flow direction of the heat exchange fluid. More particularly, the check valve **320** prevents a flow of heat exchange fluid from the downstream juncture **316** towards the respective heat source exchanger **308** or heat sink exchanger **310**.

[0103] As alluded to earlier, the inventors discovered, unexpectedly during the course of gas turbine engine design—i.e., designing gas turbine engines having a variety of different high pressure compressor exit areas, total thrust outputs, redline exhaust gas temperatures, and supporting technology characteristics and evaluating an overall engine performance and other qualitative turbofan engine characteristics—a significant relationship between a total sea level static thrust output, a compressor exit area, and a redline exhaust gas temperature that enables increased engine core operating temperatures and overall engine propulsive efficiency. The relationship can be thought of as an indicator of the ability of a turbofan engine to have a reduced weight or volume as represented by a high pressure compressor exit area, while maintaining or even improving upon an overall thrust output, and without overly detrimentally affecting overall engine performance and other qualitative turbofan engine characteristics. The relationship applies to an engine that incorporates a cooled cooling air system, builds portions of the core using material capable of operating at higher temperatures, or a combination of the two. Significantly, the relationship ties the core size (as represented by the exit area of the higher pressure compressor) to the desired thrust and exhaust gas temperature associated with the desired propulsive efficiency and practical limitations of the engine design, as described below.

[0104] Referring to the case of an engine that utilizes cooled cooling air for operating at higher temperatures, the inventors discovered, unexpectedly, that the costs associated with achieving a higher compression, enabled by reducing gas temperatures used to cool core components to accommodate higher core gas temperatures, may indeed produce a net benefit, contrary to expectations in the art. Referring to the case of utilizing more temperature-resistant material, such as a Carbon Matrix Composite (CMC), it was found that certain aspects of the engine size, weight and operating characteristics can be positively affected while taking into account the complexities and/or drawbacks associated with such material. In either case, the relationship now described can apply to identify the interrelated operating conditions and core size—i.e., total sea level static thrust, redline exhaust gas temperature, and compressor exit area, respectively.

[0105] The inventors of the present disclosure discovered bounding the relationship between a product of total thrust output and redline exhaust gas temperature at a takeoff power level and the high pressure compressor exit area squared (corrected specific thrust) can result in a higher power density core. This bounded relationship, as described herein, takes into due account the amount of overall complexity and cost, and/or a low amount of reliability associated with implementing the technologies required to achieve the operating temperatures and exhaust gas temperature associated with the desired thrust levels. The amount of overall complexity and cost may be prohibitively high for gas turbine engines outside the bounds of the relationship as described herein, and/or the reliability may prohibitively low outside the bounds of the relationship as described herein. The relationship discovered, infra, can therefore identify an improved engine configuration suited for a particular mission requirement, one that takes into account efficiency, weight, cost, complexity, reliability, and other factors influencing the optimal choice for an engine configuration.

[0106] In addition to yielding an improved gas turbine engine, as explained in detail above, utilizing this relationship, the inventors found that the number of suitable or feasible gas turbine engine designs capable of meeting the above design requirements could be greatly diminished, thereby facilitating a more rapid down selection of designs to consider as a gas turbine engine is being developed. Such a benefit provides more insight to the requirements for a given gas turbine engine well before specific technologies, integration and system requirements are developed fully. Such a benefit avoids late-stage redesign.

[0107] The desired relationship providing for the improved gas turbine engine, discovered by the inventors, is expressed as:

[00002] $CST = Fn_{Total} \times EGT / (A_{HPCExit^2} \times 1000)$, Expression(2) [0108] where CST is corrected specific thrust; Fn.sub.Total is a total sea level static thrust output of the gas turbine engine in pounds; EGT is redline exhaust gas temperature in degrees Celsius; and A.sub.HPCExit is a high pressure compressor exit area in square inches.

[0109] CST values of an engine defined by Expression (2) in accordance with various embodiments of the present disclosure are from 42 to 90, such as from 45 to 80, such as from 50 to 80. The units of the CST values may be pounds-degrees Celsius over square inches.

[0110] Referring now to FIGS. **6** and **7**, various exemplary gas turbine engines are illustrated in accordance with one or more exemplary embodiments of the present disclosure. In particular, FIG. **6** provides a table including numerical values corresponding to several of the plotted gas turbine engines in FIG. **7**. FIG. **7** is a plot **400** of gas turbine engines in accordance with one or more exemplary embodiments of the present disclosure, showing the CST on a Y-axis **402** and the EGT on an X-axis **404**.

[0111] As shown, the plot **400** in FIG. **7** depicts a first range **406**, with the CST values between 42 and 90 and EGT values from 800 degrees Celsius to 1400 degrees Celsius. FIG. **7** additionally depicts a second range **408**, with the CST values between 50 and 80 and EGT values from 1000 degrees Celsius to 1300 degrees Celsius. It will be appreciated that in other embodiments, the EGT value may be greater than 1100 degree Celsius and less than 1250 degrees Celsius, such as greater than 1150 degree Celsius and less than 1250 degrees Celsius, such as greater than 1000 degree Celsius and less than 1300 degrees Celsius.

[0112] It will be appreciated that although the discussion above is generally related to an open rotor engine having a particular cooled cooling air system **250** (FIG. **2**), in various embodiments of the present disclosure, the relationship outlined above with respect to Expression (2) may be applied to any other suitable engine architecture, including any other suitable technology(ies) to allow the gas turbine engine to accommodate higher temperatures to allow for a reduction in the high pressure compressor exit area, while maintaining or even increasing the maximum turbofan engine thrust output without, e.g., prematurely wearing various components within the turbomachine exposed the working gas flowpath.

[0113] For example, reference will now be made to FIG. **8**. FIG. **8** provides a schematic view of an engine **100** in accordance with another exemplary embodiment of the present disclosure. The exemplary embodiment of FIG. **8** may be configured in substantially the same manner as the exemplary engine **100** described above with respect to FIGS. **1** through **4**, and the same or similar reference numerals may refer to the same or similar parts. However, as will be appreciated, for the embodiment shown, the engine **100** further includes an outer housing or nacelle **298** circumferentially surrounding at least in part a fan section **150** and a turbomachine **120**. The nacelle

298 defines a bypass passage **194** between the nacelle **298** and the turbomachine **120**.
[0114] Briefly, it will be appreciated that the exemplary engine **100** of FIG. **8** is configured as a two-stream engine, i.e., an engine without a third stream (e.g., fan stream **172** in the exemplary engine **100** of FIG. **2**). With such a configuration, a total sea level static thrust output Fn.sub.Total of the engine **100** may generally be equal to a sum of: a fan stream thrust Fn.sub.Fan (i.e., an amount of thrust generated by a fan **152** through a bypass passage **194**) and a turbomachine thrust Fn.sub.TM (i.e., an amount of thrust generated by an airflow through a turbomachine exhaust nozzle **140**), each during the static, sea level, standard day conditions.
[0115] Further, for the exemplary embodiment of FIG. **8**, the engine **100** additionally includes a cooled cooling air system **250** configured to provide a turbine section with cooled cooling air during operation of the engine **100**, to allow the engine **100** to accommodate higher temperatures to allow for a reduction in a high pressure compressor exit area, while maintaining or even increasing a maximum turbofan engine thrust output.
[0116] It will be appreciated that in other exemplary embodiments of the present disclosure, the cooled cooling air system **250** of the engine **100** may be configured in any other suitable manner. For example, the exemplary cooled cooling air system **250** described above with reference to FIGS. **2** and **3** is generally configured as a thermal bus cooled cooling air system. However, in other embodiments, the cooled cooling air system **250** may instead be a dedicated heat exchanger cooled cooling air system (i.e., a cooled cooling air system including a heat exchanger that transfers heat directly to a cooling medium). Additionally, in other embodiments, the cooled cooling air system **250** may be a bypass heat exchanger cooled cooling air system having a heat sink heat exchanger thermally coupled to an airflow through a bypass passage (see, e.g., FIG. **9**, discussed below). Additionally, or alternatively, in other embodiments, the cooled cooling air system **250** may be one of an air-to-air cooled cooling air system (a cooled cooling air system having a heat sink heat exchanger configured to transfer heat to an airflow; see, e.g., FIG. **9**, discussed below); an oil-to-air cooled cooling air system (a cooled cooling air system having a heat sink heat exchanger configured to transfer heat to an oil flow); or a fuel-to-air cooled cooling air system (a cooled cooling air system having a heat sink heat exchanger configured to transfer heat to a fuel flow, such as a Jet A fuel flow, a liquid hydrogen or hydrogen gas fuel flow, etc.; see, e.g., FIG. **4**).
[0117] More particularly, referring generally to FIGS. **9** through **11**, in other exemplary embodiments, the cooled cooling air system **250** of the engine **100** may be configured in any other suitable manner. The exemplary engines **100** depicted in FIGS. **9** through **11** may be configured in a similar manner as exemplary engine **100** described above with reference to FIGS. **1** through **4**, and the same or similar numbers may refer to the same or similar parts.
[0118] For example, each of the exemplary engines **100** depicted in FIGS. **9** through **11** generally includes a turbomachine **120** having an LP compressor **126**, an HP compressor **128**, a combustion section **130**, an HP turbine **132**, and an LP turbine **134** collectively defining at least in part a working gas flowpath **142** and arranged in serial flow order. The exemplary turbomachine **120** depicted additionally includes a core cowl **122**, and the engine **100** includes a fan cowl **170**. The engine **100** includes or defines a fan duct **172** positioned partially between the core cowl **122** and the fan cowl **170**. Moreover, a bypass passage **194** is defined at least in part by the core cowl **122**, the fan cowl **170**, or both and extends over the turbomachine **120**.
[0119] Moreover, the exemplary engines **100** depicted in FIGS. **9** to **11** additionally include a cooled cooling air system **250**. The cooled cooling air system **250** generally includes a duct assembly **252** and a CCA heat exchanger **254**.
[0120] However, referring particular to FIG. **9**, it will be appreciated that for the exemplary embodiment depicted, the CCA heat exchanger **254** is positioned in thermal communication with the bypass passage **194**, and more specifically, it is exposed to an airflow through or over the bypass passage **194**. For the embodiment of FIG. **9**, the CCA heat exchanger **254** is positioned on the core cowl **122**. In such a manner, the CCA heat exchanger **254** may be an air-to-air CCA heat

exchanger configured to exchange heat between an airflow extracted from the HP compressor **128** and the airflow through the bypass passage **194**.

[0121] As is depicted in phantom, the cooled cooling air system **250** may additionally or alternatively be positioned at any other suitable location along the bypass passage **194**, such as on the fan cowl **170**. Further, although depicted in FIG. **9** as being positioned on the core cowl **122**, in other embodiments, the CCA heat exchanger **254** may be embedded into the core cowl **122**, and airflow through the bypass passage **194** may be redirected from the bypass passage **194** to the CCA heat exchanger **254**.

[0122] As will be appreciated, a size of the CCA heat exchanger **254** may affect the amount of drag generated by the CCA heat exchanger **254** being positioned within or exposed to the bypass passage **194**. Accordingly, sizing the cooled cooling air system **250** in accordance with the present disclosure may allow for a desired reduction in a HP compressor **128** exit area, while maintaining or even increasing a total thrust output for the engine **100**, without creating an excess amount of drag on the engine **100** in the process.

[0123] Referring now particular to FIG. **10**, it will be appreciated that for the exemplary embodiment depicted, the cooled cooling air system **250** is configured to receive the cooling airflow from an air source upstream of a downstream half of the HP compressor **128**. In particular, for the exemplary embodiment of FIG. **10**, the exemplary cooled cooling air system **250** is configured to receive the cooling airflow from a location upstream of the HP compressor **128**, and more specifically, still, from the LP compressor **126**. In order to allow for a relatively low pressure cooling airflow to be provided to a first stage **214** of HP turbine rotor blades **206** of the HP turbine **132**, the cooled cooling air system **250** further includes a pump **299** in airflow communication with the duct assembly **252** to increase a pressure of the cooling airflow through the duct assembly **252**. For the exemplary aspect depicted, the pump **299** is positioned downstream of the CCA heat exchanger **254**. In such a manner, the pump **299** may be configured to increase the pressure of the cooling airflow through the duct assembly **252** after the cooling airflow has been reduced in temperature by the CCA heat exchanger **254**. Such may allow for a reduction in wear on the pump **299**.

[0124] Referring now particularly to FIG. **11**, it will be appreciated that the cooled cooling air system **250** includes a high-pressure portion and a low-pressure portion operable in parallel. In particular, the duct assembly **252** includes a high-pressure duct assembly **252**A and a low-pressure duct assembly **252**B, and the CCA heat exchanger **254** includes a high-pressure CCA heat exchanger **254**A and a low-pressure CCA heat exchanger **254**B.

[0125] The high-pressure duct assembly **252**A is in fluid communication with the HP compressor **128** at a downstream half of the high-pressure compressor and is further in fluid communication with a first stage **214** of HP turbine rotor blades **206**. The high-pressure duct assembly **252**A may be configured to receive a high-pressure cooling airflow from the HP compressor **128** through the high-pressure duct assembly **252**A and provide such high-pressure cooling airflow to the first stage **214** of HP turbine rotor blades **206**. The high-pressure CCA heat exchanger **254**A may be configured to reduce a temperature of the high-pressure cooling airflow through the high-pressure duct assembly **252**A at a location upstream of the first stage **214** of HP turbine rotor blades **206**.

