

(CSP) in Newtons squared times degrees Celsius over meters squared, wherein the corrected specific power is determined as follows:

$$\left(\frac{T_{OUT}}{\sqrt{A_{DTExit}}} \right)^2 * \frac{EGT}{A_{HPCExit}} * 10^{-11};$$

wherein CSP is greater than $0.0001194 \times EGT^2 - 0.103 \times EGT + 22.14$ and less than $0.0003294 \times EGT^2 - 0.306 \times EGT + 77.91$; and wherein EGT is greater than 525 degrees Celsius and less than 1250 degrees Celsius.

20 Claims, 28 Drawing Sheets

Related U.S. Application Data

application No. 18/650,586, filed on Apr. 30, 2024, now Pat. No. 12,196,131, which is a continuation-in-part of application No. 18/500,517, filed on Nov. 2, 2023, now Pat. No. 12,078,107, which is a continuation of application No. 18/481,515, filed on Oct. 5, 2023, which is a continuation-in-part of application No. 17/978,629, filed on Nov. 1, 2022, now abandoned.

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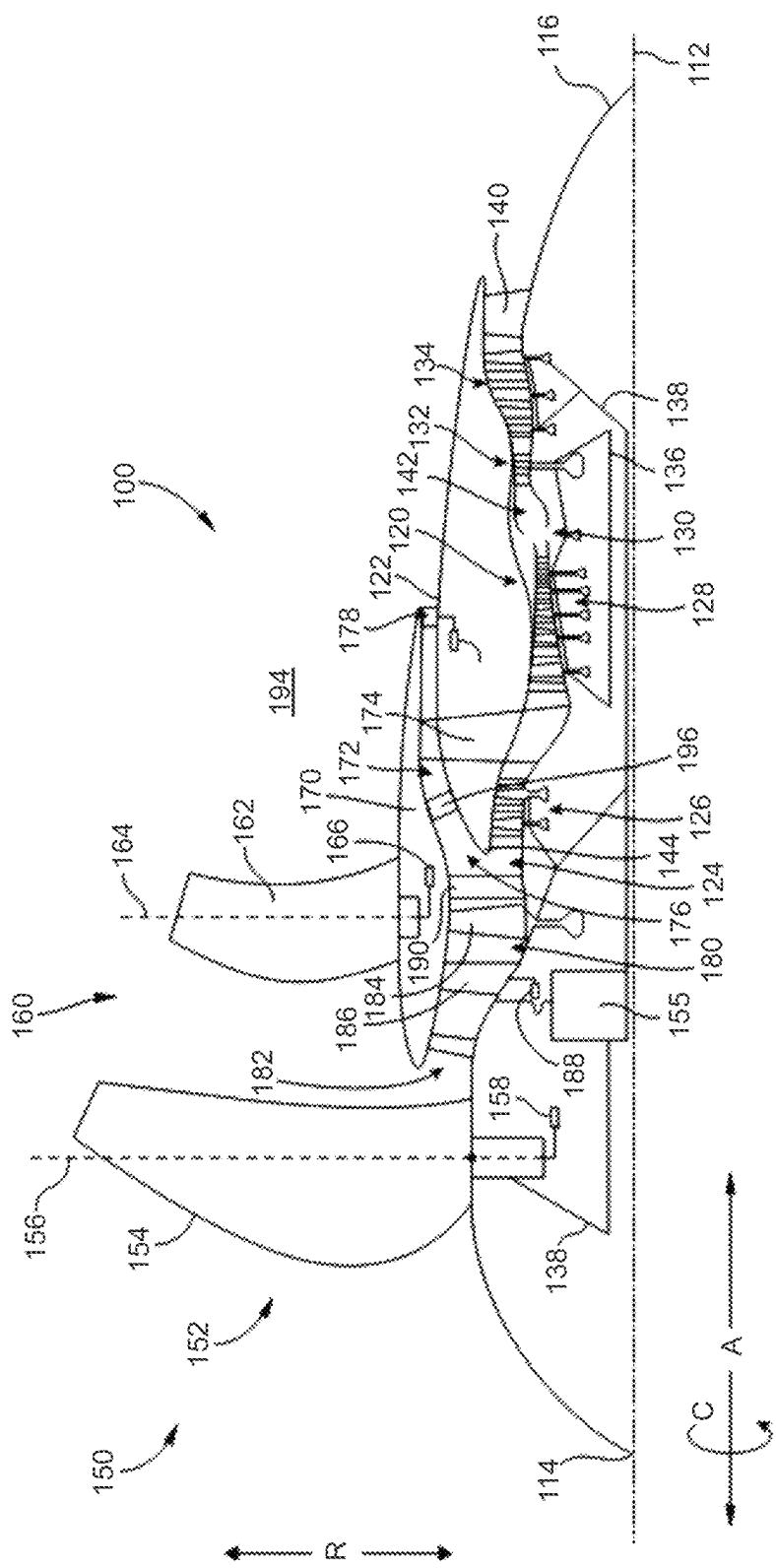