[0126] The low-pressure duct assembly **252**B is in fluid communication with a location upstream of the downstream half of the high-pressure compressor **128** and is further in fluid communication with the HP turbine **132** and a location downstream of the first stage **214** of HP turbine rotor blades **206**. In particular, for the embodiment depicted, the low-pressure duct assembly **252**B is in fluid communication with the LP compressor **126** and a second stage (not labeled) of HP turbine rotor blades **206**. The low-pressure duct assembly **252**B may be configured to receive a low-pressure cooling airflow from the LP compressor **126** through the low-pressure duct assembly **252**B and provide such low-pressure cooling airflow to the second stage of HP turbine rotor blades **206**. The low-pressure CCA heat exchanger **254**B may be configured to reduce a temperature of the low-pressure cooling airflow through the low-pressure duct assembly **252**B upstream of the second stage of HP turbine rotor blades **206**.

[0127] Inclusion of the exemplary cooled cooling air system **250** of FIG. **11** may reduce an amount of resources utilized by the cooled cooling air system **250** to provide a desired amount of cooling for the turbomachine **120**.

[0128] Further, for the exemplary embodiment of FIG. **11**, it will be appreciated that the cooled cooling air system **250** may further be configured to provide cooling to one or more stages of LP turbine rotor blades **210**, and in particular to a first stage (i.e., upstream-most stage) of LP turbine rotor blades **210**. Such may further allow for, e.g., the higher operating temperatures described herein.

[0129] Reference will now be made briefly to FIG. **12**. FIG. **12** provides a schematic view of an engine **500** in accordance with another exemplary embodiment of the present disclosure. The exemplary embodiment of FIG. **12** may be configured in substantially the same manner as the exemplary engine **100** described above with respect to FIGS. **1** through **4**, and the same or similar reference numerals may refer to the same or similar parts. However, as will be appreciated, for the embodiment shown, the engine **500** is configured as a three-spool engine, instead of a two-spool engine.

[0130] For example, the exemplary engine **500** includes a fan section **502** and a turbomachine **504**. The fan section includes a fan **506**. The turbomachine includes a first compressor **508**, a second compressor **510**, a combustion section **512**, a first turbine **514**, a second turbine **516**, and a third turbine **518**. The first compressor **508** may be a high pressure compressor, the second compressor **510** may be a medium pressure compressor (or intermediate pressure compressor), the first turbine **514** may be a high pressure turbine, the second turbine **516** may be a medium pressure turbine (or intermediate pressure turbine), and the third turbine **518** may be a low pressure turbine. Further, the engine **500** includes a first shaft **520** extending between, and rotatable with both of, the first compressor **508** and first turbine **514**; a second shaft **522** extending between, and rotatable with both of, the second compressor **510** and second turbine **516**; and a third shaft **524** extending between, and rotatable with both of, the third turbine **518** and fan **506**. In such a manner, it will be appreciated that the engine **500** may be referred to as a three-spool engine.

[0131] For the embodiment of FIG. **12**, the term redline EGT refers to a maximum temperature of an airflow after the first stator downstream of the last stage of rotor blades of the intermediate speed turbine, e.g., at location **526** in FIG. **12** (assuming the intermediate speed turbine **516** includes a stage of stator vanes downstream of the last stage of rotor blades).

[0132] It will further be appreciated that the exemplary cooled cooling air systems **250** described hereinabove are provided by way of example only. In other exemplary embodiments, aspects of one or more of the exemplary cooled cooling air systems **250** depicted may be combined to generate still other exemplary embodiments. For example, in still other exemplary embodiments, the exemplary cooled cooling air system **250** of FIGS. **2** through **4** may not be utilized with a thermal transport bus (e.g., thermal transport bus **300**), and instead may directly utilize a CCA heat exchanger **254** positioned within the fan duct **172**. Similarly, in other example embodiment, the exemplary cooled cooling air systems **250** of FIGS. **9** through **11** may be utilized with a thermal transport bus (e.g., thermal transport bus **300** of FIG. **2**, **4** or **5**) to reject heat for the CCA heat exchanger **254**. Additionally, although the exemplary cooled cooling air systems **250** depicted schematically in FIGS. **9** through **11** depict the duct assembly **252** as positioned outward of the working gas flow path **142** along the radial direction R, in other exemplary embodiments, the duct assemblies **252** may extend at least partially inward of the working gas flow path **142** along the radial direction R (see, e.g., FIG. **4**). In still other exemplary embodiments, the cooled cooling air system **250** may include duct assemblies **252** positioned outward of the working gas flow path **142** along the radial direction R and inward of the working gas flow path **142** along the radial direction R (e.g., in FIG. **11**, the high-pressure duct assembly **252**A may be positioned inwardly of the working gas flow path **142** along the radial direction R and the low-pressure duct assembly **252**B may be positioned outwardly of the working gas flow path **142** along the radial direction R).

[0133] Moreover, it will be appreciated that in still other exemplary aspects, the gas turbine engine may include additional or alternative technologies to allow the gas turbine engine to accommodate higher temperatures while maintaining or even increasing the maximum turbofan engine thrust output, as may be indicated by a reduction in the high pressure compressor exit area, without, e.g., prematurely wearing on various components within the turbomachine exposed to the working gas flowpath.

[0134] For example, in additional or alternative embodiments, a gas turbine engine may incorporate advanced materials capable of withstanding the relatively high temperatures at downstream stages of a high pressure compressor exit (e.g., at a last stage of high pressure compressor rotor blades), and downstream of the high pressure compressor (e.g., a first stage of an HP turbine, downstream stages of the HP turbine, an LP turbine, an exhaust section, etc.).

[0135] In particular, in at least certain exemplary embodiments, a gas turbine engine of the present disclosure may include an airfoil (e.g., rotor blade or stator vane) in one or more of the HP compressor, the first stage of the HP turbine, downstream stages of the HP turbine, the LP turbine, the exhaust section, or a combination thereof formed of a ceramic-matrix-composite or “CMC.” As used herein, the term CMC refers to a class of materials that include a reinforcing material (e.g., reinforcing fibers) surrounded by a ceramic matrix phase. Generally, the reinforcing fibers provide structural integrity to the ceramic matrix. Some examples of matrix materials of CMCs can include, but are not limited to, non-oxide silicon-based materials (e.g., silicon carbide, silicon nitride, or mixtures thereof), oxide ceramics (e.g., silicon oxycarbides, silicon oxynitrides, aluminum oxide (Al.sub.2O.sub.3), silicon dioxide (SiO.sub.2), aluminosilicates, or mixtures thereof), or mixtures thereof. Optionally, ceramic particles (e.g., oxides of Si, Al, Zr, Y, and combinations thereof) and inorganic fillers (e.g., pyrophyllite, wollastonite, mica, talc, kyanite, and montmorillonite) may also be included within the CMC matrix.

[0136] Some examples of reinforcing fibers of CMCs can include, but are not limited to, non-oxide silicon-based materials (e.g., silicon carbide, silicon nitride, or mixtures thereof), non-oxide carbon-based materials (e.g., carbon), oxide ceramics (e.g., silicon oxycarbides, silicon oxynitrides, aluminum oxide (Al.sub.2O.sub.3), silicon dioxide (SiO.sub.2), aluminosilicates such as mullite, or mixtures thereof), or mixtures thereof.

[0137] Generally, particular CMCs may be referred to as their combination of type of fiber/type of matrix. For example, C/SiC for carbon-fiber-reinforced silicon carbide; SiC/SiC for silicon carbide-fiber-reinforced silicon carbide, SiC/SiN for silicon carbide fiber-reinforced silicon nitride; SiC/SiC—SiN for silicon carbide fiber-reinforced silicon carbide/silicon nitride matrix mixture, etc. In other examples, the CMCs may include a matrix and reinforcing fibers comprising oxide-based materials such as aluminum oxide (Al.sub.2O.sub.3), silicon dioxide (SiO.sub.2), aluminosilicates, and mixtures thereof. Aluminosilicates can include crystalline materials such as mullite (3Al2O3 2SiO2), as well as glassy aluminosilicates.

[0138] In certain embodiments, the reinforcing fibers may be bundled and/or coated prior to inclusion within the matrix. For example, bundles of the fibers may be formed as a reinforced tape, such as a unidirectional reinforced tape. A plurality of the tapes may be laid up together to form a preform component. The bundles of fibers may be impregnated with a slurry composition prior to forming the preform or after formation of the preform. The preform may then undergo thermal processing, such as a cure or burn-out to yield a high char residue in the preform, and subsequent chemical processing, such as melt-infiltration with silicon, to arrive at a component formed of a CMC material having a desired chemical composition.

[0139] Such materials, along with certain monolithic ceramics (i.e., ceramic materials without a reinforcing material), are particularly suitable for higher temperature applications. Additionally, these ceramic materials are lightweight compared to superalloys, yet can still provide strength and durability to the component made therefrom. Therefore, such materials are currently being considered for many gas turbine components used in higher temperature sections of gas turbine engines, such as airfoils (e.g., turbines, and vanes), combustors, shrouds and other like components, that would benefit from the lighter-weight and higher temperature capability these materials can offer.

[0140] One or more of these components formed of a CMC material may include an environmental-barrier-coating or "EBC." The term EBC refers to a coating system including one or more layers of ceramic materials, each of which provides specific or multi-functional protections to the underlying CMC. EBCs generally include a plurality of layers, such as rare earth silicate coatings (e.g., rare earth disilicates such as slurry or APS-deposited yttrium ytterbium disilicate (YbYDS)), alkaline earth aluminosilicates (e.g., including barium-strontium-aluminum silicate (BSAS), such as having a range of BaO, SrO, Al_2O_3, and/or SiO_2 compositions), hermetic layers (e.g., a rare earth disilicate), and/or outer coatings (e.g., comprising a rare earth monosilicate, such as slurry or APS-deposited yttrium monosilicate (YMS)). One or more layers may be doped as desired, and the EBC may also be coated with an abradable coating.

[0141] In such a manner, it will be appreciated that the EBCs may generally be suitable for application to "components" found in the relatively high temperature environments noted above. Examples of such components can include, for example, combustor components, turbine blades, shrouds, nozzles, heat shields, and vanes.

[0142] Additionally, or alternatively still, in other exemplary embodiments, a gas turbine engine of the present disclosure may include an airfoil (e.g., rotor blade or stator vane) in one or more of an HP compressor, a first stage of an HP turbine, downstream stages of the HP turbine, an LP turbine, an exhaust section, or a combination thereof formed in part, in whole, or in some combination of materials including but not limited to titanium, nickel, and/or cobalt based superalloys (e.g., those available under the name Inconel® available from Special Metals Corporation). One or more of these materials are examples of materials suitable for use in an additive manufacturing processes.

[0143] Further, it will be appreciated that in at least certain exemplary embodiments of the present disclosure, a method of operating a gas turbine engine is provided. The method may be utilized with one or more of the exemplary gas turbine engines discussed herein, such as in FIGS. **1** through **4** and **8** through **11**. The method includes operating the gas turbine engine at a takeoff power level, the gas turbine engine having a turbomachine with a high pressure compressor defining a high pressure compressor exit area ($A_{HPCExit}$) in square inches. The gas turbine engine further defines a redline exhaust gas temperature (EGT) in degrees Celsius, a total sea level static thrust output (Fn_{Total}) in pounds, and a corrected specific thrust. The corrected specific thrust is greater than or equal to 42 and less than or equal to 90, the corrected specific thrust determined as follows: $Fn_{Total} \times EGT/(A_{HPCExit}^2 \times 1000)$.

[0144] In certain exemplary aspects, operating the gas turbine engine at the takeoff power level further includes reducing a temperature of a cooling airflow provided to a high pressure turbine of the gas turbine engine with a cooled cooling air system. For example, in certain exemplary aspects, reducing the temperature of the cooling airflow provided to the high pressure turbine of the gas turbine engine with the cooled cooling air system comprises providing a temperature reduction of the cooling airflow equal to at least 15% of the EGT and up to 45% of the EGT.

[0145] As will be appreciated from the description herein, various embodiments of a gas turbine engine are provided. Certain of these embodiments may be an unducted, single rotor gas turbine engine (see FIG. **1**), a turboprop engine, or a ducted turbofan engine (see FIG. **8**). Another example of a ducted turbofan engine can be found in U.S. patent application Ser. No. 16/811,368 (Published as U.S. Patent Application Publication No. 2021/0108597), filed Mar. 6, 2020 (FIG. 10, Paragraph [0062], et al.; including an annular fan case **13** surrounding the airfoil blades **21** of rotating element **20** and surrounding vanes **31** of stationary element **30**; and including a third stream/fan duct **73** (shown in FIG. **10**, described extensively throughout the application)). Various additional aspects of one or more of these embodiments are discussed below. These exemplary aspects may be combined with one or more of the exemplary gas turbine engine(s) discussed above with respect to the FIGS.

[0146] For example, in some embodiments of the present disclosure, the engine may include a heat exchanger located in an annular duct, such as in a third stream. The heat exchanger may extend substantially continuously in a circumferential direction of the gas turbine engine (e.g., at least 300 degrees, such as at least 330 degrees).

[0147] In one or more of these embodiments, a threshold power or disk loading for a fan (e.g., an unducted single rotor or primary forward fan) may range from 25 horsepower per square foot (hp/ft.sup.2) or greater at cruise altitude during a cruise operating mode. In particular embodiments of the engine, structures and methods provided herein generate power loading between 80 hp/ft.sup.2 and 160 hp/ft.sup.2 or higher at cruise altitude during a cruise operating mode, depending on whether the engine is an open rotor or ducted engine.

[0148] In various embodiments, an engine of the present disclosure is applied to a vehicle with a cruise altitude up to approximately 65,000 ft. In certain embodiments, cruise altitude is between approximately 28,000 ft and approximately 45,000 ft. In still certain embodiments, cruise altitude is expressed in flight levels based on a standard air pressure at sea level, in which a cruise flight condition is between FL280 and FL650. In another embodiment, cruise flight condition is between FL280 and FL450. In still certain embodiments, cruise altitude is defined based at least on a barometric pressure, in which cruise altitude is between approximately 4.85 psia and approximately 0.82 psia based on a sea level pressure of approximately 14.70 psia and sea level temperature at approximately 59 degrees Fahrenheit. In another embodiment, cruise altitude is between approximately 4.85 psia and approximately 2.14 psia. It should be appreciated that in certain embodiments, the ranges of cruise altitude defined by pressure may be adjusted based on a different reference sea level pressure and/or sea level temperature.

[0149] In various exemplary embodiments, the fan (or rotor) may include twelve (12) fan blades. From a loading standpoint, such a blade count may allow a span of each blade to be reduced such that the overall diameter of the primary fan may also be reduced (e.g., to twelve feet in one exemplary embodiment). That said, in other embodiments, the fan may have any suitable blade count and any suitable diameter. In certain suitable embodiments, the fan includes at least eight (8) blades. In another suitable embodiment, the fan may have at least twelve (12) blades. In yet another suitable embodiment, the fan may have at least fifteen (15) blades. In yet another suitable embodiment, the fan may have at least eighteen (18) blades. In one or more of these embodiments, the fan includes twenty-six (26) or fewer blades, such as twenty (20) or fewer blades. Alternatively, in certain suitable embodiments, the fan may only include at least four (4) blades, such as with a fan of a turboprop engine.

[0150] Further, in certain exemplary embodiments, the rotor assembly may define a rotor diameter (or fan diameter) of at least 10 feet, such as at least 11 feet, such as at least 12 feet, such as at least 13 feet, such as at least 15 feet, such as at least 17 feet, such as up to 28 feet, such as up to 26 feet, such as up to 24 feet, such as up to 18 feet.

[0151] In various embodiments, it will be appreciated that the engine includes a ratio of a quantity of vanes to a quantity of blades that could be less than, equal to, or greater than 1:1. For example, in particular embodiments, the engine includes twelve (12) fan blades and ten (10) vanes. In other embodiments, the vane assembly includes a greater quantity of vanes to fan blades. For example, in particular embodiments, the engine includes ten (10) fan blades and twenty-three (23) vanes. For example, in certain embodiments, the engine may include a ratio of a quantity of vanes to a quantity of blades between 1:2 and 5:2. The ratio may be tuned based on a variety of factors including a size of the vanes to ensure a desired amount of swirl is removed for an airflow from the primary fan.

[0152] Additionally, in certain exemplary embodiments, where the engine includes the third stream and a mid-fan (a ducted fan aft of the primary, forward fan), a ratio R1/R2 may be between 1 and 10, or 2 and 7, or at least 3.3, at least 3.5, at least 4 and less than or equal to 7, where R1 is the radius of the primary fan and R2 is the radius of the mid-fan.

[0153] It should be appreciated that various embodiments of the engine, such as the single unducted rotor engine depicted and described herein, may allow for normal subsonic aircraft cruise altitude operation at or above Mach 0.5. In certain embodiments, the engine allows for normal aircraft operation between Mach 0.55 and Mach 0.85 at cruise altitude. In still particular embodiments, the engine allows for normal aircraft operation between Mach 0.75 and Mach 0.85. In certain embodiments, the engine allows for rotor blade tip speeds at or less than 750 feet per second (fps). In other embodiments, the rotor blade tip speed at a cruise flight condition can be 650 to 900 fps, or 700 to 800 fps. Alternatively, in certain suitable embodiments, the engine allows for normal aircraft operation of at least Mach 0.3, such as with turboprop engines.

[0154] A fan pressure ratio (FPR) for the primary fan of the fan assembly can be 1.04 to 2.20, or in some embodiments 1.05 to 1.2, or in some embodiments less than 1.08, as measured across the fan blades of the primary fan at a cruise flight condition.

[0155] In order for the gas turbine engine to operate with a fan having the above characteristics to define the above FPR, a gear assembly may be provided to reduce a rotational speed of the fan assembly relative to a driving shaft (such as a low pressure shaft coupled to a low pressure turbine). In some embodiments, a gear ratio of the input rotational speed to the output rotational speed is between 3.0 and 4.0, between 3.2 and 3.5, or between 3.5 and 4.5. In some embodiments, a gear ratio of the input rotational speed to the output rotational speed is greater than 4.1. For example, in particular embodiments, the gear ratio is within a range of 4.1 to 14.0, within a range of 4.5 to 14.0, or within a range of 6.0 to 14.0. In certain embodiments, the gear ratio is within a range of 3.2 to 12 or within a range of 4.5 to 11.0.

[0156] With respect to a turbomachine of the gas turbine engine, the compressors and/or turbines can include various stage counts. As disclosed herein, the stage count includes the number of rotors or blade stages in a particular component (e.g., a compressor or turbine). For example, in some embodiments, a low pressure compressor may include 1 to 8 stages, a high-pressure compressor may include 4 to 15 stages, a high-pressure turbine may include 1 to 2 stages, and/or a low pressure turbine (LPT) may include 1 to 7 stages. In particular, the LPT may have 4 stages, or between 4 and 6 stages. For example, in certain embodiments, an engine may include a one stage low pressure compressor, an 11 stage high pressure compressor, a two stage high pressure turbine, and 4 stages, or between 4 and 7 stages for the LPT. As another example, an engine can include a three stage low-pressure compressor, a 10 stage high pressure compressor, a two stage high pressure turbine, and a 7 stage low pressure turbine.

[0157] A core engine is generally encased in an outer casing defining one half of a core diameter (Dcore), which may be thought of as the maximum extent from a centerline axis (datum for R). In certain embodiments, the engine includes a length (L) from a longitudinally (or axial) forward end to a longitudinally aft end. In various embodiments, the engine defines a ratio of L/Dcore that provides for reduced installed drag. In one embodiment, L/Dcore is at least 2. In another embodiment, L/Dcore is at least 2.5. In some embodiments, the L/Dcore is less than 5, less than 4, and less than 3. In various embodiments, it should be appreciated that the L/Dcore is for a single unducted rotor engine.

[0158] The reduced installed drag may further provide for improved efficiency, such as improved specific fuel consumption. Additionally, or alternatively, the reduced installed drag may provide for cruise altitude engine and aircraft operation at the above describe Mach numbers at cruise altitude. Still particular embodiments may provide such benefits with reduced interaction noise between the blade assembly and the vane assembly and/or decreased overall noise generated by the engine by virtue of structures located in an annular duct of the engine.

[0159] Additionally, it should be appreciated that ranges of power loading and/or rotor blade tip speed may correspond to certain structures, core sizes, thrust outputs, etc., or other structures of the core engine. However, as previously stated, to the extent one or more structures provided herein may be known in the art, it should be appreciated that the present disclosure may include combinations of structures not previously known to combine, at least for reasons based in part on conflicting benefits versus losses, desired modes of operation, or other forms of teaching away in the art.

[0160] Although depicted above as an unshrouded or open rotor engine, it should be appreciated that aspects of the disclosure provided herein may be applied to shrouded or ducted engines, partially ducted engines, aft-fan engines, or other gas turbine engine configurations, including those for marine, industrial, or aero-propulsion systems. Certain aspects of the disclosure may be applicable to turbofan, turboprop, or turboshaft engines. However, it should be appreciated that certain aspects of the disclosure may address issues that may be particular to unshrouded or open rotor engines, such as, but not limited to, issues related to gear ratios, fan diameter, fan speed, length (L) of the engine, maximum diameter of the core engine (Dcore) of the engine, L/Dcore of the engine, desired cruise altitude, and/or desired operating cruise speed, or combinations thereof.

[0161] The operating environment within gas turbine engines, such as discussed above, is both thermally and chemically hostile. Although high temperature iron, nickel and cobalt-based superalloys have been developed for engine components, components formed from such alloys often cannot withstand long service exposures if located in certain sections of a gas turbine engine, such as the turbine and/or combustor sections. Hot corrosion of gas turbine engine components generally occurs when sulfur compounds and/or dust attacks the components' surfaces. Sources of sulfur compounds include fuel and ingestion from environment. Dust predominately comes from environment ingestion. The presence of corrosive compounds and/or dust is responsible for corrosion of hot section components like compressors or disks. Therefore, improved anti-corrosion coatings are needed.

[0162] In various exemplary aspects of the present disclosure, a three-stream gas turbine engine is provided that includes at least one coated component having a substrate and a corrosion resistant coating thereon. For example, incorporating the coated component comprising the corrosion resistant coating in the gas turbine engine can allow for the gas turbine engine to increase efficiency by, e.g., providing particular properties to the coated component or coated components within particular sections of the engine that may increase efficiency. For example, the properties of the coated component(s) may have a combination of desirable properties, such as tailored thermal properties (e.g., capable of use in hotter conditions). The composition of the corrosion resistant coating may also be tailored to the particular component. In one embodiment, for example, a coated component including the corrosion resistant coating may be capable of running within the gas turbine engine at the same stress conditions but at hotter temperatures than components formed from other materials. Furthermore, disclosed hereinbelow are exemplary coated components that may be utilized within the engine. Thus, such a gas turbine engine can exhibit enhanced operability during certain mission requirements by designing the gas turbine engine to include the coated component(s) comprising a corrosion resistant coating.

[0163] Anti-corrosion coatings for substrates are also provided, which can be applied to components of the gas turbine engines described herein. For example, certain aspects of the present disclosure are directed to coatings for inhibiting corrosion of high temperature components, such as gas turbine components.

[0164] This disclosure relates to a corrosion resistant coating composition including a particulate corrosion resistant component, a glass-forming matrix component, and a glass-forming additive. This disclosure also broadly relates to a gas turbine engine component coated with at least one layer of the corrosion resistant coating composition. A method for coating the component with the corrosion resistant coating is also provided.

[0165] Turbine engine components for use at the highest operating temperatures are typically made of superalloys of iron, nickel, cobalt or a combination thereof, or other corrosion resistant materials such as stainless steels selected for good, elevated temperature toughness and fatigue resistance. Illustrative superalloys are designated by such trade names as Inconel®, for example, Inconel® 600, Inconel® 722 and Inconel® 718, Nimonic®, Rene®, for example, Rene® 65, Rene® 88DT, Rene® 104, Rene® 95, Rene® 100, Rene® 80 and Rene® 77, and Udimet®, for example, Udimet® 500, Hastelloy, for example, Hastelloy X, HS 188 and other similar alloys. These materials have resistance to oxidation and corrosion damage, but that resistance is generally not sufficient to protect them at sustained operating temperatures now being reached in gas turbine engines. Engine components, such as disks and other rotor components, are made from newer generation alloys that contain lower levels of chromium, and can therefore be more susceptible to corrosion attack. These engine components include turbine disks, turbine seal elements, turbine shafts, airfoils categorized as either rotating blades or stationary vanes, turbine blade retainers, center bodies, engine liners, flaps, and spools. Engine components could also include spools (e.g., compressor spools) and other compressor components, especially those components in the high pressure turbine and/or compressor section. This list is exemplary and not meant to be inclusive.

[0166] While all of the above listed components may find advantage for the present disclosure, engine components such as the turbine disks, turbine seal elements and turbine shafts are not directly within the gas path of the products of combustion, and are not typically identified with corrosive products experienced as a result of exposure to these highly corrosive and oxidative gases. Nevertheless, these components have experienced higher operating temperatures and are experiencing greater corrosion effects as a result of these higher operating temperatures.

[0167] Accordingly, anti-corrosion coating systems and coatings have been developed for protecting these components from oxidation and hot corrosion. The present disclosure is directed to a coating comprising a matrix, corrosion resistant particles, and a glass-forming additive. Advantageously, the inclusion of the glass-forming additive facilitates the formation of glass during curing of the applied coating or use of the coated component, which provides improved binding strength of the coating to components at high temperatures. Such improved binding strength of the present coating eliminates the need for additional sealant coats or top-coats. Furthermore, the present coating is free from harmful chromium-based materials, (e.g., hexavalent chromium) and is compliant with Registration, Evaluation, Authorization, and Restriction of Chemicals “REACH” regulations.

[0168] FIG. **13** is a perspective view of a gas turbine engine disk **882**, such as first stage disk or second stage disk of FIG. **1** or **8**, which is typically made of a superalloy material, such as one of the superalloy materials previously discussed. The disk **882** includes a hub **874** along typically the engine centerline that includes a bore through which a shaft (not shown) extends. The disk includes dovetail slots **886** along the disk outer periphery into which the turbine blades (e.g., turbine blades from FIG. **1**) are inserted. A web section **878** of the disk **882** extends between the outer periphery, where the dovetail slots are located, and the hub **874**. While the disclosed coating, including the glass-forming matrix including silica, corrosion resistant particles, and glass-forming additive, may be utilized anywhere along disk **882**, including the dovetail slots **886**, it finds particular use along the surfaces of web section **878** and the dovetail slots **886**, which unlike the bore in hub **874**, are directly exposed to the high temperature cooling air.

[0169] FIG. **14** depicts, in cross-section, the coating of the present disclosure deposited on an engine component. Corrosion resistant coating **864** is deposited on the surface **862** of substrate **860**. The substrate **860** may be any component used in a gas turbine engine, such as a turbine engine disks, blades, and/or retainers as described hereinabove. The surface **862** can be any surface of an engine component, for example, the surface **862** can be a web section of a turbine disk (as shown in FIG. **13**). The substrate **860** comprises a superalloy based on nickel, cobalt, iron or a combination thereof. The corrosion resistant coating **864** includes one or more corrosion resistant particles **865** dispersed throughout a matrix **863**.