FIG. 1

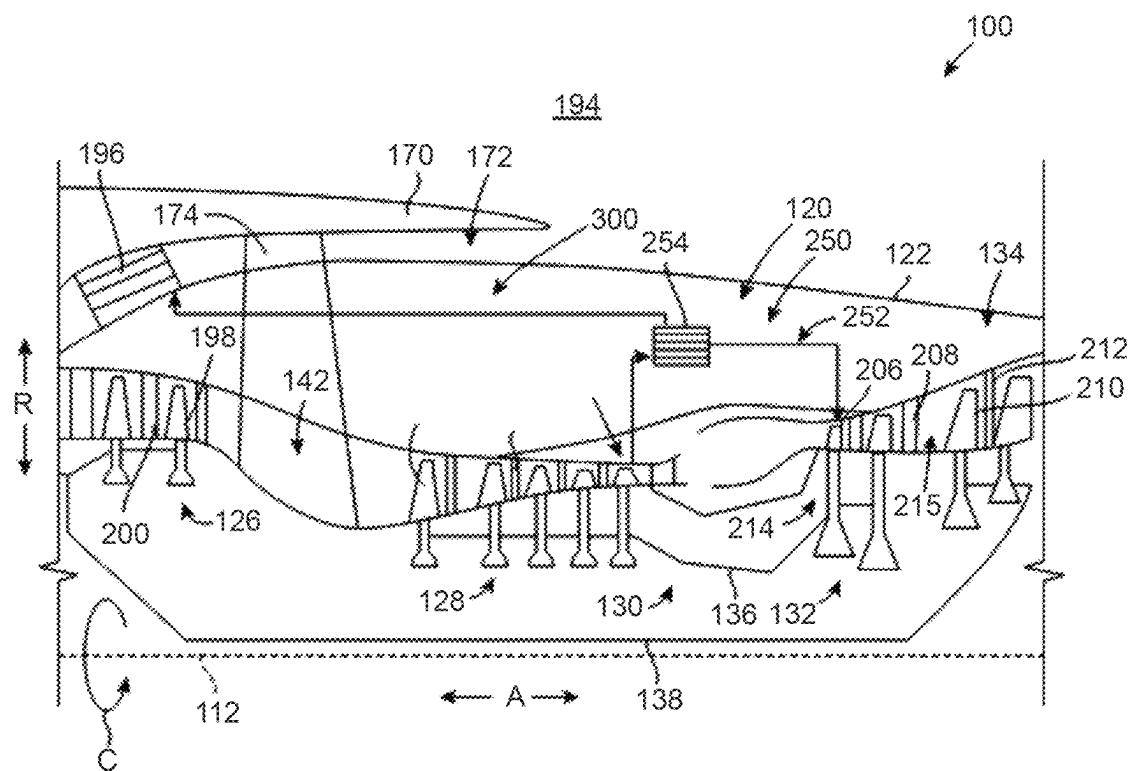


FIG. 2

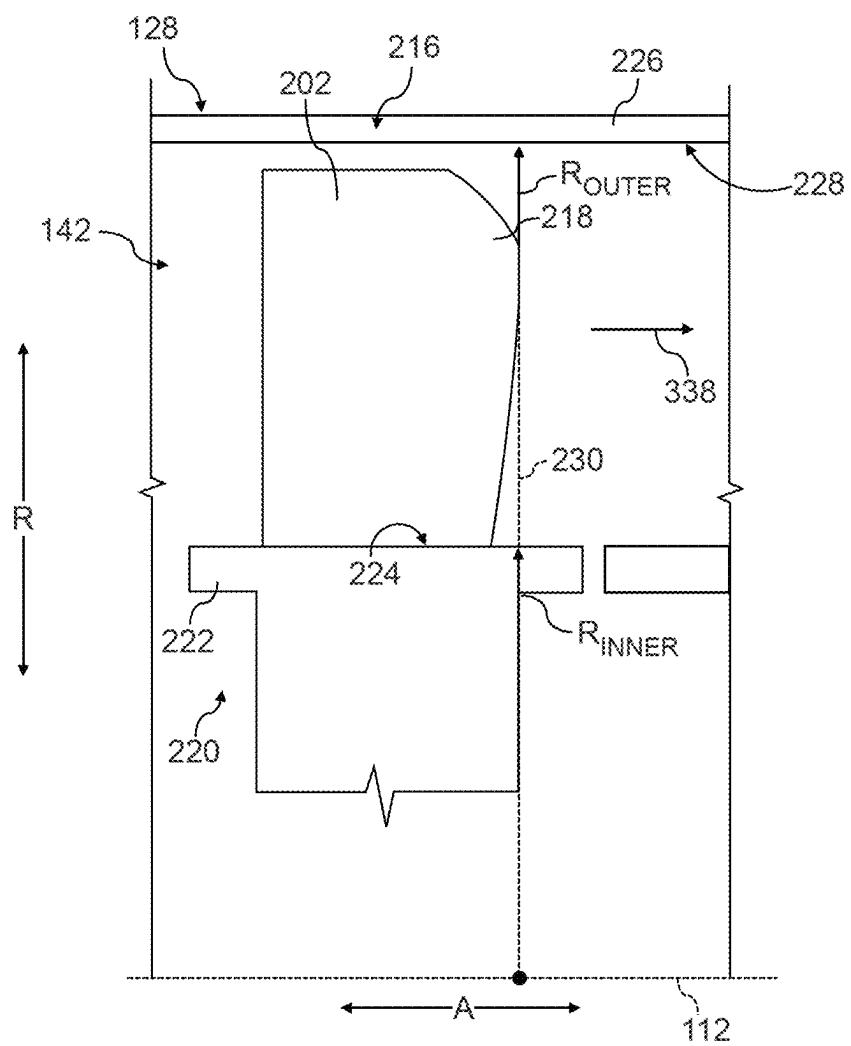


FIG. 3

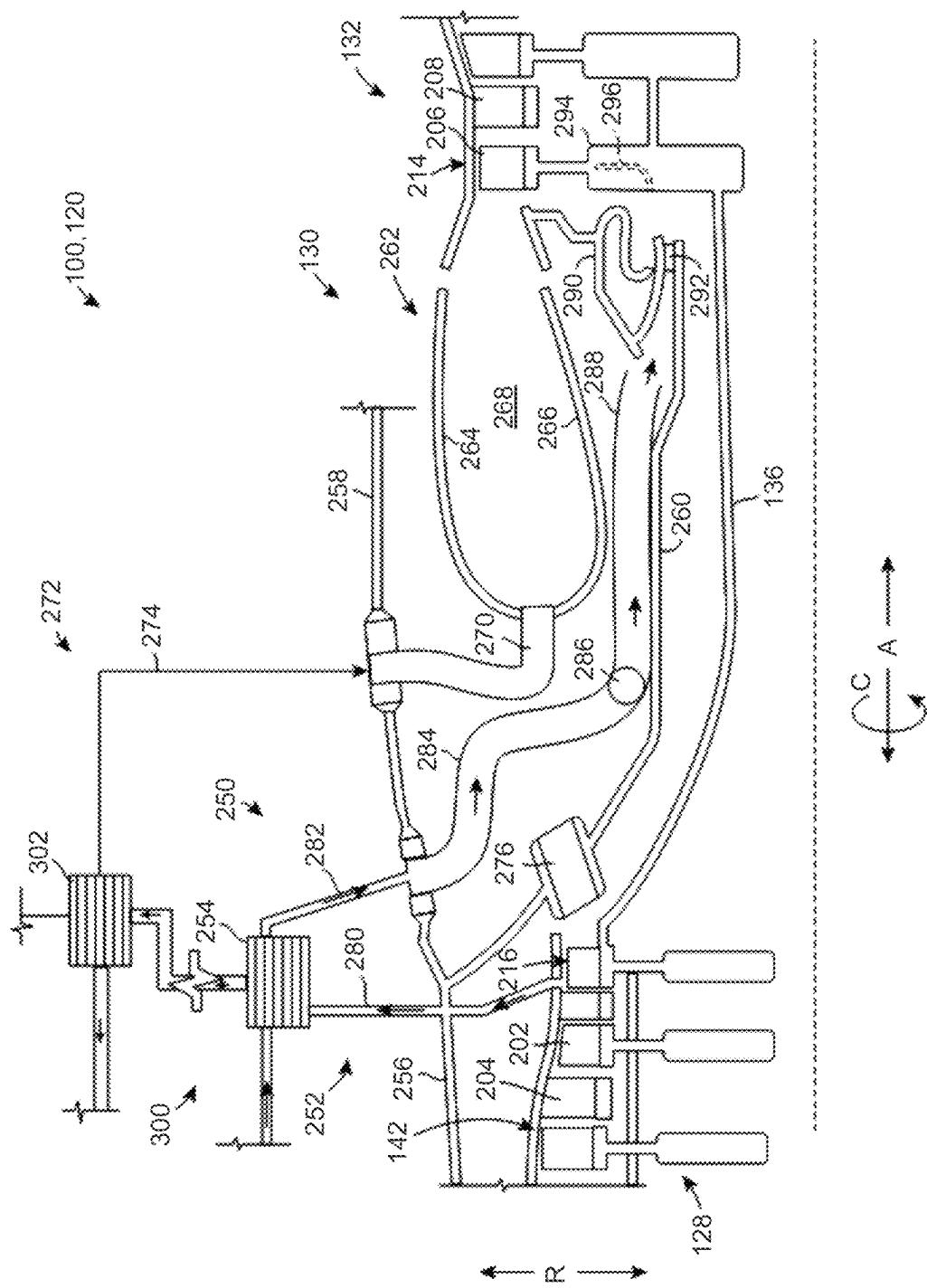


FIG. 4

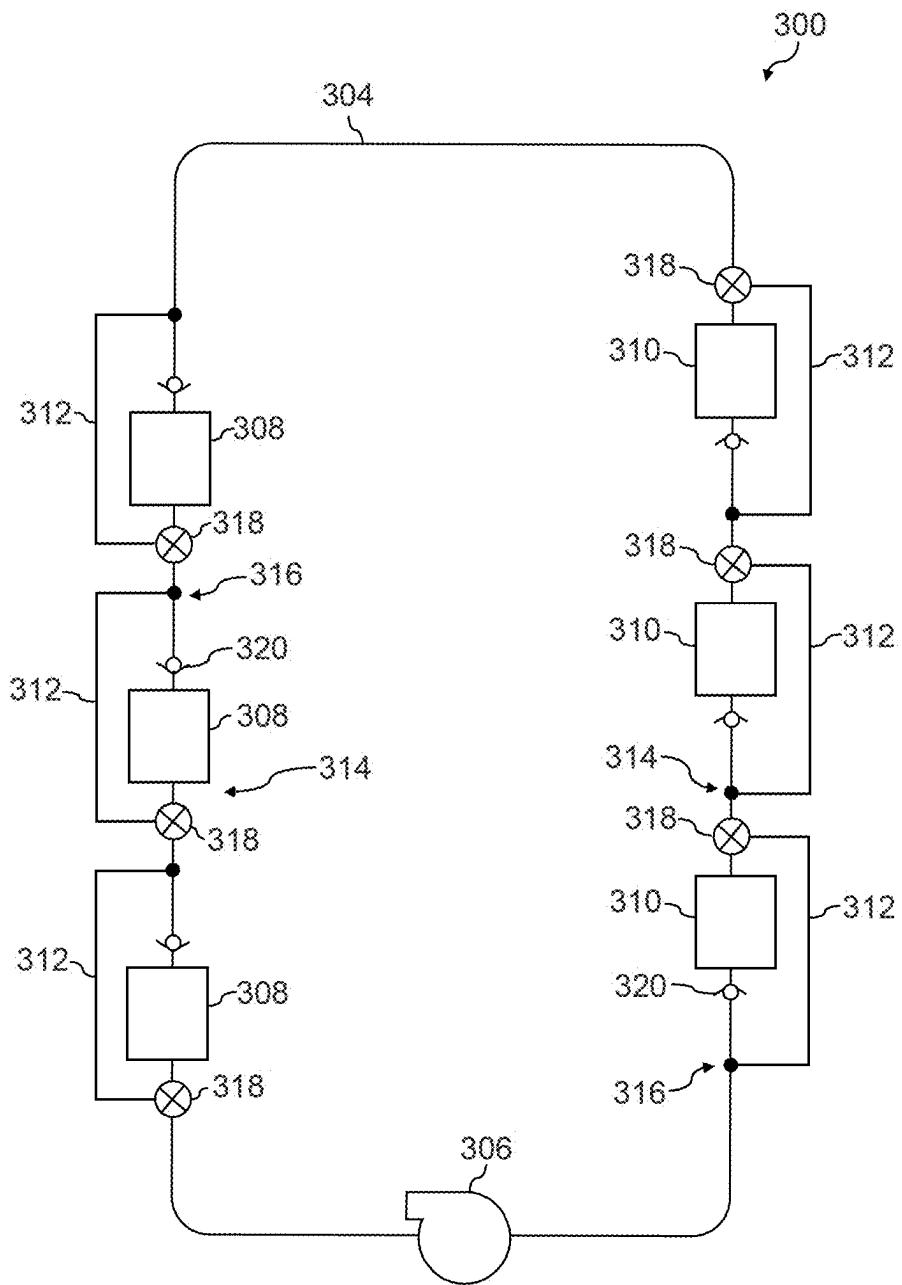
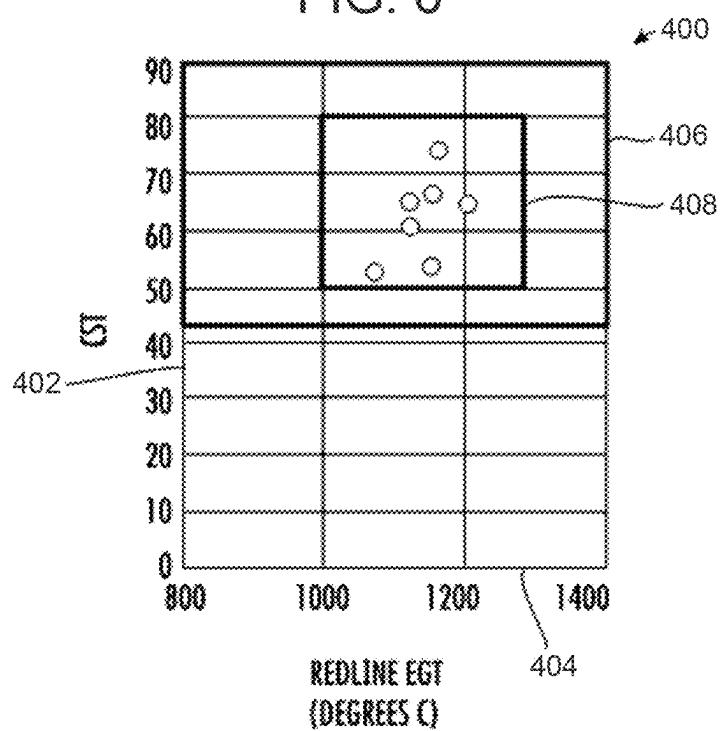


FIG. 5

HPC EXIT AREA	REDLINE EGT	SIS THRUST	GTF
29.1	1153	39262	53.6
27.1	1205	39258	64.5
28.4	1071	39292	52.2
26.1	1123	39288	64.6
24.7	1160	39285	74.5
26.2	1155	39262	66.2
27.0	1207	39257	64.9
27.0	1125	39289	60.6
24.6	1161	39285	75.0
29.1	1152	39262	53.3
28.4	1071	39292	52.1
27.0	1123	39287	60.3

FIG. 6



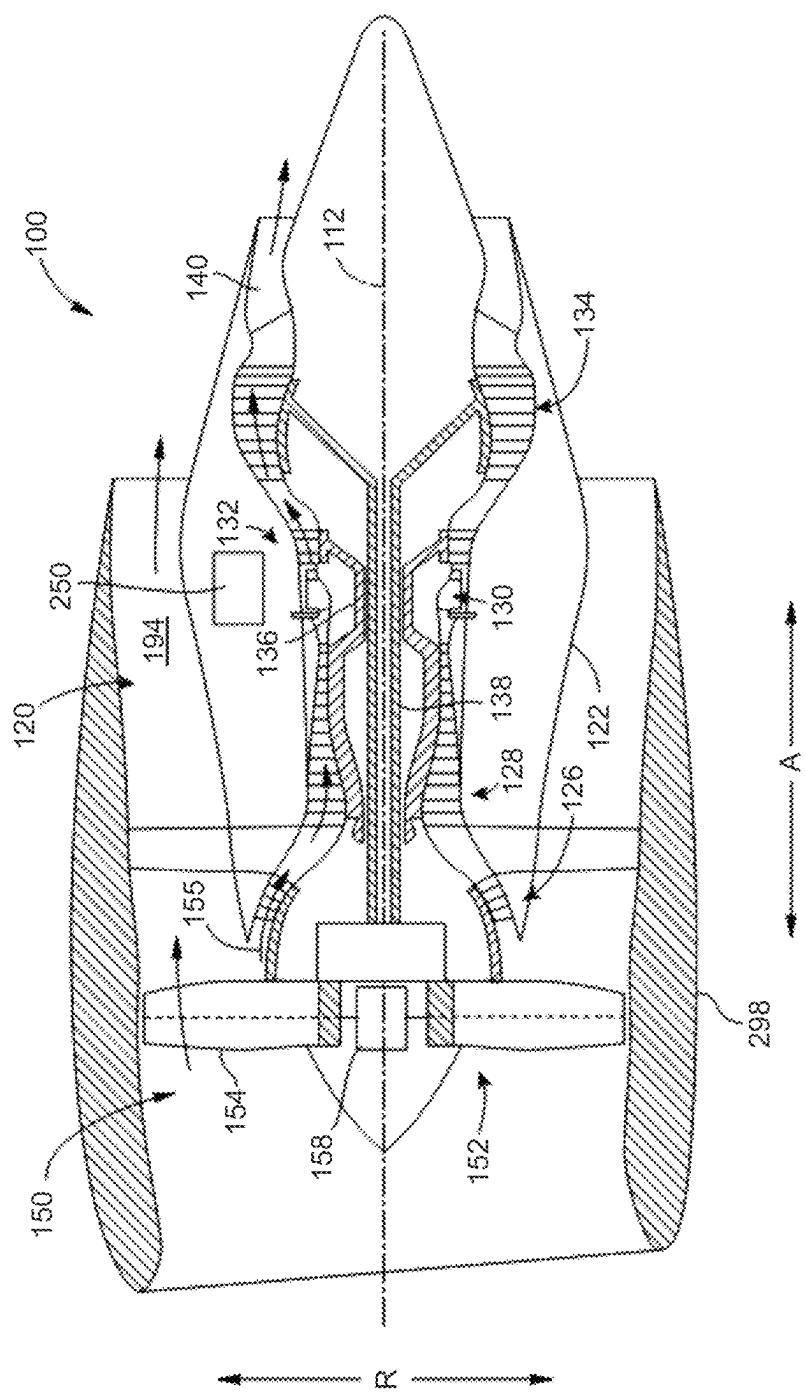


FIG. 8

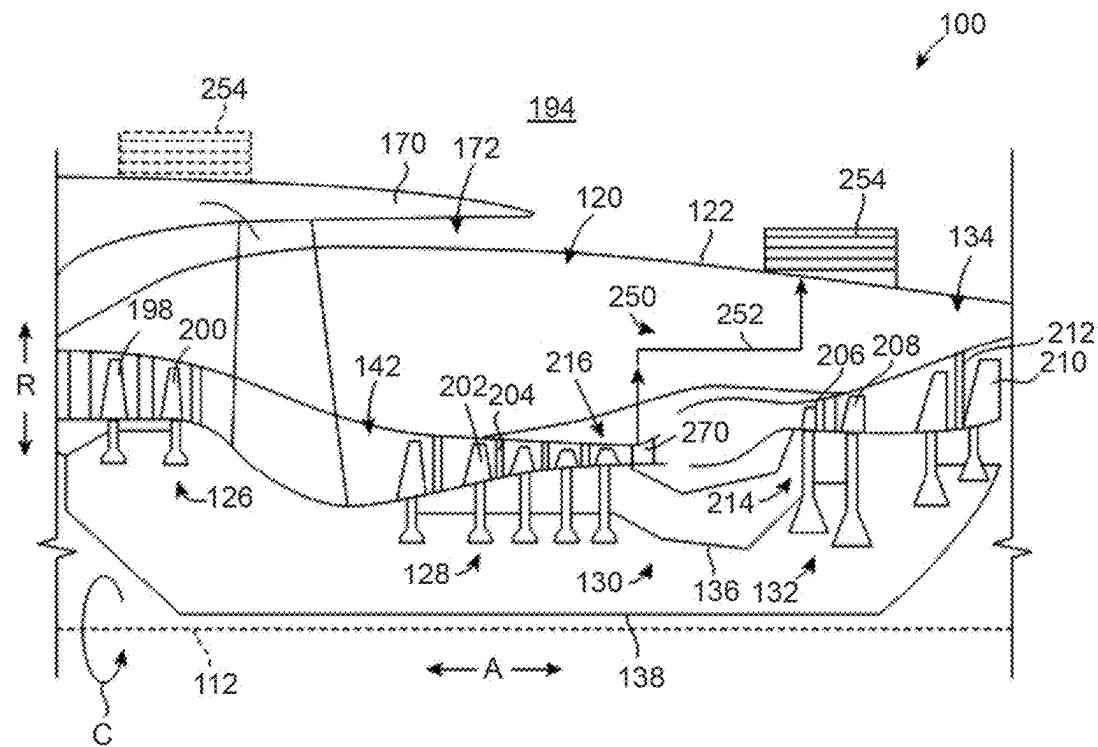


FIG. 9

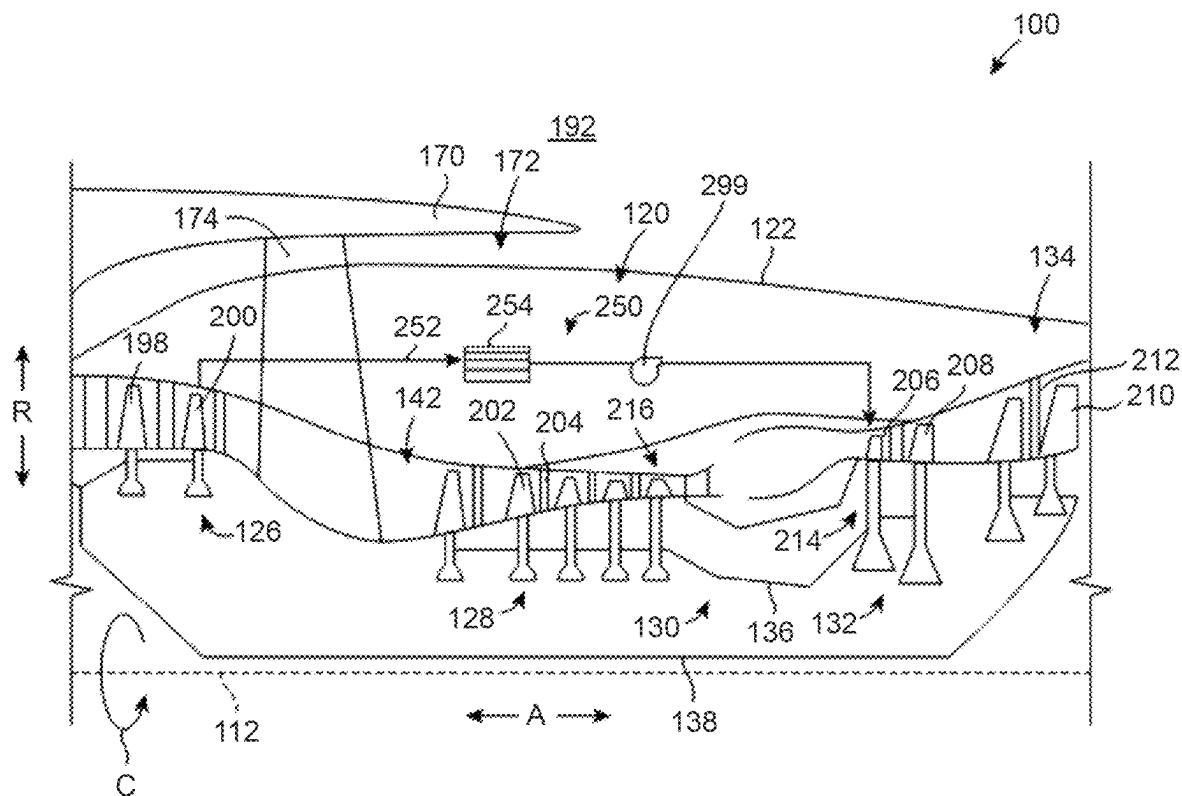


FIG. 10

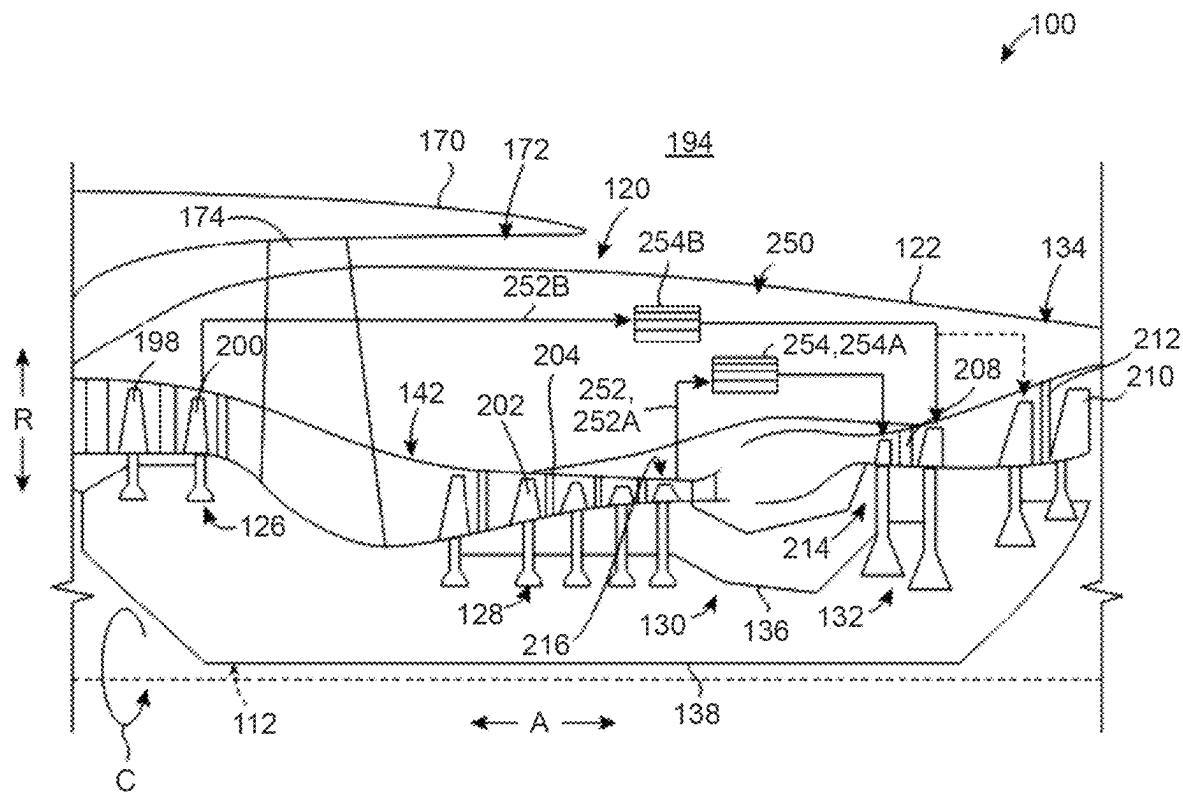


FIG. 11

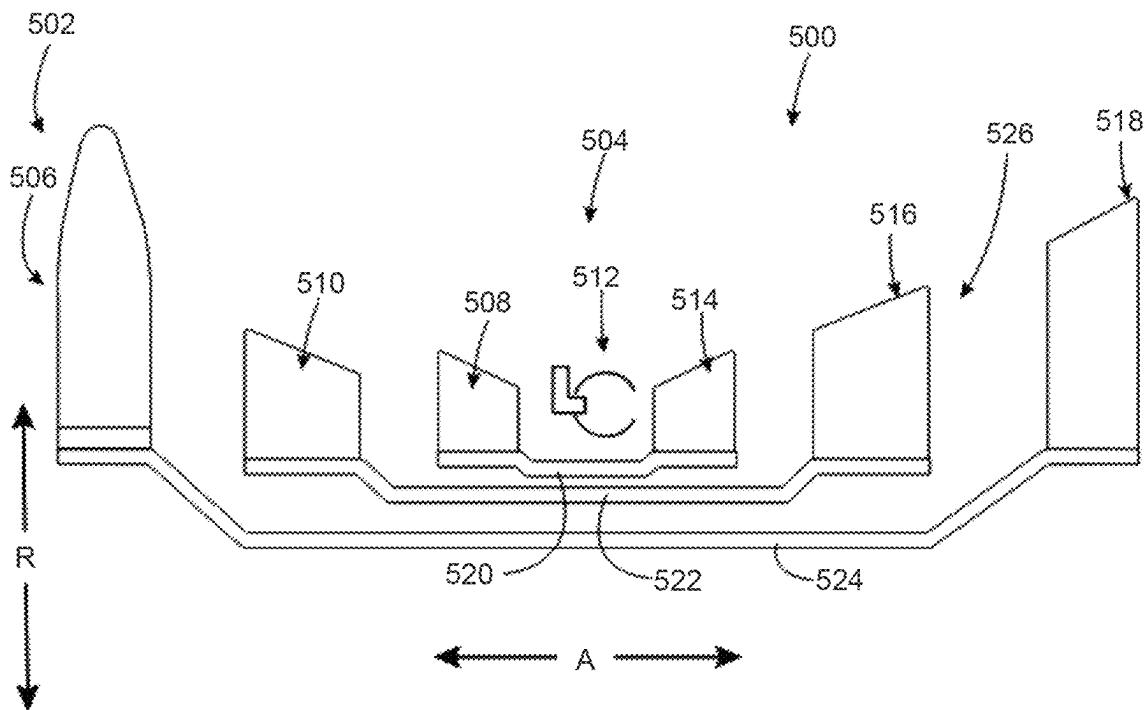
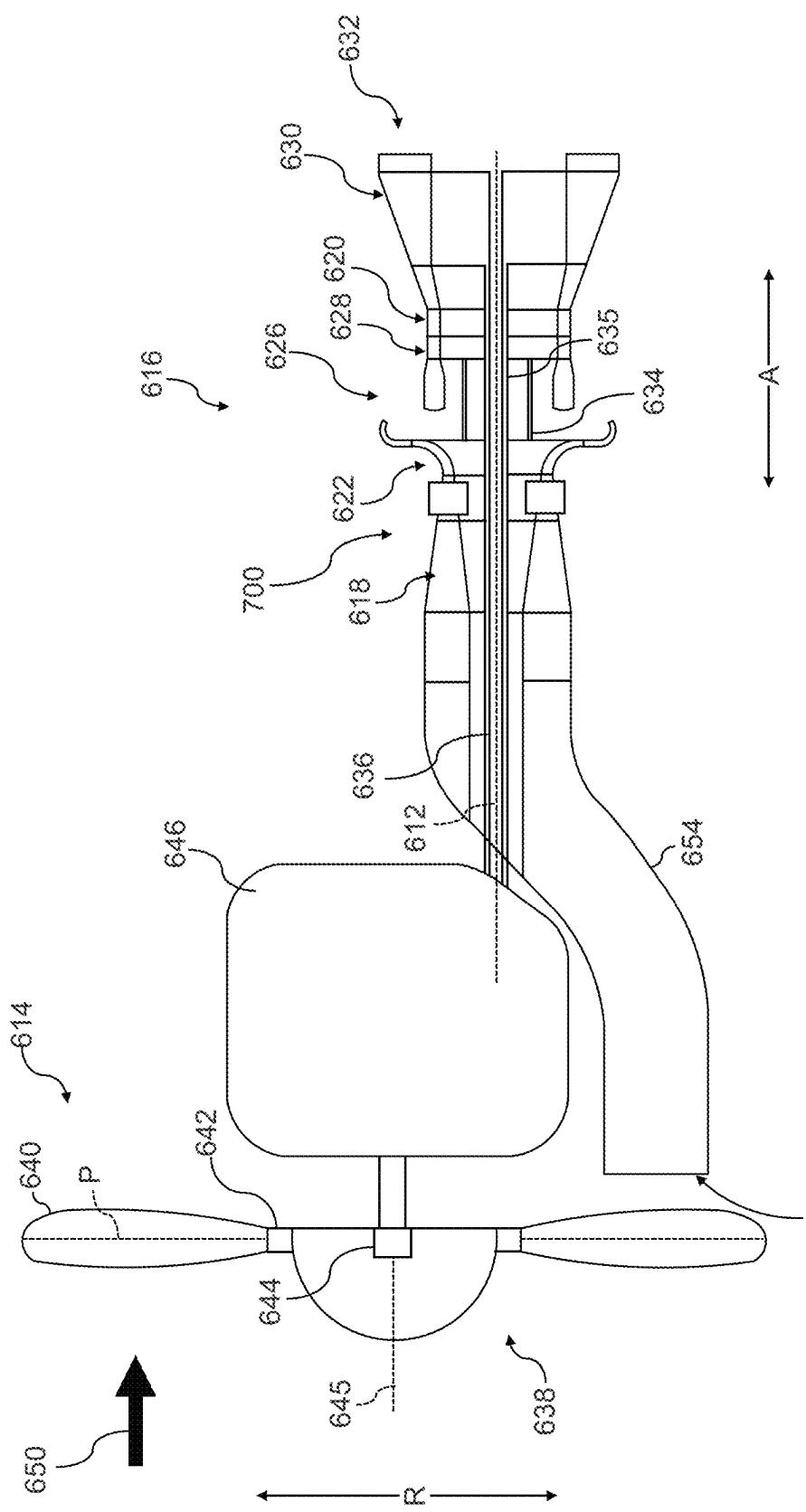