[0170] Matrix forming materials preferably comprise a silicon-based material, silicone-based material, or a combination thereof. In certain embodiments, the matrix forming material contains silica. For example, silicone resins can be used, such as those with high ceramic yield, including polycarbosilane-based resins, polysilazane-based resins, polysilane-based resins, poly siloxane based resins, or combinations thereof. Commercially available resins such as SR 350™ available from Arkema® S.A., and DOWSIL™ RSN-0249 Flake Resin available from Dow® Chemical, can also be used.
[0171] Optionally, additional phosphate binder materials can be added. The phosphate binders can be in the form of phosphoric acid or more typically the respective phosphate compounds/compositions, including orthophosphates, pyrophosphates, and other phosphate compounds. These phosphate compounds/compositions can be monobasic, dibasic, tribasic or any combination thereof. The phosphate-containing binder component can optionally comprise other binder material, including one or more chromates, molybdates, or similar binder material. In certain embodiments, the matrix material is substantially free of chromates, including hexavalent chromium.
[0172] Corrosion resistant particles are combined with the matrix forming material to form a coating composition. For example, corrosion resistant particles can be combined with the matrix forming material and formed into a slurry, which will be discussed in more detail below. The corrosion resistant particles may include refractory oxide particles that can impart corrosion resistance to the coating, such as alumina (Al.sub.2O.sub.3), yttrium oxide (Y.sub.2O.sub.3), zirconium oxide (ZrO.sub.2), titanium oxide (TiO.sub.2) or a combination thereof. While oxides of other metals may be used, such as tungsten, chromium and rhenium, these are less preferred as they are not deemed to be as environmentally friendly as the refractory oxides. In certain embodiments, the corrosion resistant particles include MCrAlX, MCr, MAI, MCrX, or MAlX particles, where M is an element selected from nickel, iron, cobalt, or a combination thereof and X is an element selected from the group consisting of La, Ta, Re, Y, Zr, Hf, Si, B, C or a combination thereof. The corrosion resistant particles including MCrAlX, MCr, MAI, MCrX, or MAlX particles may be added to the coating, either alone or in combination with the refractory oxide particles. For example, in one embodiment, the coating includes CoNiCrAlY particles in combination with zirconia particles. The dispersed zirconia particles and the CoNiCrAlY particles provide the coating with corrosion resistance.
[0173] In embodiments, the corrosion resistant particles can include a plurality of particles having a multimodal distribution. For example, the corrosion resistant particles include a plurality of small particles having a median particle size of less than 1 micron, a plurality of medium particles having a particle size of between 2 microns and 8 microns, and a plurality of large particles having a particle size of between 9 microns and 60 microns. The plurality of small particles is present in an amount of about 10 volume % to about 30 volume %, the plurality of medium particles is present in an amount of about 30 volume % to about 50 volume %, and the plurality of large particles is present in an amount of about 30 volume % to about 50 volume %.
[0174] A glass-forming additive is also combined with the matrix material and corrosion resistant particles. Specifically, the glass-forming additive reacts with one or more of the matrix materials (e.g., silicon species such as silica) to facilitate glass formation during heat treatment of the component. Suitable heat treatments include curing, sintering, or use of the coated component. As previously stated, facilitation and formation of a glassy-phase in the coating provides improved binding strength at high temperatures. Advantageously, use of one or more glass-forming additives to form a glassy-phase in the coating diminishes the need for additional sealing top-coats or other coating layers. Furthermore, formation of the glassy-phase improves adhesion strength of the coating. The glass-forming additive can include oxides such as iron oxide, gallium oxide, manganese oxide, aluminum oxide, nickel oxide, titanium oxide, boron oxide, alkaline earth oxides, or a combination thereof. The glass-forming additive can also include iron, aluminum,

nickel, alloys thereof, or a combination thereof. In certain embodiments, the glass-forming additive includes boron, such as elemental boron, an unoxidized boron in a boron alloy or compound, or a combination thereof. In embodiments wherein the glass-forming additive includes boron, boron and silicon species (e.g., silica) can react to form a silica-rich or borosilicate-rich glass when cured or heated.

[0175] The glass-forming additive can include a powder having plurality of particles. Generally, the particles of the glass-forming additive are selected such that the overall particle size distribution of the glass-forming additive particles are much smaller than the corrosion resistant particles. For example, in embodiments the glass-forming additive can include a plurality of particles having particle sizes in the range of about 50 nm to about 200 nm.

[0176] When formulated as a slurry, the glass-forming additive is present in the slurry in an amount of about 0.01 wt. % to about 0.07 wt. %, such as about 0.05 wt. % based on the total weight of the slurry.

[0177] Advantageously, the components (e.g., the matrix, corrosion resistant particles, and glass-forming additive) utilized to form the coating disclosed herein are substantially free from harmful chromates, such as hexavalent chromium.

[0178] In embodiments, a nucleating agent may also be present in the coating slurry. Components coated with certain glass matrix particulate composite coatings can operate at temperatures lower than glass transition temperature. Glass transition temperature is the temperature where the macro-mobility of glass structure is observed. Normally, the glass transition temperature is lower than the glass melting (formation) temperature. Therefore, the operating temperature of the coated components is restricted well below the processing temperature of the coating. Interaction of nucleating agents with the glass matrix leads to the formation of nanocrystalline structures throughout the glass matrix. Formation of such nanocrystalline structures allows the coating to be used on components subjected to higher service temperatures, as the melting temperatures of these crystalline phases are higher than amorphous phases of similar compositions. In some cases, due to crystallinity in the glass as result of adding the nucleating agent, operating temperatures may well exceed the processing temperature of the coating. Exemplary materials suitable for use as nucleating agents include noble metals, such as ruthenium (Ru), rhodium (Rh), palladium (Pd), silver (Ag), osmium (Os), iridium (Ir), platinum (Pt) and gold (Au), fluorides, titanium dioxide, phosphorus pentoxide, chromium (III) oxide, or a combination thereof. This list is exemplary only and any other known, suitable nucleating agents can be used in accordance with the present disclosure.

[0179] As noted, the matrix forming material, corrosion resistant particles, and glass-forming additive can be combined and dispersed in a carrier fluid to form a slurry. The slurry viscosity can then be adjusted by either adding liquid or adding additional particles to the mixture. For example, one or more viscosity modifiers can be added to the slurry. Suitable viscosity modifiers include polyethylene glycol (PEG), dimethylsiloxane, silicone oil, phthalates, adipates, glycerin, or a combination thereof. The viscosity should be adjusted, if required, to be consistent with the intended method of application. For example, if the slurry is to be sprayed, the viscosity should be adjusted to be very low, whereas if the slurry is to be applied as a gel, using, for example, a doctor blade to adjust the thickness, then liquid should be removed so that the slurry does not flow readily. Even more liquid should be removed if the slurry is to be formed into a tape. In the last two examples, the final viscosity adjustment may be made after mixing is complete. Regardless of the intended method of application, the mixture is thoroughly agitated. Agitation can be accomplished by any convenient method. Optionally, surfactants and dispersants may also be added to the slurry depending on the method of application.

[0180] FIG. **15** illustrates a flowchart for a method (**800**) of coating a metal component with the corrosion resistant coating disclosed herein. At (**802**), the method includes disposing a corrosion resistant coating on the surface of a metal component. The coating includes a matrix, corrosion

resistant particles, and glass-forming additive. The coating may be applied by any known coating technique, including, but not limited to brushing, rolling, spraying, and dipping.
[0181] In certain embodiments, the corrosion resistant coating is disposed by spraying. For example, a coating slurry can be sprayed onto a surface of the component. As previously described, the viscosity of the coating slurry can be adjusted so that the coating can easily be sprayed. In such embodiments, the slurry is continuously agitated by placing it on a ball mill until it is ready for application. Even as the slurry is sprayed, the slurry can be pneumatically agitated by using a pot on a spray gun. The slurry can be applied using a spray gun having an adjustable orifice. The orifice size should be larger than the largest particles in the slurry. The slurry is sprayed at a pressure of about 20-60 psi. The corrosion resistant coating is applied to a preselected thickness, with a larger orifice being selected when a thicker coating is desired. The corrosion resistant coating can be applied at a coating thickness of about 50 μm to about 1,000 μm, such as from about 100 μm to about 950 μm, such as from about 150 μm to about 900 μm, such as from about 200 μm to about 850 μm, such as from about 250 μm to about 800 μm, such as from about 300 μm to about 750 μm, such as from about 350 μm to about 700 μm, such as from about 400 μm to about 650 μm, such as from about 450 μm to about 600 μm. The coating can be applied to such thicknesses as a single layer, or can be applied as a plurality of distinct layers to achieve an overall thickness in these ranges.
[0182] Optionally, prior to applying the corrosion resistant coating on the surface of the metal component, the metal component can be pretreated mechanically, chemically, or both to make the surface more receptive for the corrosion resistant coating. Suitable pretreatment methods include grit blasting, with or without masking of surfaces that are not to be subjected to grit blasting, micromachining, shot peening, laser etching, treatment with chemical etchants such as those containing hydrochloric acid, hydrofluoric acid, nitric acid, ammonium bifluorides and mixtures thereof, treatment with water under pressure (i.e., water jet treatment), with or without loading with abrasive particles, as well as various combinations of these methods. Typically, the surface of the metal component is pretreated by grit blasting where the surface is subjected to the abrasive action of silicon carbide particles, steel particles, alumina particles or other types of abrasive particles. These particles used in grit blasting are typically alumina particles and typically have a particle size from about 600 to about 35 mesh (from about 25 to about 500 micrometers), more typically from about 360 to about 35 mesh (from about 35 to about 500 micrometers). Other suitable pretreatment methods can include shot-peening and/or laser shock peening. One or more pretreatment methods can be utilized. For example, certain embodiments can include pretreating the surface via shot-peening followed by grit blasting.
[0183] Optionally, after the coating slurry is applied to the surface of the component, it is allowed to dry. Drying can be accomplished via any suitable method. Drying can remove unbound water or provide solvent evaporation from the applied slurry coating. Drying can take place at ambient temperature or the article can be placed in an oven and dried at a temperature greater than ambient. In other embodiments, the environmental humidity may be adjusted to facility drying of the slurry. For example, in embodiments the coated component can be heated to a temperature of 100° C. for about 60 minutes to dry the applied coating.
[0184] At (**804**), the coating is heat treated. Suitable heat treatments include curing, sintering, or use of the coated component at an elevated temperature. For example, the coating component can be heated to drive off unbound water and to cure the coating. Alternatively, heat can be applied to the surface of the coated component to cure the coating or facilitate curing of the coating. For example, heat may be applied to the surface to decrease the time required to cure. In certain embodiments, the coated component can be heated to a temperature of about 800° C. or greater for time period of about 30 minutes to 90 minutes, such as about 60 minutes to cure the coating. Further, the coated component can be subjected to more than one curing treatment or heat treatment. Additionally, heating or curing the coated component can take place in situ, such as

during operation of the part, facilitating on-wing repair. For example, the coating can be subjected to an initial treatment prior to use to dry and/or partially cure the coating, and the curing process is completed while the component is in use in an elevated temperature environment. A component in the turbine section of a turbine engine, for example, may experience elevated temperatures sufficient to complete curing of the coating during the first engine cycle following engine service.

[0185] A coating composition is generally provided that can be cured to form a corrosion resistant coating when applied over a turbine engine component or similar substrate. The composition includes a carrier liquid, a silicone binder, and corrosion-resistant particles selected from the group consisting of refractory particles and non-refractory particles. The corrosion-resistant particles provide the coating with the key corrosion resistance, while the silicon-based material is the binder during application and forms the matrix after curing. The corrosion resistant particles are substantially uniformly distributed in a silicon-based binder and carrier liquid to form a sprayable coating composition. On curing, the silicone binder forms a glassy silicate matrix, which upon firing, may convert at least partially to a glassy ceramic matrix.

[0186] In embodiments, the corrosion resistant particles are refractory particles such as alumina, zironcia, hafnia, stabilized zirconia and hafnia (e.g. yttria stabilized), ceria, chromia, magnesia, iron oxide, titania, yttria, and yttrium aluminum garnet (YAG), for example.

[0187] In embodiments, the corrosion resistant particles include refractory particles and at least one non-refractory particulate material having a CTE that is greater than alumina. Exemplary non-refractory materials include MAI, MAlX, MCr, MCrX, MCrAlX particles, or combinations thereof, where M is an element selected from iron, nickel, cobalt and combinations thereof and X is an element selected from the group of gamma prime formers, and solid solution strengtheners, consisting of, for example, Ta, Re or reactive elements, such as Y, Zr, Hf, Si, La or grain boundary strengtheners consisting of B, and C. Preferably, the non-refractory particles have a CTE that approximates the CTE of the underlying substrate.

[0188] Corrosion resistant coatings can provide resistance against corrosion caused by various corrodants, including metal (e.g., alkaline) sulfates, sulfites, chlorides, carbonates, oxides, and other corrodant salt deposits resulting from ingested dirt, volcanic ash, fly ash, concrete dust, sand, sea salt, etc., at temperatures as high as 2100° F. (1150° C.) and lower, although the components that the coating operate typically reach temperatures of about 1500° F. (815° C.). It is also possible to modify the coating composition by addition of elements to form a silicate-based ceramic coating upon firing, the coating having temperature capabilities in excess of 2100° F.