FIG. 12



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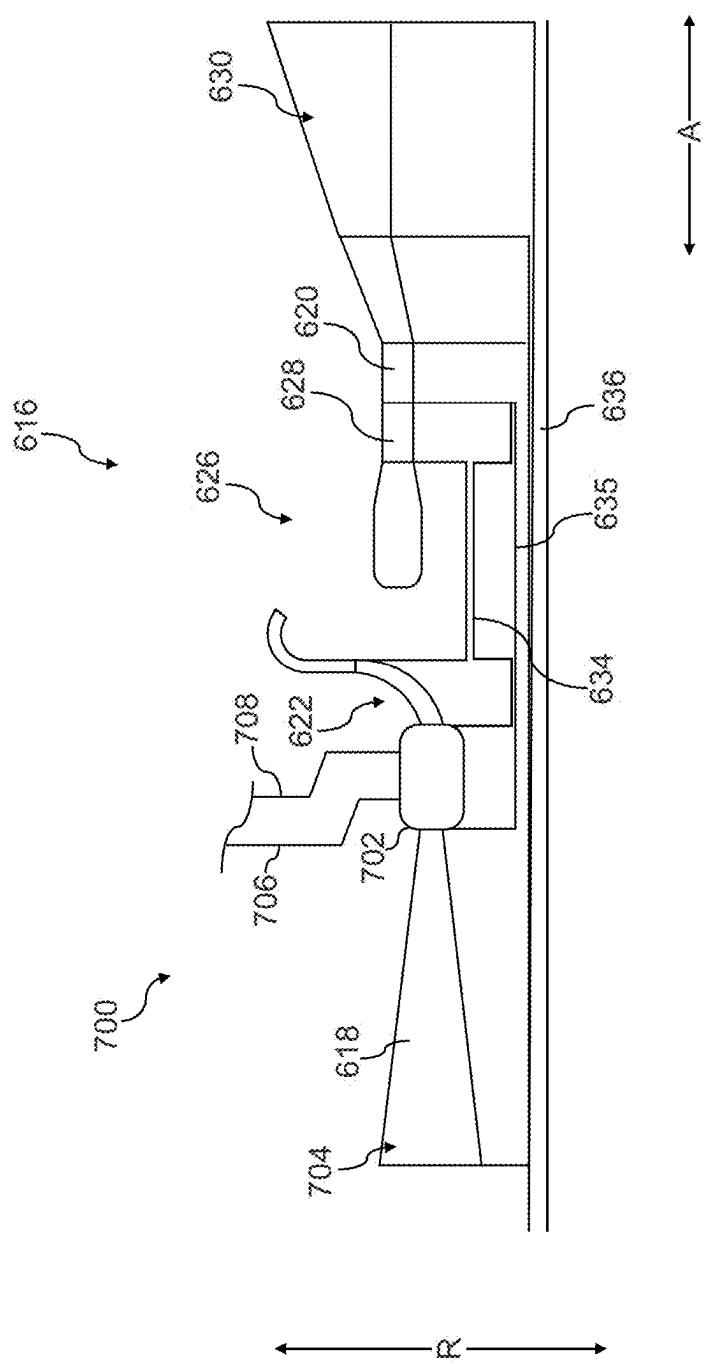


FIG. 14

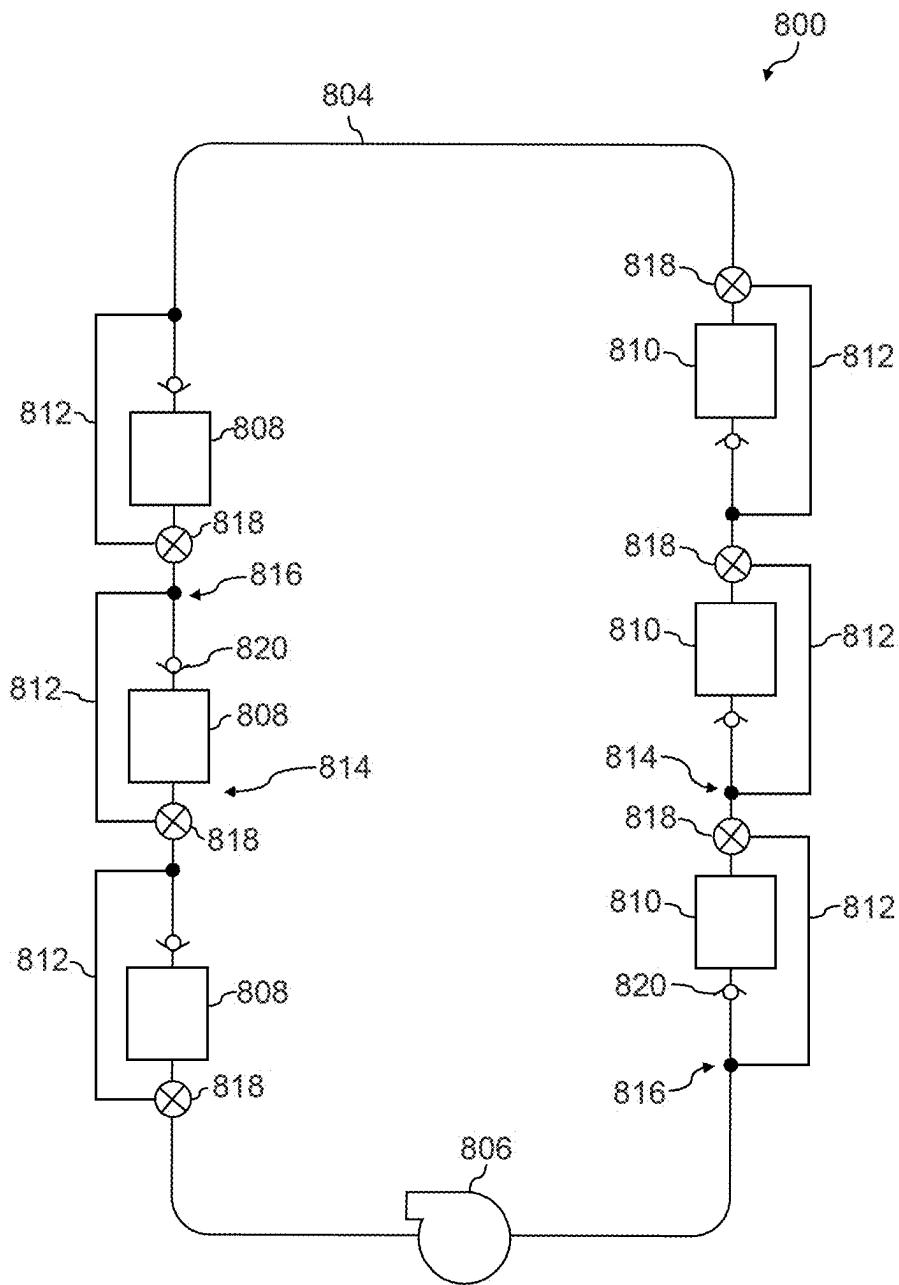


FIG. 15

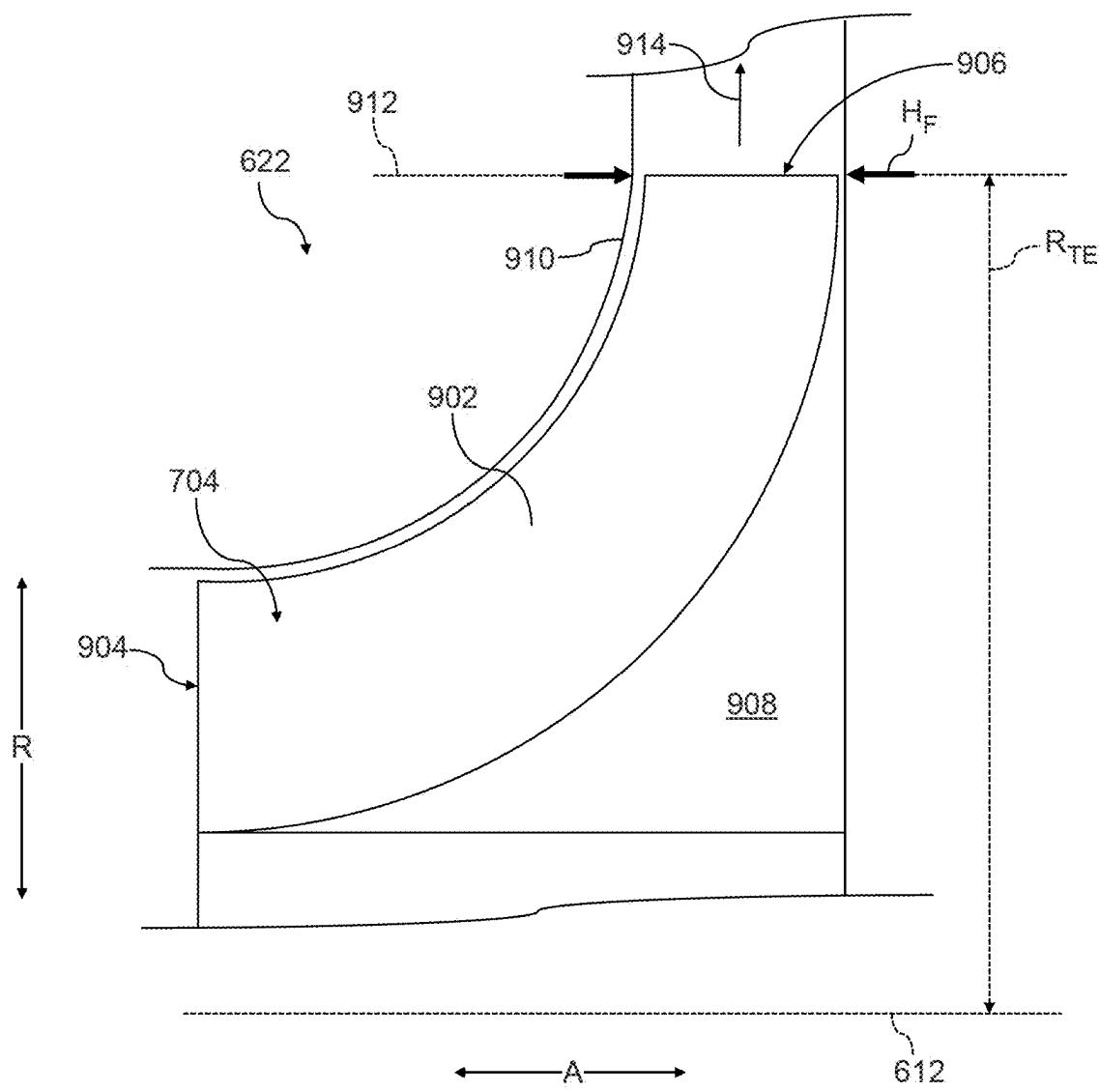


FIG. 16

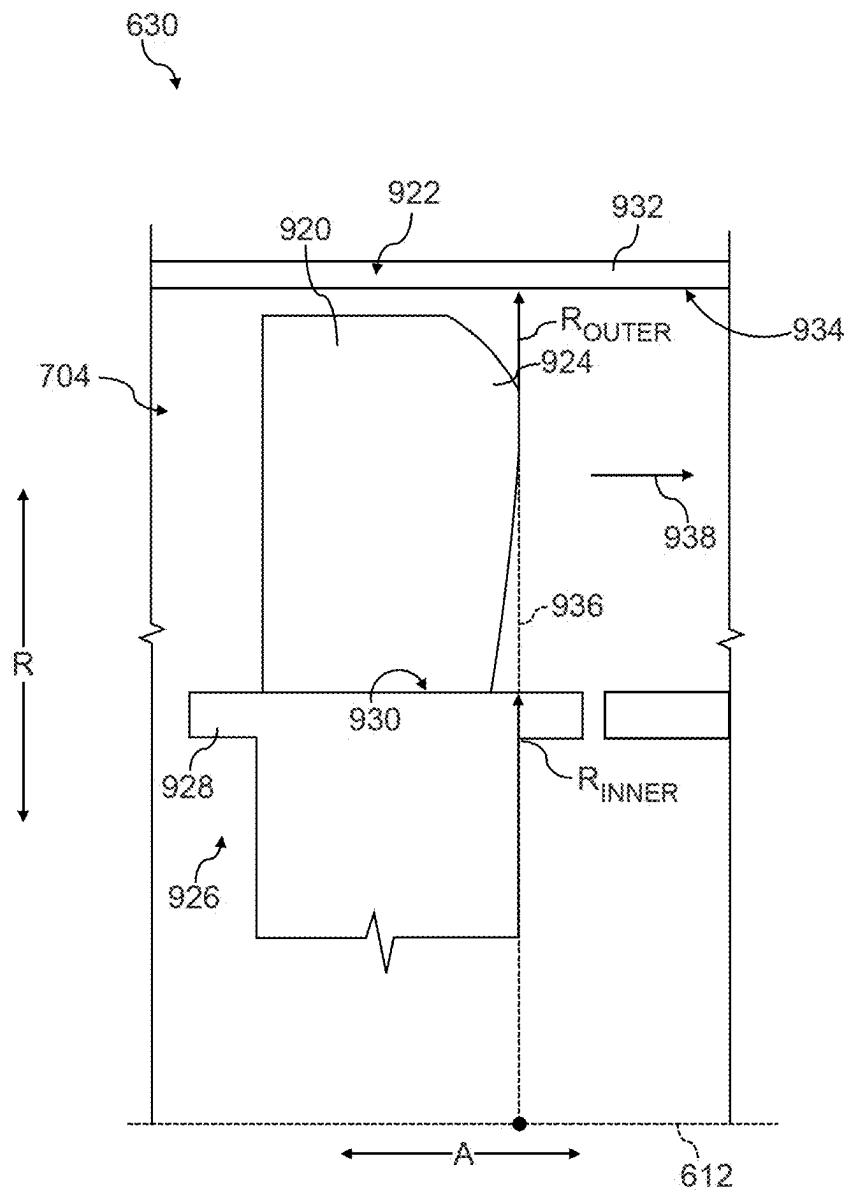


FIG. 17

	Prop pwr	T _{LPT}	EGT	A _{HPCExit}	A _{LPTExit}	CSP
Unit	kW	N·m	°C	m ²	m ²	N ² ·C/m ²
Example 1	7530.8	4737.2	972.2	0.0231	0.1951	48.4
Example 2	5844.9	3762.3	1222.6	0.0129	0.1199	111.7
Example 3	5557.9	3524.2	621.1	0.0398	0.2767	7.0
Example 4	5101.5	3243.5	537.8	0.0359	0.3320	4.7
Example 5	5165.2	3274.7	1176.6	0.0078	0.0991	163.6
Example 6	744.4	530.0	721.1	0.0085	0.0602	3.4
Example 7	4372.7	2795.6	537.7	0.0312	0.2881	4.7
Example 8	3401.0	2171.4	1148.9	0.0055	0.0686	143.3
Example 9	2126.2	1382.3	1176.7	0.0060	0.0555	67.0
Example 10	2364.0	1529.7	1176.7	0.0065	0.0596	71.4
Example 11	2359.5	1527.1	1093.4	0.0072	0.0638	55.9
Example 12	2567.8	1658.0	760.0	0.0129	0.0999	16.2
Example 13	2351.0	1522.3	972.2	0.0084	0.0715	37.4
Example 14	5165.2	3274.7	1176.6	0.0078	0.0991	163.6
Example 15	4858.5	3083.2	1093.4	0.0083	0.1013	124.1
Example 16	2915.2	1872.8	760.0	0.0094	0.0981	28.9
Example 17	5096.1	3239.3	972.2	0.0175	0.1309	44.5
Example 18	5105.1	3243.0	972.2	0.0149	0.1269	54.0
Example 19	5111.0	3245.3	972.2	0.0133	0.1243	61.9
Example 20	5158.7	3270.1	1176.7	0.0091	0.0989	140.2
Example 21	5045.3	3208.7	787.8	0.0168	0.1598	30.2