[0189] As noted above, because of the versatility of the coating, allowing it to be applied by different methods, the corrosion resistant coatings can be applied to thicknesses consistent with required engineering requirements as a monolithic layer, or can comprise a plurality of discrete layer(s) overlying the metal substrate. The particles are bound in the matrix, which may be glassy or glassy-ceramic depending upon the firing temperature. Typically, if desired, a glassy top coat can be applied over the corrosion resistant layer. The top coat can be applied for any number of reasons, for cosmetic purposes, for sealing, to provide anti-stick properties so that corrosion byproducts do not adhere to the component or for surface roughness improvements. A silicate glass or phosphate ($AlPO_4$ or $MgPO_4$) glass top coat is preferred, such as those commercially available phosphate top coats marketed by Sermatech International of Pottstown, PA under the trade names SermaSeal 565, SermaSeal 570A and by Coatings for Industry of Souderton, PA under the trade name Alseal 598.

[0190] FIG. **16** is a cross-sectional view depicting a portion of the turbine section of a gas turbine engine along the centerline of the engine. The turbine section **930** is a two stage turbine, although any number of stages may be employed depending on the turbine design. The present disclosure is not limited by the number of stages in the turbine. Turbine disks **932** are mounted on a shaft (not shown) extending through a bore in disks **932** along the centerline of the engine. A first stage blade **938** is attached to first stage disk **936**, while second stage blade **942** is attached to second stage disk

940. A vane **1310** extends from a casing **1320**. The inner surface of casing **1320** forms a liner **1330** for the hot gases of combustion, which flow in the gas flow path. The first stage blade **938**, the second stage blade **942** and the vane **1310** extend into the hot gas flow path. The vane is stationary and serves to direct the hot gas flow while blades **938**, **942** mounted on disks **936**, **940** rotate as the hot gases impinge on them, extracting energy to operate the engine.

[0191] Sealing elements **934**, a forward seal **944**, an aft seal **946**, an interstage seal **948**, a stage 1 aft blade retainer **950** and a stage 2 aft blade retainer **952**, serve to seal and complete the compressor air cooling circuits to the turbine blades and nozzles. These seals are attached to the disks and rotate with the disks. Interstage seal **948** is positioned inboard of vane **1310** and between the first stage disk **936** and the second stage disk **940**. Also shown are optional blade retainers **950**, **952** which lock the blades to the disks. The design of such retainers will vary dependent on engine design, with some engine designs not requiring them.

[0192] These disks, seals and blade retainers are heated to the temperatures of the cooling circuit air they direct. In addition, the parts closest to the combustion path are also heated by conductive heat transfer from the combustion path parts. For example, the rims of the turbine disks are conductively heated by the turbine blades. Contaminants in the cooling air, as previously discussed, deposit on the surfaces of the disks, seals and retainers that include the cooling cavities that provide the air that is the source of contamination at these elevated temperatures. Thus, the present disclosure can provide protection to any of these surfaces that are subject to corrosion due to deposition or accumulation of the cooling air contaminants.

[0193] FIG. **18** is a perspective view of a typical gas turbine engine disk **982** such as disk **936** or **940** of FIG. **16**, which is typically made of a superalloy material, such as one of the superalloy materials previously discussed. The disk **982** includes a hub **974** along typically the engine centerline that includes a bore through which a shaft (not shown) extends. The disk includes dovetail slots **986** along the disk outer periphery into which the turbine blades are inserted. A web section **978** of the disk **982** extends between the outer periphery, where the dovetail slots are located, and the hub. While the presently disclosed coatings may be utilized anywhere along disk **982**, including the dovetail slots, it finds particular use along the surfaces of web section **978** and the dovetail slots **986**, which unlike the hub **974**, is directly exposed to cooling air.

[0194] FIG. **17** depicts, in cross-section, the coating **964** in its simplest form, deposited on an engine component. Corrosion resistant coating **964** is deposited on the surface **962** of substrate **960**. The substrate **960** may be a turbine engine disk such as first stage disk **936** or second stage disk **940**. The substrate **960** may be a typical surface such as web section **978** of a turbine disk **982**. Substrate **960** may comprise superalloy based on nickel, cobalt, iron and combinations thereof, has deposited thereon a coating **964**. Optionally, an undercoating may be provided (not shown), such as a MCrAlX coating, for example a NiCrAlY or a CoNiCrAlY, an aluminide such as NiAl or noble metal-modified aluminide such as (Pt,Ni)Al. As discussed previously, coating **964** can be cured as a single layer of graded coating and surface **966** is exposed to the cooling air forming the environment for the surface. Alternatively coating **964** may be of substantially uniform composition. If the coating is to be graded, then additional layers are applied over coating layer **964**, the first layer being applied over outer surface **966** and additional layers being applied over subsequent outer coating layers.

[0195] Prior to forming the corrosion resistant coating **964** on the surface **962** of metal substrate **960**, metal surface **962** is typically pretreated mechanically, chemically or both to make the surface more receptive for coating **964**. Suitable pretreatment methods include grit blasting, with or without masking of surfaces that are not to be subjected to grit blasting (see U.S. Pat. No. 5,723,078 to Nagaraj et al, issued Mar. 3, 1998, especially col. 4, lines 46-66, which is incorporated by reference), micromachining, laser etching (see U.S. Pat. No. 5,723,078 to Nagaraj et al, issued Mar. 3, 1998, especially col. 4, line 67 to col. 5, line 3 and 14-17, which is incorporated by reference), treatment with chemical etchants such as those containing hydrochloric acid, hydrofluoric acid, nitric acid, ammonium bifluorides and mixtures thereof, (see, for example, U.S. Pat. No. 5,723,078 to Nagaraj et al, issued Mar. 3, 1998, especially col. 5, lines 3-10; U.S. Pat. No. 4,563,239 to Adinolfi et al, issued Jan. 7, 1986, especially col. 2, line 67 to col. 3, line 7; U.S. Pat. No. 4,353,780 to Fishter et al, issued Oct. 12, 1982, especially col. 1, lines 50-58; and U.S. Pat. No. 4,411,730 to Fishter et al, issued Oct. 25, 1983, especially col. 2, lines 40-51, all of which are incorporated by reference), treatment with water under pressure (i.e., water jet treatment), with or without loading with abrasive particles, as well as various combinations of these methods. Typically, the surface **962** of metal substrate **960** is pretreated by grit blasting where surface **962** is subjected to the abrasive action of silicon carbide particles, steel particles, alumina particles or other types of abrasive particles. These particles used in grit blasting are typically alumina particles and typically have a particle size of from about 600 to about 35 mesh (from about 25 to about 500 micrometers), more typically from about 360 to about 35 mesh (from about 35 to about 500 micrometers).

[0196] When additional layers of coating are to be applied over surface **966** in order to obtain a graded, multi-layer coating, it is generally not necessary to prepare coating surface **966** prior to application of additional layers. The coatings may be spray applied as a coating in thicknesses of from about 0.0001″ (0.1 mils) to about 0.005″ (5 mils), and preferably in thicknesses from about 0.0005″ (0.5 mils) to about 0.0025″ (2.5 mils). The coating can be applied to such thicknesses as a single layer, or can be applied as a plurality of distinct layers to achieve an overall thickness in these ranges.

[0197] The coating composition is applied and dried and cured to form a silica-based matrix having corrosion resistant particles substantially uniformly dispersed throughout. Corrosion resistance is provided by the corrosion resistant particles comprising refractory particles (designated “RP”) such as refractory oxides and nitrides, and non-refractory particles (designated “NRP”) such as MAI, MAlX, MCr, MCrX, MCrAlX, and combinations of these particles. Therefore, an embodiment having both RP and NRP is consistent with embodiments having both refractory and non-refractory corrosion resistant particles.

[0198] The silica-based matrix can be formulated in any one of a number of ways. For example, a solvent based system utilizes a silicone binder material that is mixed with a solvent (also referred to herein as a liquid carrier). A typical silicone binder material is SR-350 available from General Electric Silicones. An alternate silicone binder material is SR-355 available from General Electric Company, Wilton, Connecticut. SR350 and SR355 are understood to be methylsesquisiloxane mixtures of the polysiloxane family in amounts of up to about 45 weight percent of the binder composition. The solvent, typically an evaporable organic solvent, such as an alcohol (methanol, ethanol, propanol etc), acetone or other suitable solvent is mixed to obtain a viscosity consistent with the preferred method of application, as will be discussed. Next, the corrosion resistant particles are added to the solvent and silicone material solution. These particles may include refractory particles that can impart corrosion resistance to a coating such as, for example, alumina, yttrium oxide (Y2O5), zirconium oxide (Zr2O3), titanium oxide (TiO2), zirconia, hafnia, stabilized zirconia or hafnia (e.g. yttria stabilized or stabilized by other oxides-rare earths, magnesia, calcia, scandia), ceria (CeO.sub.2), chromia (Cr.sub.2O.sub.3), iron oxide (Fe.sub.2O.sub.3, Fe.sub.3O.sub.4), titania (TiO.sub.2), yttria (Y.sub.2O.sub.3), YAG (Y.sub.3Al.sub.5O.sub.12), magnesia (MgO), and combinations thereof. The selected refractory material must fit the following two criteria to be acceptable: (1) the particle must have a CTE equal to or higher than alumina (alumina has a CTE of about 4×10{circumflex over ( )}6 to about 5×10{circumflex over ( )}6 in/in/° F. at 1200° F.); and (2) must be more corrosion resistant than the substrate, preferably substantially inert to corrosion. To prepare the coating composition of the second embodiment, non-refractory particles are next added. As previously described, exemplary non-refractory particles include MAI, MAlX, MCr, MCrX, and MCrAlX, and combinations thereof. After the corrosion resistant particles have been added to the solution to form a slurry, the slurry viscosity is adjusted by either adding or removing solvent to the mixture to yield a composition viscosity that is consistent with the intended method of application. If the slurry is to be sprayed, the viscosity should be adjusted to be very low, whereas if the slurry is to be applied as a paste, using for example a doctor blade to adjust the thickness, then liquid should be removed so that the slurry does not flow readily. Additionally, surfactants and dispersants may optionally be added to the slurry when required. The selection and amount of corrosion resistant particles, binder, and solvent provide a coating composition that can be applied and cured to provide a corrosion resistant coating layer having a predetermined CTE.

[0199] In either embodiment of the coating composition, the corrosion resistant particles are added to the solvent and silicone so that the particles comprise from up to about 92% of the total solution by weight, the balance being the binder and solvent to render a sprayable composition. In the first embodiment of the coating composition, the slurry contains by weight from about 5% to about 45% binder, from about 3% to about 50% solvent and from about 15% to about 92% refractory particles by weight. In the second embodiment of the coating composition, the slurry contains by weight from about 5% to about 45% binder, from about 3% to about 50% solvent, from about 10% to about 87% non-refractory particles and from about 5% to about 82% refractory particles by weight.

[0200] In either embodiment, the corrosion resistant particles are provided in a size range of 25 microns and smaller. Preferably the particles are 10 microns and smaller in size. The particles may be substantially equiaxed (spherical) or non-equiaxed (flake). If a high particle density is desired, the particles should be provided in at least two sizes. In such a circumstance, the average particle size preferably should differ by a factor of about 7 to 10. The size difference between the particles allows the smaller particles to fill the areas between the larger particles. This is particularly evident when the particles are substantially equiaxed. Thus, if high packing density is required and the size of particles is about 5 microns, then a second size range of particles should also be included wherein the particles are 0.5 microns and smaller.

[0201] Regardless of the intended method of application, the coating composition mixture is thoroughly agitated. Agitation can be accomplished by any convenient method for about 0.1-5 hours. Preferably, mixing is accomplished for a period of about 0.1-0.5 hours. This is an important step, for it is not only important that the particles be substantially uniformly and thoroughly distributed throughout the slurry, it is also important that the solution completely “wet” or coat the particles. Depending on the particles, it is believed that the surfaces of the particles may become hydrolyzed, which, as will be discussed, will allow bonding with the silica-based material.

[0202] In a preferred embodiment, the viscosity is adjusted so that the slurry can be applied by spraying. In this circumstance, the slurry is continuously agitated by placing a stirrer into mix until it is ready for application. Even as the slurry is sprayed, the slurry can be pneumatically agitated by using a stirring pot in the spray application. The slurry is preferably applied by using a Paasch spray gun having an adjustable orifice. The orifice size must be larger than the largest particles in the slurry. The slurry is preferably sprayed at a pressure of about 20-60 psi. The coating composition is applied to a preselected thickness, with a larger orifice being selected when a thicker coating is desired.

[0203] After the coating composition mixture is applied to the surface of the component, the applied composition is allowed to dry. Drying is accomplished in two steps. In the first step, drying is accomplished to remove unbound solvent. This is accomplished after application of the composition to the surface of the component by raising the temperature to less than 212° F. (100° C.). It will be recognized by those skilled in the art that higher humidities and/or lower temperatures will also provide drying, but will require longer times to achieve the necessary drying. When the coating is applied to a thickness of about 0.001″ (one mil) or greater, heating must be accomplished at a rate of no greater than about 2-10° F./min. to prevent blistering. Next, the coating is heated to a temperature of about 400° F. to drive off water and initiate a cure of the material.

[0204] After the initial cure, the coated substrate is fired to an elevated temperature to convert the coating into a glass or a glassy ceramic with substantially uniformly dispersed particles throughout. Preferably, firing is accomplished at a temperature at or above the expected operating temperature of the component, but not less than about 700° F. The coating may be fired up to about 2100° F. The higher the firing temperature, the higher percentage of the glass that is converted from glass to ceramic.