FIG. 18

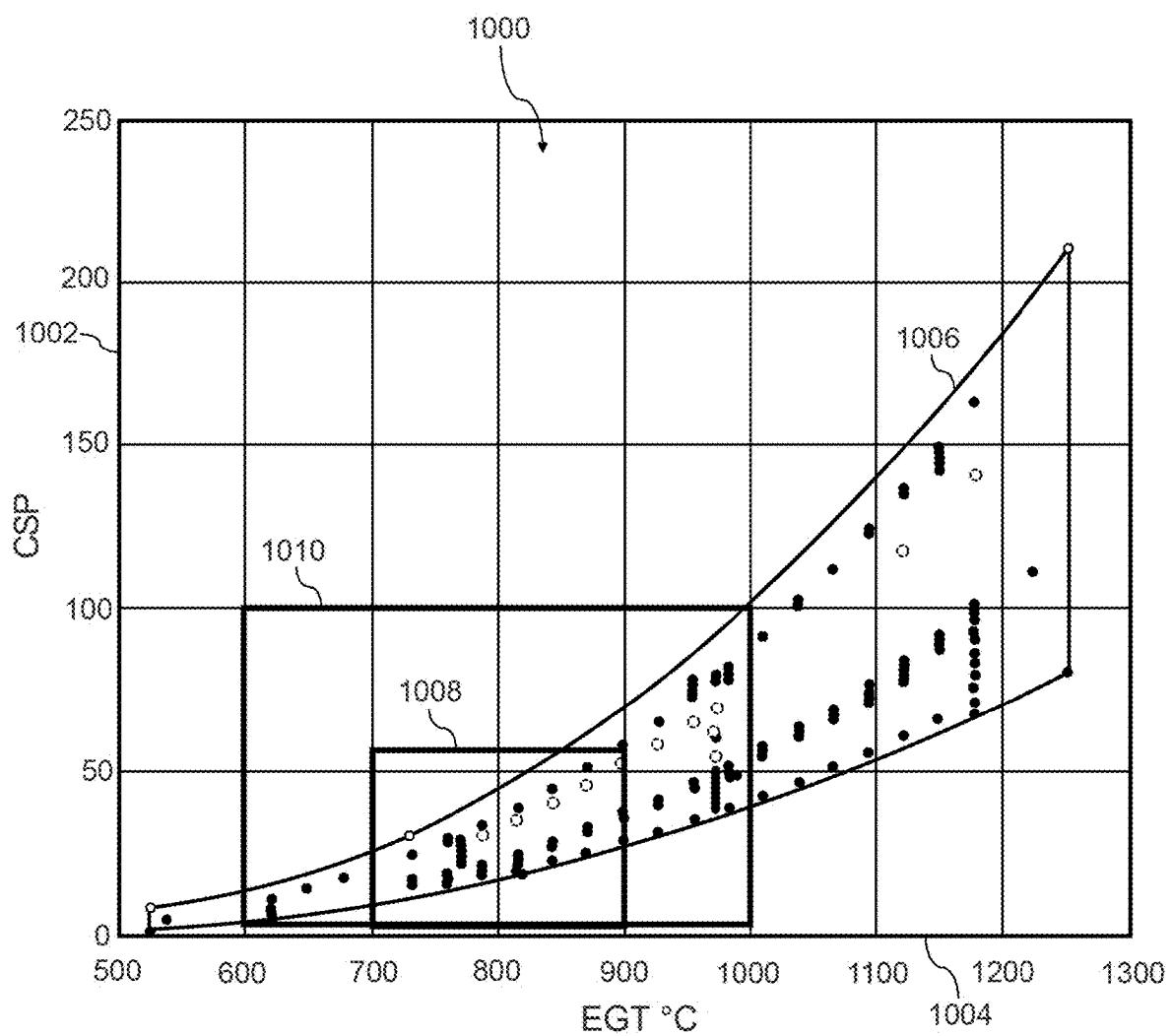


FIG. 19

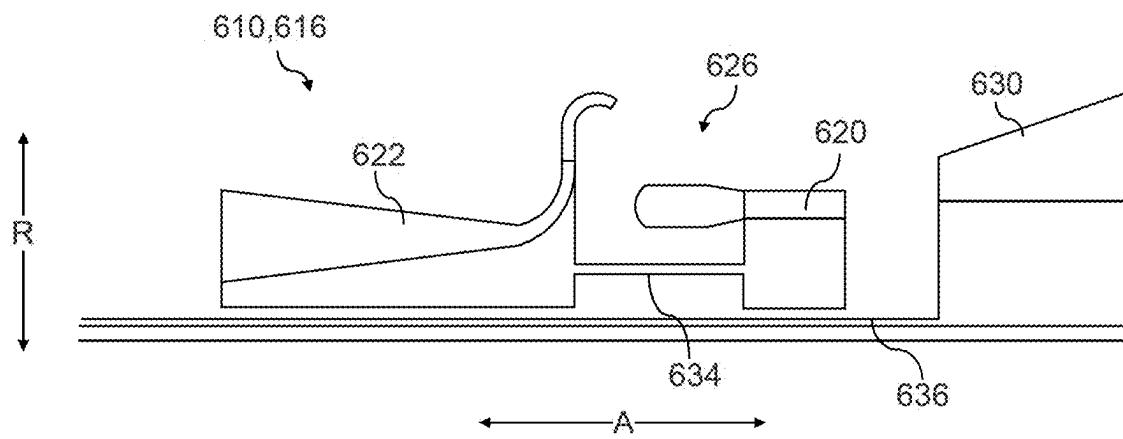


FIG. 20

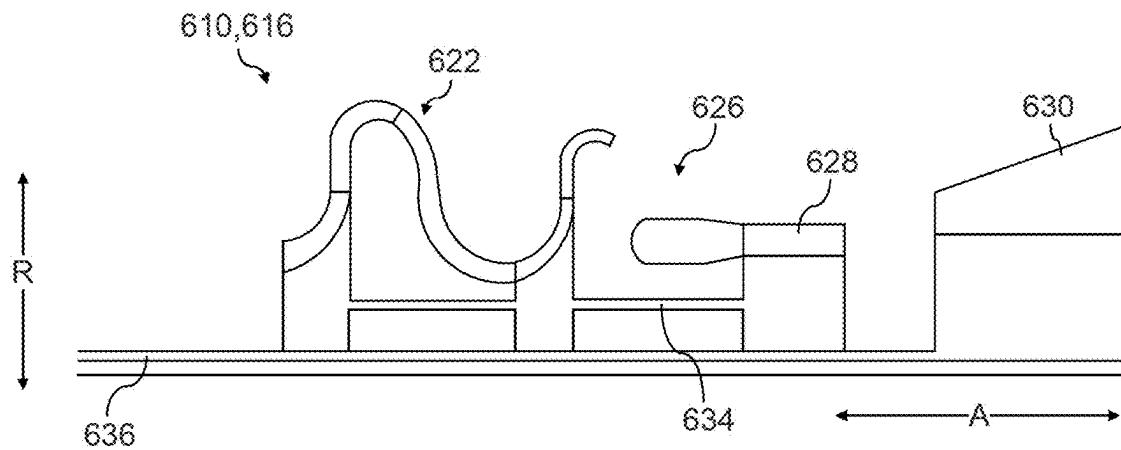


FIG. 21

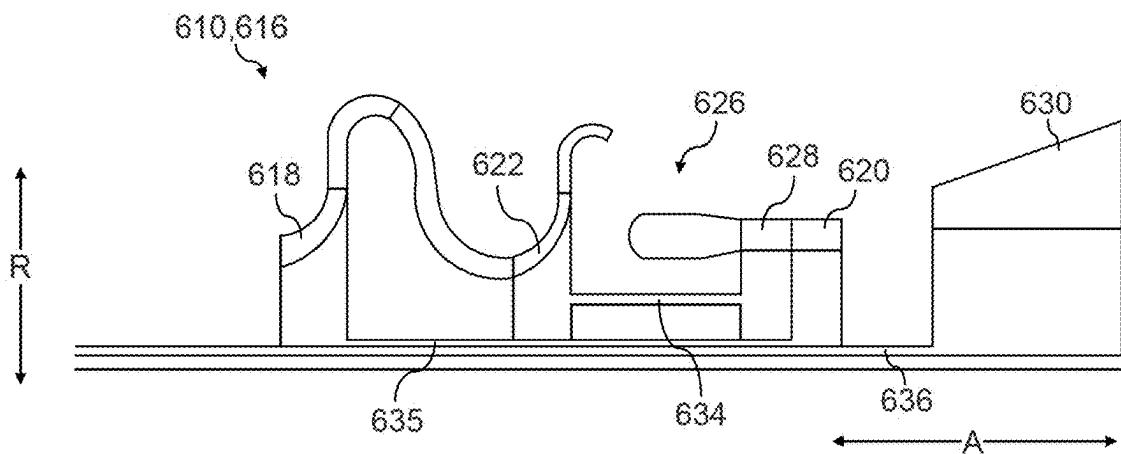


FIG. 22

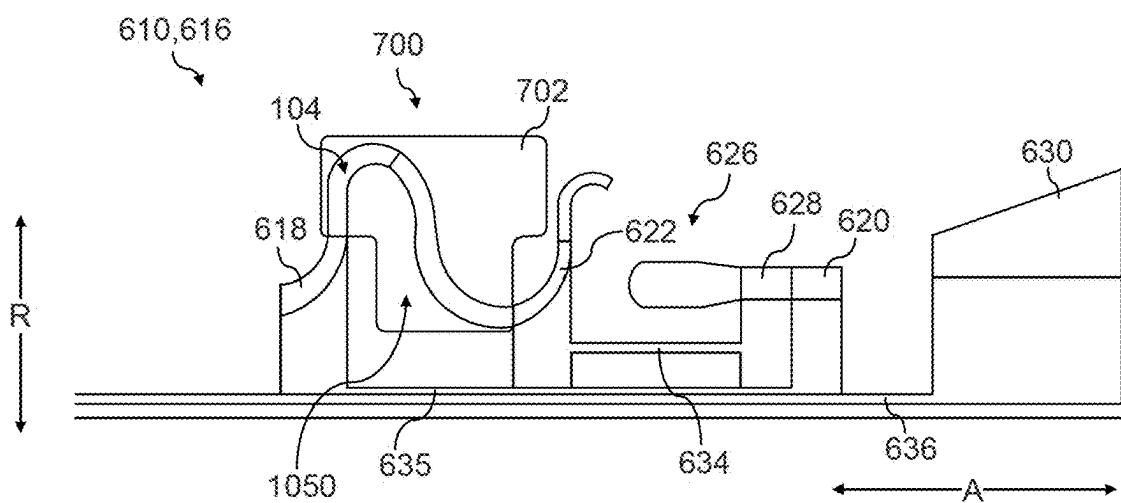


FIG. 23

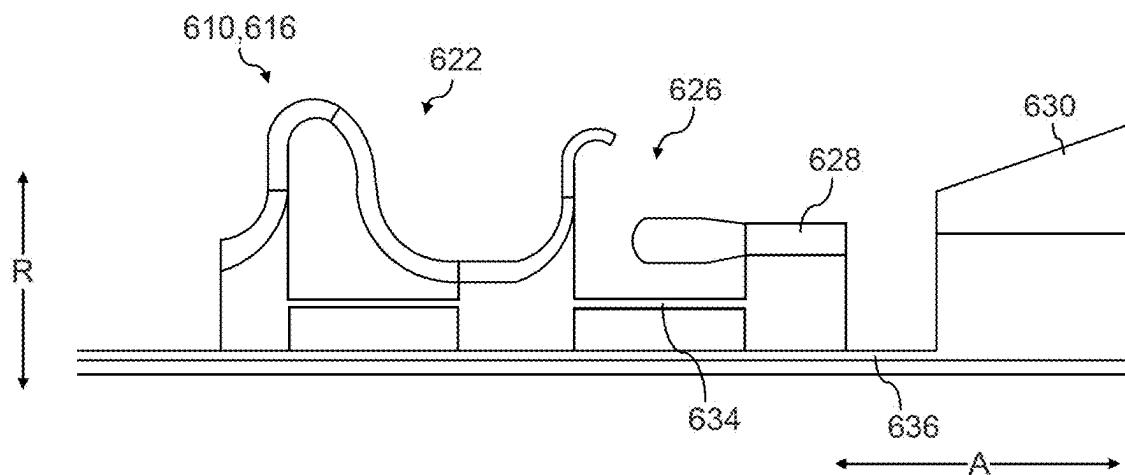


FIG. 24

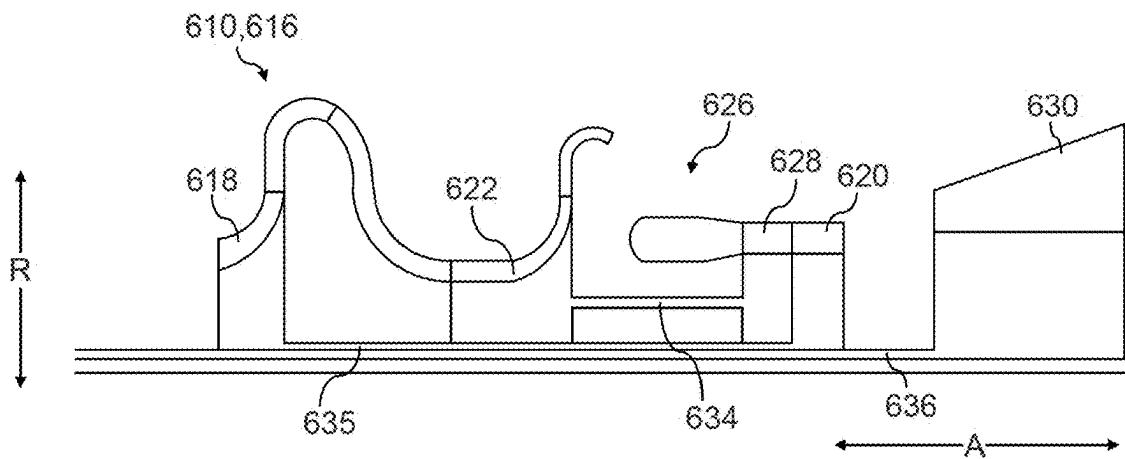


FIG. 25

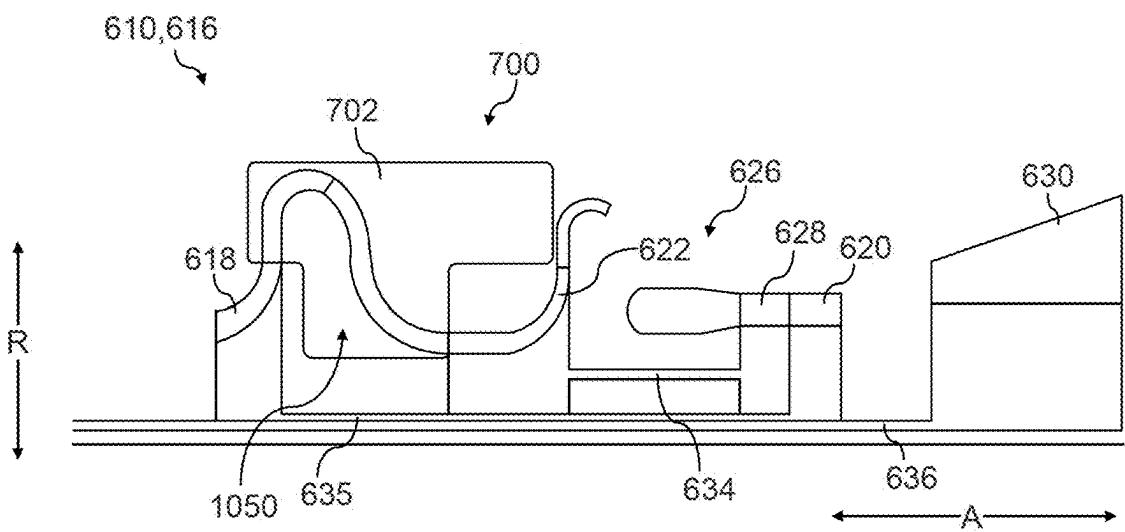


FIG. 26

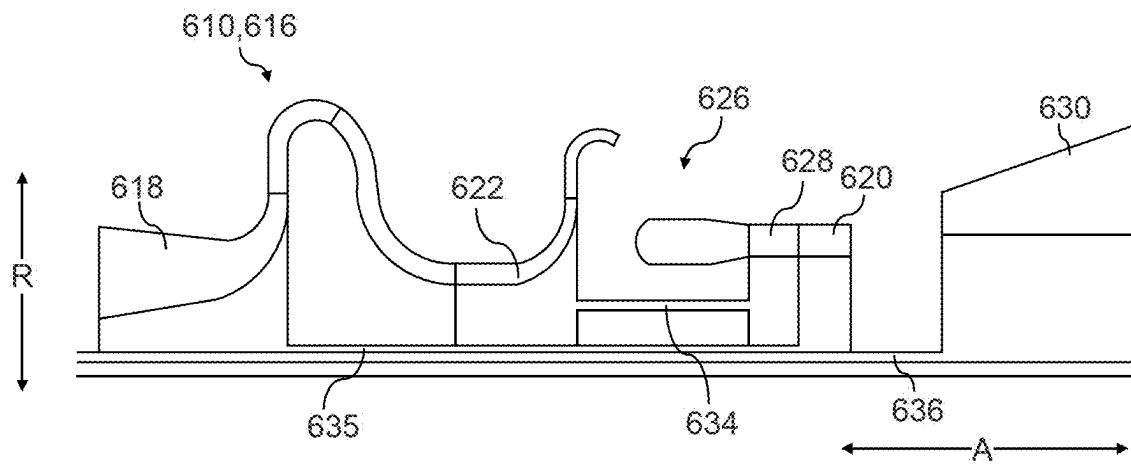


FIG. 27

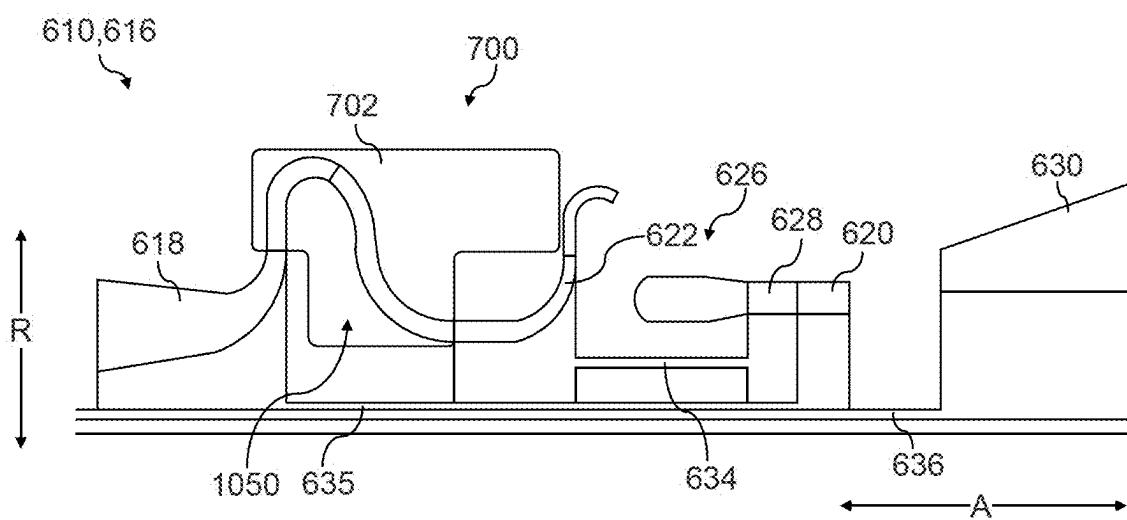


FIG. 28

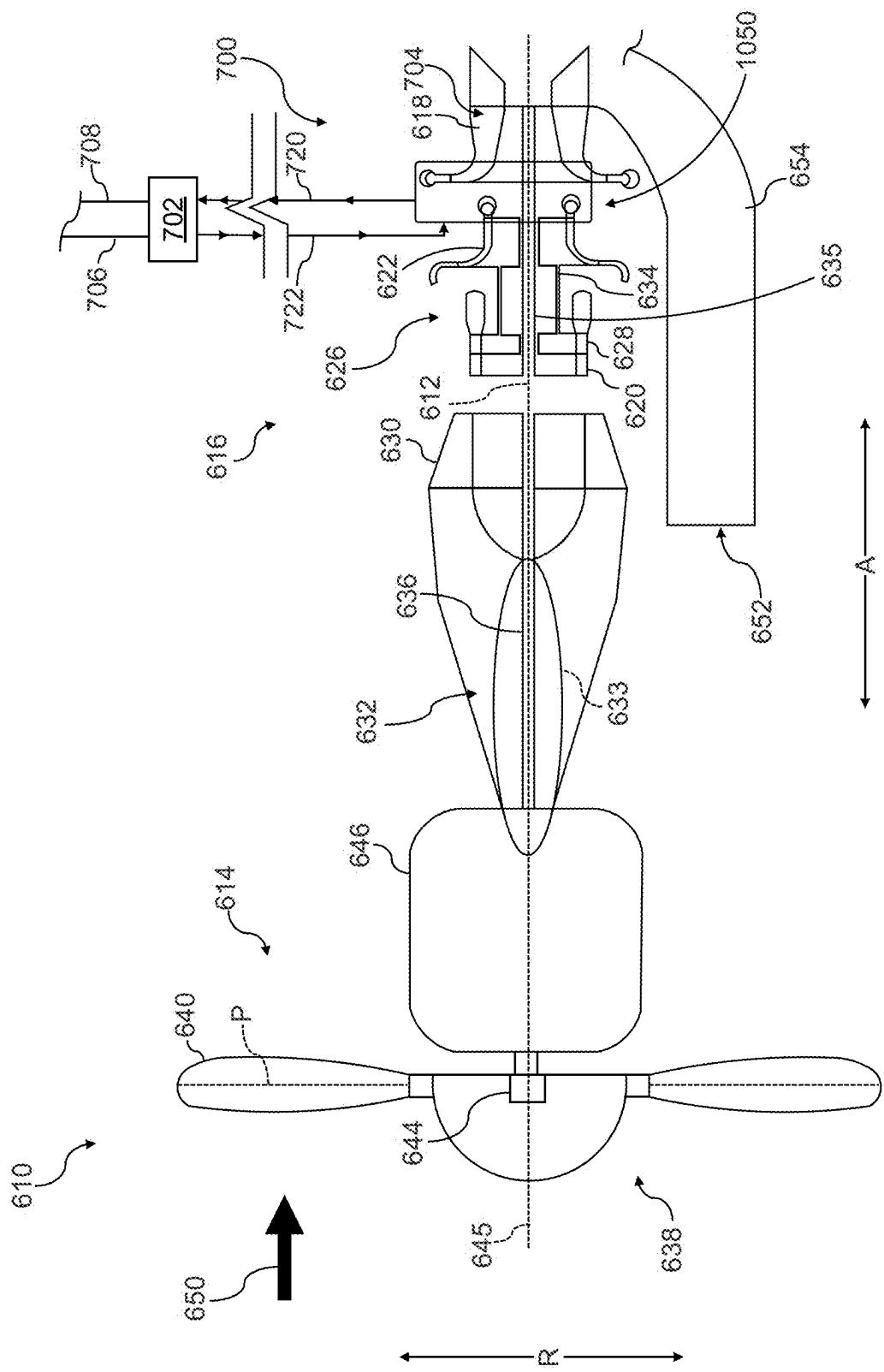


FIG. 29

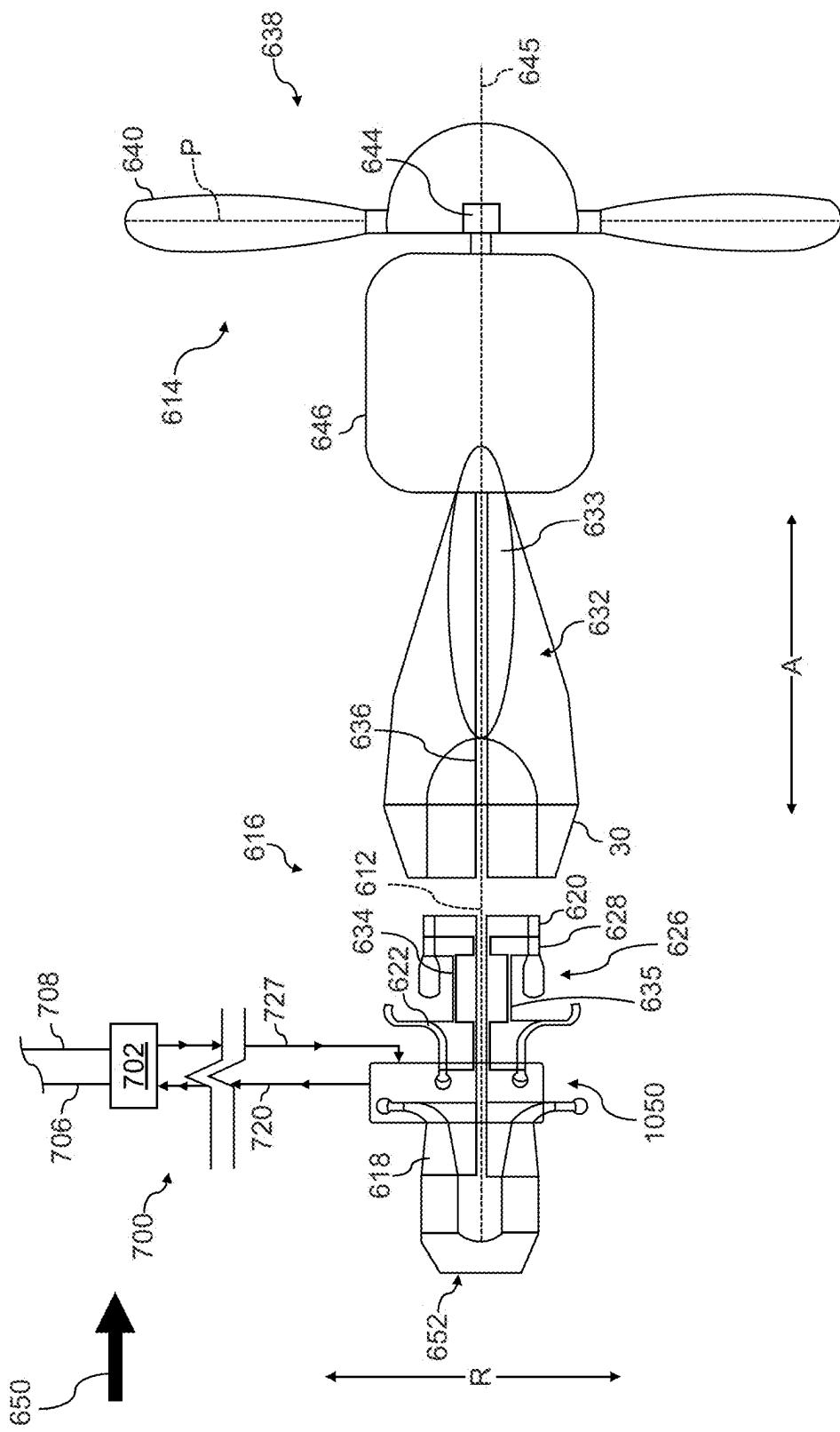


FIG. 30

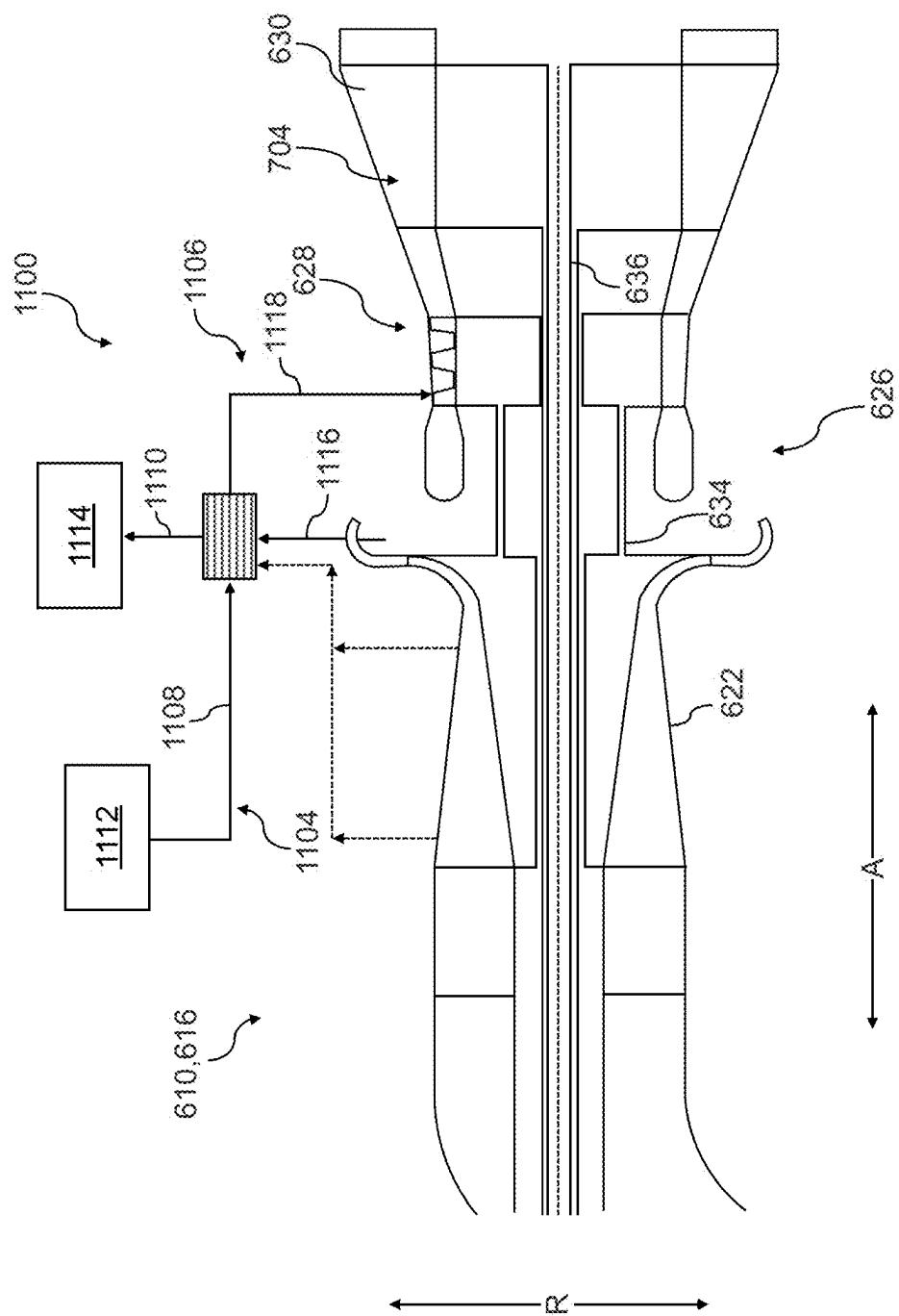


FIG. 31

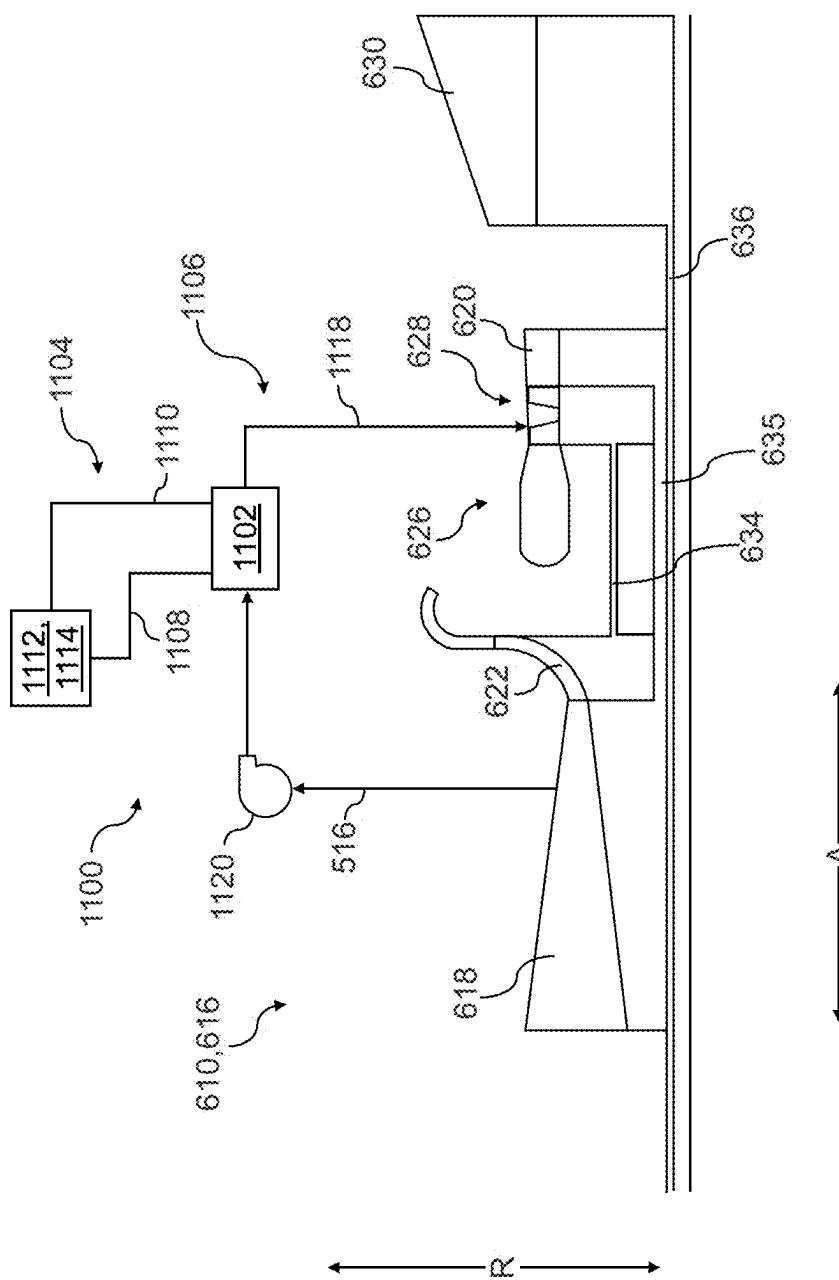
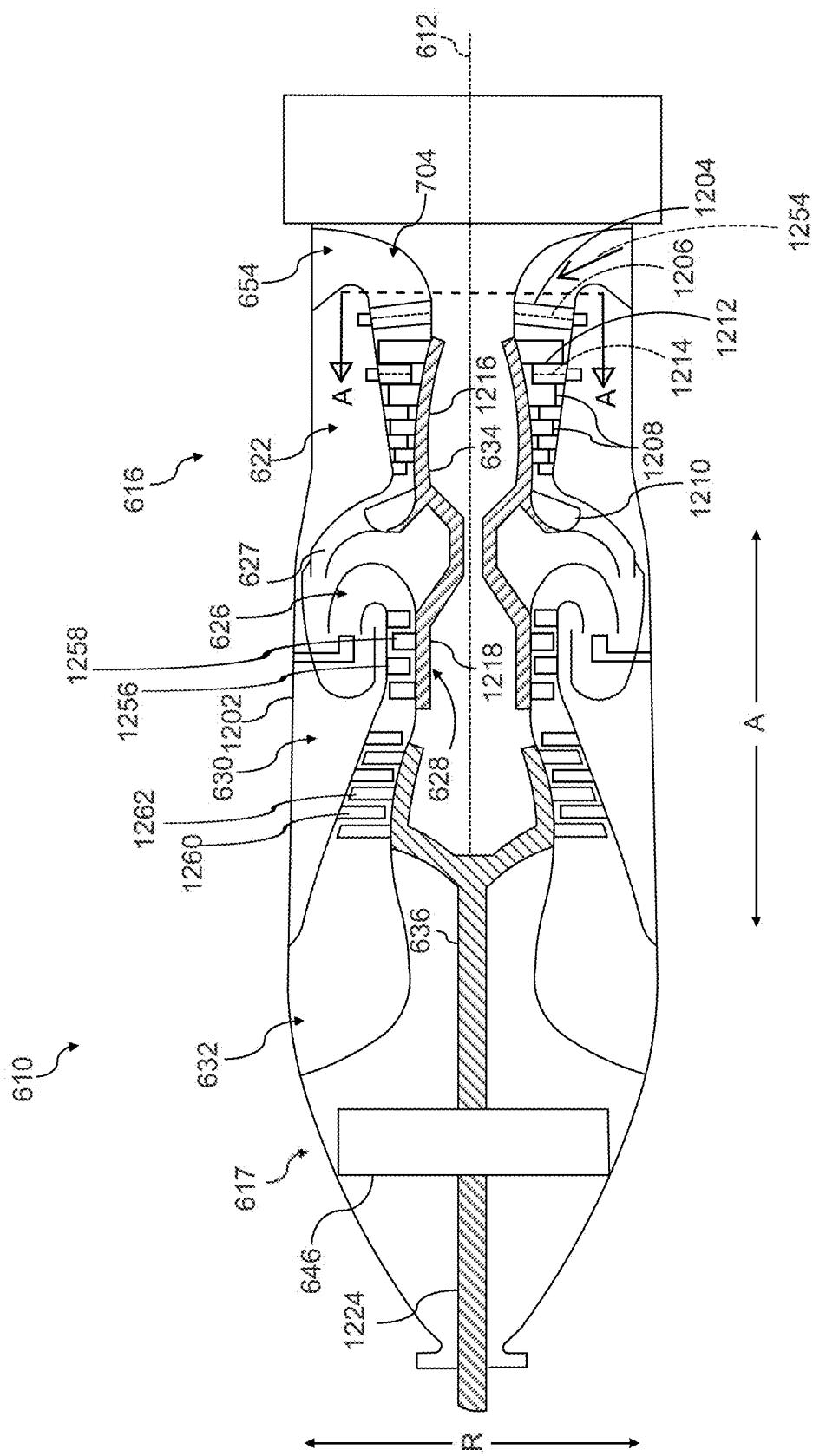


FIG. 32



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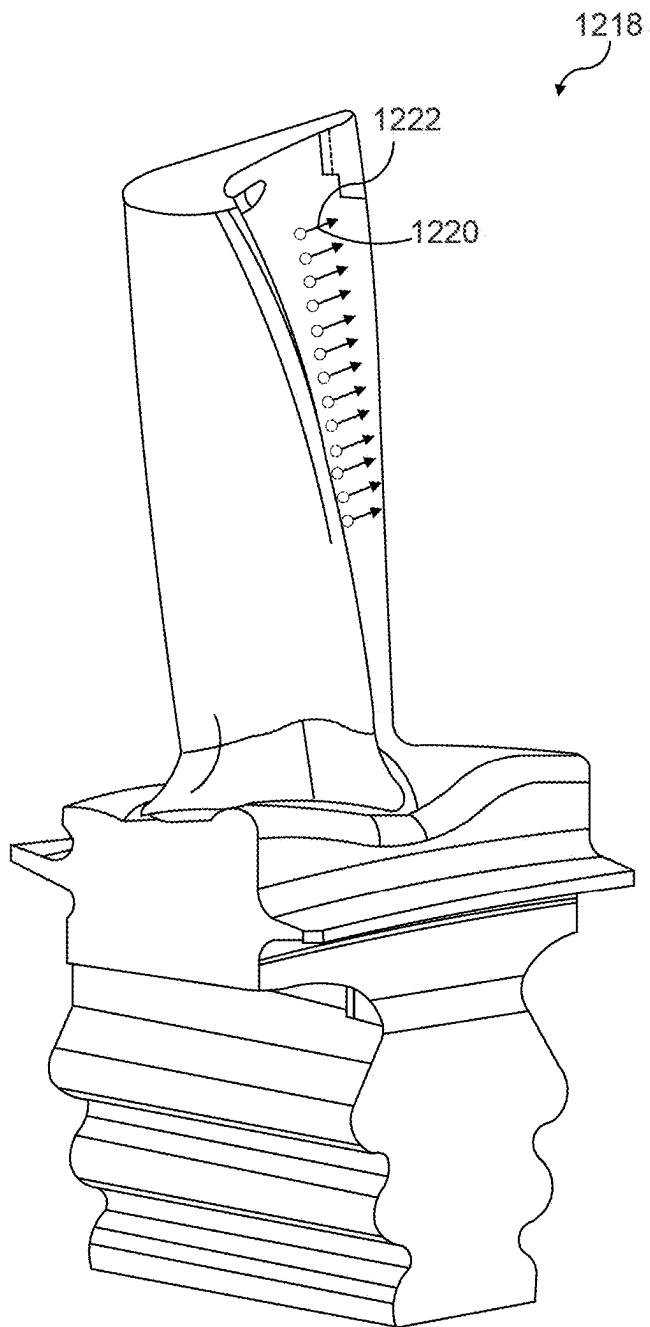


FIG. 34

1
GAS TURBINE ENGINE

CROSS-REFERENCE TO RELATED
APPLICATIONS

This application is a continuation patent application of U.S. application Ser. No. 18/976,748, filed Dec. 11, 2024, which is a continuation patent application of U.S. application Ser. No. 18/650,586 (now U.S. Pat. No. 12,19,131), filed Apr. 30, 2024, which is a continuation-in-part