[0205] A graded coating may be achieved by applying additional layers over the first layer and subsequent layers, each subsequent layer applied after drying to remove unbound water and optionally fired to cure the layer. Of course, each layer is adjusted to have a different loading of particles and/or particles of different compositions, the loading and type of particles determining the CTE of the layer. If the graded coating is applied in this manner and without firing between layers, there may be some mixing of the loadings at the interface between layers. On curing, there will be strong chemical bonding between the layers, and except for the loadings, the "layer" aspect will disappear and the coating will act as a uniform coating. Since the CTE can be tailored with thickness, the resulting stresses and strains can be designed as a function of coating thickness. This permits, if desired, the use of a highly corrosion resistant, low CTE particle such as alumina, in a coating layer, which layer can be applied over a higher CTE coating layer, such as a layer that includes CoNiCrAlY particles without negatively affecting the adhesion of the coating to the substrate. Optionally, additional mixed layers of the coating composition may be applied as overcoats to transition from high CTE at the substrate to lower CTE at the surface of the coated article. Where additional layers are applied over the first layer and subsequent layers, each subsequent layer is applied after drying to remove bound water and curing, and optionally firing. Again, each layer is adjusted to have a different loading of particles, the loading of particles determining the CTE of the layer. For example, adjusting the ratio of refractory particles such as alumina and non-refractory particles such as CoNiCrAlY will alter the CTE of a given coating layer. When the graded coating is applied in this manner, there is substantially no mixing of the loadings at the interface between layers and the layers are distinct. Where one or more layers are optionally fired before applying an overcoat layer, it is expected that the resulting discrete layers will not be crosslinked, and will therefore permit the multilayer coating to incur spalling in an outer layer without incurring damage to undercoat layers.

EXAMPLES AND TESTING

[0206] The formulations shown herein are exemplary, and are not limiting. The goal is to provide a coating composition that can be applied to a substrate, and that will cure to form a thin silica-ceramic matrix coating that will protect the substrate from corrosion without using a formulation that contains hexavalent chromium.

[0207] The first example, shown in Table 1, shows an exemplary coating in accordance with embodiments having only refractory corrosion resistant particulates. The second example, shown in Table 2, demonstrates a coating composition in accordance with embodiments that includes both refractory and non-refractory corrosion resistant particles to yield a coating having a CTE that is compatible with the underlying substrate, thereby further reducing the tendency of the coating to spall when subjected to the operational heating and cooling cycles of a gas turbine engine.

TABLE-US-00001 TABLE 1 Example of First Embodiment (Refractory only) Sprayed Materials for Corrosion Protection by Weight Percent Material Mod #3-4 SM8 33-40 A17SG 45-55 SR350 25-35 Ethyl 95%-Isop 5% 40-60

TABLE-US-00002 TABLE 2 Example of Second Embodiment (Refractory plus Non-refractory) Sprayed Materials for Corrosion Protection by Weight Percent Material Mod #8-18 Fe-125 60-80 SM8 15-25 SR350 25-35 Ethyl 95%-Isop 5% 40-60

[0208] In embodiments, the corrosion resistant base coating may be applied over a turbine engine component with a temporary organic coating disposed thereon. The corrosion resistant coating comprises refractory oxide particles, alumina, MAI, MAlX, MCr, MCrX, MCrAlX particles or a combination thereof, uniformly distributed in an inorganic matrix forming material. The particles provide the coating with corrosion resistance, while inorganic material in the inorganic matrix forming material is the binder during application and forms the matrix after curing. On curing, the inorganic material forms the matrix containing the corrosion resistant particle that, upon firing, forms the base coating. The system further comprises a temporary organic coating applied on at least a portion of the base coating.

[0209] Binder components for formation of the matrix of the base coating suitable for use herein typically comprise a phosphate, chromate, silicone or silica binder, with or without other binder materials. These binders can be in the form of an acid or more typically the respective compounds/compositions. For example, phosphate binders may be present as orthophosphates, pyrophosphates, etc. The compounds/compositions can be monobasic, dibasic, tribasic or any combination thereof. Binder components can comprise one or more metal compounds. For example, metal phosphates may include aluminum phosphates, magnesium phosphates, chromium phosphates, zinc phosphates, iron phosphates, lithium phosphates, calcium phosphates, etc, or any combination thereof. Typically, the phosphate-containing binder component comprises an aluminum phosphate, a magnesium phosphate, a chromium phosphate, or a combination thereof. The binder component can optionally comprise other binder materials, including one or more chromates, molybdates, etc. See, for example, U.S. Pat. No. 3,248,249 (Collins, Jr.), issued Apr. 26, 1966; U.S. Pat. No. 3,248,251 (Allen), issued Apr. 26, 1966; U.S. Pat. No. 4,889,858 (Mosser), issued Dec. 26, 1989; U.S. Pat. No. 4,975,330 (Mosser), issued Dec. 4, 1990, the relevant portions of which are incorporated by reference, in their entirety. The phosphate-containing binder component can also be substantially free of other binder materials, e.g., a substantially chromate free phosphate-containing binder component. See, for example, U.S. Pat. No. 6,368,394 (Hughes et al.), issued Apr. 9, 2002 (substantially chromate free phosphate binder component), the relevant portion of which is incorporated by reference, in its entirety.

[0210] As used herein, the term “CTE” refers to the coefficient of thermal expansion of a material, and is referred to herein in units of 10.sup.−6/° F. For example, alumina, which has a coefficient of thermal expansion of about 4 to 5×10.sup.−6/° F. at about 1200° F. (649° C.), is referred to herein as having a CTE of about 4 to 5.

[0211] FIG. **16** is a cross-sectional view depicting a portion of the turbine section of a gas turbine engine along the centerline of the engine. The turbine section **930** is a two stage turbine, although any number of stages may be employed depending on the turbine design. The present disclosure is not limited by the number of stages in the turbine. Turbine disks **932** are mounted on a shaft (not shown) extending through a bore in disks **932** along the centerline of the engine. A first stage blade **938** is attached to first stage disk **936**, while second stage blade **942** is attached to second stage disk **940**. A vane **1310** extends from a casing **1320**. The inner surface of casing **1320** forms a liner **1330** for the hot gases of combustion, which flow in the gas flow path. The first stage blade **938**, the second stage blade **942** and the vane **1310** extend into the hot gas flow path. The vane is stationary and serves to direct the hot gas flow while blades **938**, **942** mounted on disks **936**, **940** rotate as the hot gases impinge on them, extracting energy to operate the engine.

[0212] Sealing elements **934**, a forward seal **944**, an aft seal **946**, an interstage seal **948**, a stage 1 aft blade retainer **950** and a stage 2 aft blade retainer **952**, serve to seal and complete the compressor air cooling circuits to the turbine blades and nozzles. These seals are attached to the disks and rotate with the disks. Interstage seal **948** is positioned inboard of vane **1310** and between the first stage disk **936** and the second stage disk **940**. Also shown are optional blade retainers **950**, **952** which lock the blades to the disks. The design of such retainers will vary dependent on engine design, with some engine designs not requiring them.

[0213] These disks, seals and blade retainers are heated to the temperatures of the cooling circuit air they direct. In addition, the parts closest to the combustion path are also heated by conductive heat transfer from the combustion path parts. For example, the rims of the turbine disks are conductively heated by the turbine blades. Contaminants in the cooling air, as previously discussed, deposit on the surfaces of the disks, seals and retainers that include the cooling cavities that provide the air that is the source of contamination at these elevated temperatures. Thus, the present disclosure can provide protection to any of these surfaces that are subject to corrosion due to deposition or accumulation of the cooling air contaminants.

[0214] FIG. **18** is a perspective view of a typical gas turbine engine disk **982** such as disk **936** or **940** of FIG. **16**, which is typically made of a superalloy material, such as one of the superalloy materials previously discussed. The disk **982** includes a hub **974** along typically the engine centerline that includes a bore through which a shaft (not shown) extends. The disk includes dovetail slots **986** along the disk outer periphery into which the turbine blades are inserted. A web section **978** of the disk **982** extends between the outer periphery, where the dovetail slots are located, and the hub. While the presently disclosed coatings may be utilized anywhere along disk **982**, including the dovetail slots, it finds particular use along the surfaces of web section **978** and the dovetail slots **986**, which unlike the hub **974**, is directly exposed to cooling air.

[0215] FIG. **17** depicts, in cross-section, the coating **964** in its simplest form, deposited on an engine component. Corrosion resistant coating **964** is deposited on the surface **962** of substrate **960**. The substrate **960** may be a turbine engine disk such as first stage disk **936** or second stage disk **940**. The substrate **960** may be a typical surface such as web section **978** of a turbine disk **982**. Substrate **960** may comprise superalloy based on nickel, cobalt, iron and combinations thereof, has deposited thereon a coating **964**. Optionally, an undercoating may be provided (not shown), such as a MCrAlX coating, for example a NiCrAlY or a CoNiCrAlY, an aluminide such as NiAl or noble metal-modified aluminide such as (Pt,Ni)Al. As discussed previously, coating **964** can be cured as a single layer of graded coating and surface **966** is exposed to the cooling air forming the environment for the surface. Alternatively coating **964** may be of substantially uniform composition. If the coating is to be graded, then additional layers are applied over coating layer **964**, the first layer being applied over outer surface **966** and additional layers being applied over subsequent outer coating layers.

[0216] Prior to forming the corrosion resistant coating **964** on the surface **962** of metal substrate **960**, metal surface **962** is typically pretreated mechanically, chemically or both to make the surface more receptive for coating **964**. Suitable pretreatment methods include grit blasting, with or without masking of surfaces that are not to be subjected to grit blasting (see U.S. Pat. No. 5,723,078 to Nagaraj et al, issued Mar. 3, 1998, especially col. 4, lines 46-66, which is incorporated by reference), micromachining, laser etching (see U.S. Pat. No. 5,723,078 to Nagaraj et al, issued Mar. 3, 1998, especially col. 4, line 67 to col. 5, line 3 and 14-17, which is incorporated by reference), treatment with chemical etchants such as those containing hydrochloric acid, hydrofluoric acid, nitric acid, ammonium bifluorides and mixtures thereof, (see, for example, U.S. Pat. No. 5,723,078 to Nagaraj et al, issued Mar. 3, 1998, especially col. 5, lines 3-10; U.S. Pat. No. 4,563,239 to Adinolfi et al, issued Jan. 7, 1986, especially col. 2, line 67 to col. 3, line 7; U.S. Pat. No. 4,353,780 to Fishter et al, issued Oct. 12, 1982, especially col. 1, lines 50-58; and U.S. Pat. No. 4,411,730 to Fishter et al, issued Oct. 25, 1983, especially col. 2, lines 40-51, all of which are incorporated by reference), treatment with water under pressure (i.e., water jet treatment), with or without loading with abrasive particles, as well as various combinations of these methods. Typically, the surface **962** of metal substrate **960** is pretreated by grit blasting where surface **962** is subjected to the abrasive action of silicon carbide particles, steel particles, alumina particles or other types of abrasive particles. These particles used in grit blasting are typically alumina particles and typically have a particle size of from about 600 to about 35 mesh (from about 25 to about 500 micrometers), more typically from about 360 to about 35 mesh (from about 35 to about 500 micrometers).

[0217] After the surface preparation is completed, the base coating is applied. A suitable base coating for use in embodiments disclosed herein is disclosed in U.S. patent application Ser. No. 11/011,695, entitled CORROSION RESISTANT COATING COMPOSITION, COATED

TURBINE COMPONENT AND METHOD FOR COATING SAME, filed on Dec. 15, 2004, assigned to the assignee of the present application and incorporated herein by reference, in its entirety. Another suitable base coating includes U.S. patent application Ser. No. 11/311,720, entitled STRAIN TOLERANT CORROSION PROTECTING COATING AND SPRAY METHOD OF APPLICATION, filed on Dec. 19, 2005, assigned to the assignee of the present application and incorporated herein by reference, in its entirety. Another suitable base coating includes U.S. patent application Ser. No. 11/293,448, entitled CORROSION INHIBITING CERAMIC COATING AND METHOD OF APPLICATION, filed on Dec. 2, 2005, assigned to the assignee of the present application and incorporated herein by reference, in its entirety. Other suitable base coatings include aqueous corrosion resistant coating compositions comprising phosphate/chromate binder systems and aluminum/alumina particles. See, for example, U.S. Pat. No. 4,606,967 (Mosser), issued Aug. 19, 1986 (spheroidal aluminum particles) and U.S. Pat. No. 4,544,408 (Mosser et al.), issued Oct. 1, 1985 (dispersible hydrated alumina particles), which are herein incorporated by reference in their entirely.

[0218] When additional layers of coating are to be applied over surface **1466** in order to obtain a graded, multi-layer coating, it is generally not necessary to prepare coating surface **1466** prior to application of additional layers.

[0219] While the above provide examples of preferred usages for the coating of the present disclosure, the disclosure is not so limited and may be used in any application where corrosion of base metal is evident. The base coating of the present disclosure is preferably applied as a coating in thicknesses from about 0.0001″ (0.1 mils) to about 0.005″ (5 mils), and preferably in thicknesses from about 0.0005″ (0.5 mils) to about 2.5″. The coating can be applied to such thicknesses as a single layer, or can be applied as a plurality of distinct layers to achieve an overall thickness in these ranges.

[0220] A preferred embodiment of the present disclosure includes a coating applied to form a matrix having corrosion resistant particles substantially uniformly dispersed throughout. The corrosion resistance is provided by particles of refractory oxide, alumina, MCrAlX or combinations of these particles. The matrix can be formulated in any one of a number of ways. However, a water-based system utilizes a matrix forming component. The viscosity can be adjusted by adding water or allowing water to evaporate in order to obtain the desired viscosity.

[0221] Matrix forming materials preferably comprise a phosphate binder, with or without other binder materials. Although phosphate binders are preferred, other binder materials, such as colloidal silica and silicone may be utilized. The phosphate binders can be in the form of phosphoric acid or more typically the respective phosphate compounds/compositions, including orthophosphates, pyrophosphates, and other phosphate compounds. These phosphate compounds/compositions can be monobasic, dibasic, tribasic or any combination thereof. The phosphate-containing binder component can optionally comprise other binder material, including one or more chromates, molybdates, or similar binder material.

[0222] The corrosion resistant particles are added to the matrix forming solution. Specifically, the corrosion resistant particles are added to the binder material, such as solutions or dispersions containing phosphate, colloidal silica or silicone. These particles may include refractory oxide particles that can impart corrosion resistance to a coating, such as alumina, yttrium oxide (Y.sub.2O.sub.5), zirconium oxide (Zr.sub.2O.sub.3), titanium oxide (TiO.sub.2) and combinations thereof. Other suitable materials include ceramics with a CTE greater than that of alumina and that are relatively inert or non-reactive. While oxides of other metals may be used, such as tungsten, chromium and rhenium, these are less preferred as they are not deemed to be as environmentally friendly as the preferred refractory oxides. Alternatively, MCrAlX, MCr, MAI, MCrX or MAlX particles may be added to the solution, either alone or in combination with the refractory oxide particles to provide a layer with a predetermined CTE.

[0223] For example, the particles may be added to a colloidal silica dispersion so that the particles comprise, by weight, from 5-60% of the total solution, up to 15% surfactant and the balance being one of the LP colloidal silica dispersion. Thus, for example, for an LP30 colloidal solution, when particles are added to about 30% by weight, about 21% by weight comprises silica solids, up to 10% is a surfactant and the balance of the solution, about 49% comprises water. The particles are provided in a size range of 25 microns and smaller. The particles may be substantially equiaxed (spherical) or non-equiaxed (flake). Preferably the particles are 10 microns and smaller in size. If a high particle density is desired, the particles should be provided in at least two sizes. In such a circumstance, the average particle size preferably should differ by a factor of about 10. The size difference between the particles allows the smaller particles to fill the areas between the larger particles. This is particularly evident when the particles are substantially equiaxed. Thus, if high packing density is required and the size of particles is about 5 microns, then a second size range of particles should also be included wherein the particles are 0.5 microns and smaller. The packing density of the particles will have some effect on the CTE of the layer.

[0224] The composition for formation of the base coating according to the present disclosure may include a material identified as LBK-51F, which comprises, in weight percent, about 10% Triton™-X surfactant, about 22.5% LUCALOX® alumina, the balance, about 67.5%, being colloidal silica. A second suitable composition includes LBK-51G, comprising, in weight percent, about 2% surfactant, about 24.5% alumina, −325 mesh that is acid washed and the balance, about 73.5% colloidal silica. Both compositions may be applied by spraying. LUCALOX® is a registered trademark of General Electric Company, Fairfield, Connecticut, and LUCALOX® alumina is a polycrystal alumina available from the same company. The Triton™-X series surfactants are nonionic octylphenol ethoxylate-type surfactants recognized for their wetting and detergency available from Dow Chemical.

[0225] After the corrosion resistant particles have been added to the solution to form a slurry, the slurry viscosity is adjusted by either adding liquid or adding additional particles to the mixture. Surfactants and dispersants may be added to the slurry when required. The viscosity should be adjusted, if required, to be consistent with the intended method of application. If the slurry is to be sprayed, the viscosity should be adjusted to be very low, whereas if the slurry is to be applied as a gel, using, for example, a doctor blade to adjust the thickness, then liquid should be removed so that the slurry does not flow readily. Even more liquid should be removed if the slurry is to be formed into a tape. In the last two examples, the final viscosity adjustment may be made after mixing is complete. Regardless of the intended method of application, the mixture is thoroughly agitated. Agitation can be accomplished by any convenient method. Depending on the particles, it is believed that the surfaces of the particles become hydrolyzed, which may allow bonding with the hydrolyzed silica-based material.

[0226] In a preferred embodiment, the viscosity is adjusted so that the slurry can be applied by spraying. In this circumstance, the slurry is continuously agitated by placing it on a ball mill until it is ready for application. Even as the slurry is sprayed, the slurry can be pneumatically agitated by using a pot on a spray gun. The slurry is applied preferably using a Bosch spray gun having an adjustable orifice. The orifice size must be larger than the largest particles in the slurry. The slurry is sprayed at a pressure of about 20-60 psi. The coating is applied to a preselected thickness, with a larger orifice being selected when a thicker coating is desired.

[0227] After the mixture is applied to the surface of the component, it is allowed to dry. Drying is accomplished in two steps. In the first step, drying is accomplished to remove unbound water. This is accomplished after application of the mixture, either as a spray coating, a gel, or a paste to the surface of the component, preferably by increasing the temperature to a temperature sufficient to provide increased drying, but below 212° F. (100° C.), or by reducing the humidity to below 30% relative humidity. It will be recognized by those skilled in the art that higher humidity and/or lower temperatures will also provide drying, but will require longer times to achieve the necessary drying. When the coating is applied to a thickness of 0.001″ (one mil) or greater, heating is preferably accomplished at a rate of no greater than about 5-15° F./min. to prevent blistering. Next, the coating is heated to a temperature of about 400° F. or higher to drive off unbound water and cure the material to form base coating. Thereafter, the temporary organic coating is applied to the surface and permitted to dry and/or cure. The application of the temporary organic coating is preferably sufficiently thick to provide the surface with a glossy finish. Alternatively, heat may be applied to the surface to decrease the time required to dry and/or cure. High temperature curing or firing is generally not required. The application of the temporary organic coating may be achieved utilizing any known coating technique, including, but not limited to brushing, rolling, spraying and dipping.

[0228] In another embodiment, a graded or layered coating may be achieved by applying additional layers over the first layer and subsequent layers, each subsequent layer applied after drying to remove unbound water. Of course, each layer is adjusted to have a different loading of particles and or particles of different compositions, the loading and type of particles determining the CTE of the layer. If the graded coating is applied in this manner, there may be some mixing of the loadings at the interface between layers. On curing, there will be strong bonding between the layers, and except for the loadings and/or types, the coating will act as a uniform coating. Since the CTE can be tailored with thickness, the resulting stresses and strains can be designed as a function of coating thickness. This permits, if desired, the use of a highly corrosion resistant, low CTE particle such as alumina, in a coating layer, which layer can be applied over a less corrosion resistant, higher CTE coating layer, such as a layer that includes CoNiCrAlY particles without negatively affecting the adhesion of the coating to the substrate. The temporary organic coating **1470** is applied to the graded or layered coating and provides sealing and abrasion and/or impact resistant to the surface. The organic coating seals the top layer of the coating system providing a graded or layered coating that is well bonded and is capable of withstanding large temperature fluctuations. The organic coating may or may not infiltrate the base coat. The organic coating volatilizes and/or burns away when exposed to the operating temperature of the gas turbine engine leaving the base coating substantially unaffected in the base coating's strain tolerant properties during operation.

[0229] The coating of the present disclosure is comprised of a matrix having substantially uniformly dispersed particles of R and R1 within the matrix. While R and R1 may be any of the corrosion resistant refractory oxide, alumina, MCr, MCrX, MAI, MAlX or MCrAlX particles, here R represents CoNiCrAlY particles and R1 represents zirconia particles. The CoNiCrAlY particles are depicted as surrounded by the matrix. The dispersed zirconia particles and the CoNiCrAlY particles provide the coating with corrosion resistance. The particle composition or combination of particles of various compositions are selected to provide a sufficiently similar CTE between the base coating and the substrate, while preventing spalling. If the required level of corrosion resistance and required CTE could not be achieved, then intermediate layers having intermediate CTE's could be applied over the substrate and below the layer having the required corrosion resistance.

[0230] Overlying base coating is temporary organic coating. Temporary organic coating is an organic sealant composition. Organic sealant compositions suitable for use with the disclosure preferably includes an organic material including resins such as, but not limited to, latex acrylics, solvent acrylics, polyurethanes, polysulfides and any other organic material capable of sealing pores of the base coating and capable of being removed at elevated temperatures, such as those temperatures present during operation of the gas turbine engine. The organic material may include fillers, pigments or other resin additives known in the art. Preferred organic materials include unpigmented acrylic paint, unpigmented polyurethane paint and unpigmented latex paint.

[0231] The temporary organic coating may be applied by any suitable application process, including, but not limited to, brushing, rolling, spraying or dipping. A preferred application method is spraying. The organic material is preferably permitted to dry and/or cure at room temperature, but may also be heated to facilitate the drying and/or curing of the temporary organic coating. The thickness of the dried and/or cured coating is preferably sufficient to provide a glossy finish. The coating thickness of the temporary organic coating is from about 0.0001 inches to about 0.0050 inches. The application is preferably substantially 100% coverage wherein water beading on the surface, as opposed to infiltrating the base coating, indicates sufficient coating thickness. The entire engine component may be coated with the temporary organic coating, wherein no masking is required and the heating step is optional. The elimination of masking during application of the temporary organic coating will optionally allow coverage on a variety of components including seal teeth, contact interfaces, and similar components subject to cooling air. As the temporary organic coating will volatilize and/or burn away during operation, the seal teeth, contact interfaces, and similar will return to an uncoated condition as if they were never coated with the temporary organic coating. The application and drying of the temporary organic coating on a gas turbine engine component preferably requires less than about 6 hours, more preferably less than about 4 hours. The temporary organic coating may be removed by exposure to elevated temperatures. Preferably the temporary organic coating is removed during gas turbine engine operation. For example, the temporary organic coating may be removed by exposure to temperatures of greater than about 500° C. The removal preferably occurs at temperature at which gas turbine engines operate. Removal includes volatilization of the temporary organic coating. While the organic material may volatilize, the removal may take place by other mechanisms, such as burning or delamination resulting from the exposure to the temperatures greater than about 500° C.

[0232] The temporary organic coating is applied after base coating and preferably before further processing of the component occurs. The temporary organic coating provides protection for the base coating and the substrate against damage that may occur during manufacture of the component and assembly of the gas turbine engine. The temporary organic coating provides resistance against impact damage and/or abrasion. In addition, the temporary organic coating seals the pores of the base coating, providing protection against infiltration of contaminants such as oil or grease, which may stain or damage the base coating.

[0233] 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.

[0234] Further aspects are provided by the subject matter of the following clauses:

[0235] A gas turbine engine comprising: a turbomachine comprising a compressor section, a combustion section, and a turbine section arranged in serial flow order, the compressor section having a high pressure compressor defining a high pressure compressor exit area (A.sub.HPCExit) in square inches; wherein the gas turbine engine defines a redline exhaust gas temperature (EGT) in degrees Celsius, a total sea level static thrust output (Fn.sub.Total) in pounds, and a corrected specific thrust, wherein the corrected specific thrust is greater than or equal to 42 and less than or equal to 90, the corrected specific determined as follows:

Fn.sub.Total×EGT/(A.sub.HPCExit.sup.2×1000).

[0236] The gas turbine engine of the preceding clauses wherein the corrected specific thrust is from 42 to 90, such as from 45 to 80, such as from 50 to 80.

[0237] The gas turbine engine of the preceding clauses, wherein the EGT is greater than 1000 degrees Celsius and less than 1300 degrees Celsius.

[0238] The gas turbine engine of any preceding clause, wherein the EGT is greater than 1100 degree Celsius and less than 1250 degrees Celsius.

[0239] The gas turbine engine of any preceding clause, wherein the EGT is greater than 1150 degree Celsius and less than 1250 degrees Celsius.

[0240] The gas turbine engine of any preceding clause, wherein the EGT is greater than 1000 degree Celsius and less than 1300 degrees Celsius, and wherein the corrected specific thrust is greater than or equal to 45.
[0241] The gas turbine engine of any preceding clause, wherein the EGT is greater than 1000 degree Celsius and less than 1300 degrees Celsius, and wherein the corrected specific thrust is greater than or equal to 50.
[0242] The gas turbine engine of any preceding clause, wherein the turbine section comprises a high pressure turbine having a first stage of high pressure turbine rotor blades, and wherein the gas turbine engine further comprises: a cooled cooling air system in fluid communication with the first stage of high pressure turbine rotor blades.
[0243] The gas turbine engine of one or more of the preceding clause, wherein the cooled cooling air system is further in fluid communication with the high pressure compressor for receiving an airflow from the high pressure compressor, and wherein the cooled cooling air system further comprises a heat exchanger in thermal communication with the airflow for cooling the airflow.
[0244] The gas turbine engine of any preceding clause, wherein when the gas turbine engine is operated at a takeoff power level, the cooled cooling air system is configured to provide a temperature reduction of a cooling airflow equal to at least 15% of the EGT and up to 45% of the EGT.
[0245] The gas turbine engine of any preceding clause, wherein when the gas turbine engine is operated at a takeoff power level, the cooled cooling air system is configured to receive between 2.5% and 35% of an airflow through a working gas flowpath of the turbomachine at an inlet to a compressor of the compressor section.
[0246] The gas turbine engine of any preceding clause, further comprising a primary fan driven by the turbomachine.
[0247] The gas turbine engine of any preceding clause, further comprising an inlet duct downstream of the primary fan and upstream of the compressor section of the turbomachine; and a secondary fan located within the inlet duct.
[0248] The gas turbine engine of any preceding clause, wherein the gas turbine engine defines a bypass passage over the turbomachine, and wherein the gas turbine engine defines a third stream extending from a location downstream of the secondary fan to the bypass passage.
[0249] The gas turbine engine of any preceding clause, wherein the secondary fan is a single stage secondary fan.
[0250] A method of operating a gas turbine engine, comprising: operating the gas turbine engine at a takeoff power level, the gas turbine engine having a turbomachine with a high pressure compressor defining a high pressure compressor exit area (A.sub.HPCExit) in square inches, the gas turbine engine defining a redline exhaust gas temperature (EGT) in degrees Celsius, a total sea level static thrust output (Fn.sub.Total) in pounds, and a corrected specific thrust; wherein the corrected specific thrust is greater than or equal to 42 and less than or equal to 90, the corrected specific determined as follows: Fn.sub.Total×EGT/(A.sub.HPCExit.sup.2×1000).
[0251] The method of any preceding clause, wherein the EGT defined by the gas turbine engine is greater than 1000 degree Celsius and less than 1300 degrees Celsius.
[0252] The method of any preceding clause, wherein the EGT defined by the gas turbine engine is greater than 1100 degree Celsius and less than 1300 degrees Celsius.
[0253] The method of any preceding clause, wherein the EGT defined by the gas turbine engine is greater than 1000 degree Celsius and less than 1300 degrees Celsius, and wherein the corrected specific thrust defined by the gas turbine engine is greater than or equal to 45.
[0254] The method of any preceding clause, wherein operating the gas turbine engine at the takeoff power level further comprises reducing a temperature of a cooling airflow provided to a high pressure turbine of the gas turbine engine with a cooled cooling air system.
[0255] The method of any preceding clause, wherein reducing the temperature of the cooling

airflow provided to the high pressure turbine of the gas turbine engine with the cooled cooling air system comprises providing a temperature reduction of the cooling airflow equal to at least 15% of the EGT and up to 45% of the EGT.
[0256] The gas turbine engine of any preceding clause, wherein the cooled cooling air systems includes a thermal bus cooled cooling air system (see, e.g., FIGS. **4** and **5**).
[0257] The gas turbine engine of any preceding clause, wherein the cooled cooling air systems includes a dedicated heat exchanger cooled cooling air system (i.e., a cooled cooling air system including a heat exchanger dedicated to the cooled cooling air system).
[0258] The gas turbine engine of any preceding clause, wherein the cooled cooling air systems includes a bypass heat exchanger cooled cooling air system having a heat sink heat exchanger thermally coupled to an airflow through a bypass passage (see, e.g., FIG. **9**).
[0259] The gas turbine engine of any preceding clause, wherein the cooled cooling air systems includes an air-to-air cooled cooling air system (a cooled cooling air system having a heat sink heat exchanger configured to transfer heat to an airflow; see, e.g., FIG. **9**).
[0260] The gas turbine engine of any preceding clause, wherein the cooled cooling air systems includes an oil-to-air cooled cooling air system (a cooled cooling air system having a heat sink heat exchanger configured to transfer heat to an oil flow).
[0261] The gas turbine engine of any preceding clause, wherein the cooled cooling air systems includes a fuel-to-air cooled cooling air system (a cooled cooling air system having a heat sink heat exchanger configured to transfer heat to a fuel flow, such as a Jet A fuel flow, a liquid hydrogen or hydrogen gas fuel flow, etc.; see, e.g., FIG. **4**).
[0262] The gas turbine engine of any preceding clause, wherein the cooled cooling air systems is configured to receive the cooling air from a downstream end of a high pressure compressor.
[0263] The gas turbine engine of any preceding clause, wherein the cooled cooling air systems is configured to receive the cooling air from an upstream end of the high pressure compressor.
[0264] The gas turbine engine of any preceding clause, wherein the cooled cooling air systems is configured to receive the cooling air from a downstream end of a low pressure compressor.
[0265] The gas turbine engine of any preceding clause, wherein the cooled cooling air systems is configured to receive the cooling air from an upstream end of the low pressure compressor.
[0266] The gas turbine engine of any preceding clause, wherein the cooled cooling air systems is configured to receive the cooling air from a location between compressors.
[0267] The gas turbine engine of any preceding clause, wherein the cooled cooling air systems is configured to receive the cooling air from a bypass passage.
[0268] A corrosion resistant coating, comprising: a matrix; corrosion resistant particles dispersed throughout the matrix; and a glass-forming additive, wherein the glass-forming additive and one or more materials in the matrix form a glassy-phase when heat treated.
[0269] The coating of any preceding clause, wherein the corrosion resistant particles comprise Al.sub.2O.sub.3, Y.sub.2O.sub.3, ZrO.sub.2, TiO.sub.2, or a combination thereof.
[0270] The coating of any preceding clause, wherein the corrosion resistant particles comprise MAI particles, MAlX particles, MCr particles, MCrX particles, MCrAlX particles, or a combination thereof, where M is an element selected from nickel, iron, cobalt, or a combination thereof and X is an element selected from La, Ta, Re, Y, Zr, Hf, Si, B, C, or a combination thereof.
[0271] The coating of any preceding clause, wherein the matrix comprises a silicon-based matrix, a silicone-based matrix, or a combination thereof.
[0272] The coating of any preceding clause, wherein the glass-forming additive comprises one or more metals or oxides of iron, one or more metals or oxides of aluminum, one or more metals or oxides of boron, one or more metals or oxides of nickel, or a combination thereof.
[0273] The coating of any preceding clause, further comprising a nucleating agent.
[0274] The coating of any preceding clause, wherein the coating is substantially free of hexavalent chromium.

[0275] The coating of any preceding clause, further comprising a silicone-based sealant layer, wherein the silicone-based sealant layer is substantially free of the corrosion resistant particles.
[0276] The coating of any preceding clause, wherein the corrosion resistant particles comprise a plurality of small particles having a median particle size of less than 1 micron, a plurality of medium particles having a particle size of between 2 microns and 8 microns, and a plurality of large particles having a particle size of between 9 microns and 60 microns, wherein the plurality of small particles is present in an amount of from about 10 volume % to about 30 volume %, the plurality of medium particles is present in an amount of from about 30 volume % to about 50 volume %, and the plurality of large particles is present in an amount of from about 30 volume % to about 50 volume %.
[0277] A corrosion resistant gas turbine engine component, comprising a turbine engine component having a corrosion resistant coating disposed thereon, the corrosion resistant coating comprising: a matrix; corrosion resistant particles dispersed throughout the matrix; and a glass-forming additive, wherein the glass-forming additive reacts with one or more materials in the matrix to form a glassy-phase when heat treated.
[0278] The component of any preceding clause, wherein the corrosion resistant particles comprise Al_2O_3, Y_2O_3, ZrO_2, TiO_2, or a combination thereof.
[0279] The component of any preceding clause, wherein the corrosion resistant particles comprise Mal particles, MAlX particles, MCr particles, MCrX particles, MCrAlX particles, or a combination thereof, where M is an element selected from nickel, iron, cobalt or a combination thereof and X is an element selected from La, Ta, Re, Y, Zr, Hf, Si, B, C, or a combination thereof.
[0280] The component of any preceding clause, wherein the matrix comprises a silicon-based matrix, a silicone-based matrix, or a combination thereof.
[0281] The component of any preceding clause, wherein the glass-forming additive comprises one or more metals or oxides of iron, one or more metals or oxides of aluminum, one or more metals or oxides of boron, one or more metals or oxides of nickel, or a combination thereof.
[0282] The component of any preceding clause, wherein the corrosion resistant coating further comprises a nucleating agent.
[0283] The component of any preceding clause, wherein the coating is substantially free of hexavalent chromium.
[0284] The component of any preceding clause, wherein the corrosion resistant particles comprise a plurality of small particles having a median particle size of less than 1 micron, a plurality of medium particles having a particle size of between 2 microns and 8 microns, and a plurality of large particles having a particle size of between 9 microns and 60 microns, wherein the plurality of small particles is present in an amount of from about 10 volume % to about 30 volume %, the plurality of medium particles is present in an amount of from about 30 volume % to about 50 volume %, and the plurality of large particles is present in an amount of from about 30 volume % to about 50 volume %.
[0285] The component of any preceding clause, wherein the gas turbine engine component comprises a nickel-based alloy, a cobalt-based alloy, or a combination thereof.
[0286] The component of any preceding clause, wherein the gas turbine engine component comprises a compressor spool, turbine disk, seal, or shaft.
[0287] A method for coating a metal component comprising: disposing a coating composition on a surface of the metal component, the coating composition comprising: a matrix material; corrosion resistant particles; and a glass-forming additive; and heat treating the coating composition to form a coated metal component, wherein the glass-forming additive and one or more materials in the matrix material react during heat treating to form a glassy-phase.

Claims

1. A gas turbine engine comprising: a turbomachine comprising a compressor section, a combustion section, and a turbine section arranged in serial flow order, the compressor section having a high pressure compressor defining a high pressure compressor exit area (A.sub.HPCExit) in square inches; and a coated component within the turbomachine, wherein the coated component includes a substrate and a corrosion resistant coating thereon, wherein the gas turbine engine defines a redline exhaust gas temperature (EGT) in degrees Celsius, a total sea level static thrust output (Fn.sub.Total) in pounds, and a corrected specific thrust, wherein the corrected specific thrust is greater than or equal to 42 and less than or equal to 90, the corrected specific thrust determined as follows: Fn.sub.Total×EGT/(AH.sub.PCExit.sup.2×1000).
2. The component of claim 1, wherein the corrosion resistant coating comprises a matrix, corrosion resistant particles dispersed throughout the matrix, and a glass-forming additive, wherein the glass-forming additive and one or more materials in the matrix form a glassy-phase when heat treated.
3. The component of claim 2, wherein the corrosion resistant particles comprise Al.sub.2O.sub.3, Y.sub.2O.sub.3, ZrO.sub.2, TiO.sub.2, or a combination thereof.
4. The component of claim 2, wherein the corrosion resistant particles comprise Mal particles, MAlX particles, MCr particles, MCrX particles, MCrAlX particles, or a combination thereof, where M is an element selected from nickel, iron, cobalt or a combination thereof and X is an element selected from La, Ta, Re, Y, Zr, Hf, Si, B, C, or a combination thereof.
5. The component of claim 2, wherein the matrix comprises a silicon-based matrix, a silicone-based matrix, or a combination thereof.
6. The component of claim 2, wherein the glass-forming additive comprises one or more metals or oxides of iron, one or more metals or oxides of aluminum, one or more metals or oxides of boron, one or more metals or oxides of nickel, or a combination thereof.
7. The component of claim 2, wherein the corrosion resistant coating further comprises a nucleating agent.
8. The component of claim 1, wherein the corrosion resistant particles comprise a plurality of small particles having a median particle size of less than 1 micron, a plurality of medium particles having a particle size of between 2 microns and 8 microns, and a plurality of large particles having a particle size of between 9 microns and 60 microns, wherein the plurality of small particles is present in an amount of from about 10 volume % to about 30 volume %, the plurality of medium particles is present in an amount of from about 30 volume % to about 50 volume %, and the plurality of large particles is present in an amount of from about 30 volume % to about 50 volume %.
9. The component of claim 1, wherein the coating is substantially free of hexavalent chromium.
10. The component of claim 1, wherein the gas turbine engine component comprises a nickel-based alloy, a cobalt-based alloy, or a combination thereof.
11. The component of claim 1, wherein the gas turbine engine component comprises a compressor spool, turbine disk, seal, or shaft.
12. The gas turbine engine of claim 1, wherein the EGT is greater than 1000 degrees Celsius and less than 1300 degrees Celsius.
13. The gas turbine engine of claim 1, wherein the EGT is greater than 1100 degree Celsius and less than 1250 degrees Celsius.
14. The gas turbine engine of claim 1, wherein the EGT is greater than 1150 degree Celsius and less than 1250 degrees Celsius.
15. The gas turbine engine of claim 1, wherein the EGT is greater than 1000 degree Celsius and less than 1300 degrees Celsius, and wherein the corrected specific thrust is greater than or equal to 45.
16. The gas turbine engine of claim 1, wherein the EGT is greater than 1000 degree Celsius and less than 1300 degrees Celsius, and wherein the corrected specific thrust is greater than or equal to

50.

17. The gas turbine engine of claim 1, wherein the turbine section comprises a high pressure turbine having a first stage of high pressure turbine rotor blades, and wherein the gas turbine engine further comprises: a cooled cooling air system in fluid communication with the first stage of high pressure turbine rotor blades.

18. The gas turbine engine of claim 1, further comprising a primary fan driven by the turbomachine.

19. A method of operating a gas turbine engine, comprising: operating the gas turbine engine at a takeoff power level, the gas turbine engine having a turbomachine with a high pressure compressor defining a high pressure compressor exit area (A.sub.HPCExit) in square inches, the gas turbine engine defining a redline exhaust gas temperature (EGT) in degrees Celsius, a total sea level static thrust output (Fn.sub.Total) in pounds, and a corrected specific thrust; wherein the gas turbine engine comprises a coated component within the turbomachine, wherein the coated component includes a substrate and a corrosion resistant coating thereon, and wherein the corrected specific thrust is greater than or equal to 42 and less than or equal to 90, the corrected specific thrust determined as follows: Fn.sub.Total×EGT/(A.sub.HPCExit.sup.2×1000).

20. The method of claim 19, wherein the corrosion resistant coating comprises a matrix, corrosion resistant particles dispersed throughout the matrix, and a glass-forming additive, wherein the glass-forming additive and one or more materials in the matrix form a glassy-phase when heat treated.