application of U.S. application Ser. No. 18/500,517 (now U.S. Pat. No. 12,078,107), filed Nov. 2, 2023, which is a continuation 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 (now abandoned). Each of these applications are hereby incorporated by reference in their entirety.

FIELD

The present disclosure relates to a gas turbine engine.

BACKGROUND

A gas turbine engine includes 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, e.g., propel an aircraft in flight. The turbomachine is mechanically coupled to an output shaft to, in the case of a turboprop engine, drive a propeller assembly of the gas turbine engine during operation.

BRIEF DESCRIPTION OF THE DRAWINGS

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:

FIG. 1 is a schematic cross-sectional view of a three-stream engine in accordance with an exemplary embodiment of the present disclosure.

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.

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.

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.

FIG. 5 is a schematic view of a thermal transport bus of the present disclosure.

FIG. 6 is a table depicting numerical values showing the relationships between various parameters in accordance with various example embodiments of the present disclosure.

FIG. 7 is a graph depicting a range of corrected specific thrust values and maximum exhaust gas temperature values of gas turbine engines in accordance with various example embodiments of the present disclosure.

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FIG. 8 is a schematic view of a ducted turbofan engine in accordance with an exemplary aspect of the present disclosure.

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.

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.

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.

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

FIG. 13 is a schematic view of a turboprop engine in accordance with an exemplary aspect of the present disclosure.

FIG. 14 is a schematic view of a turbomachine of the exemplary turboprop engine of FIG. 13 in accordance with an exemplary aspect of the present disclosure.

FIG. 15 is a schematic view of a thermal transport bus of the present disclosure.

FIG. 16 is a close-up view of an aft-most stage of high pressure compressor rotor blades within the exemplary turboprop engine of FIG. 13.

FIG. 17 is a close-up view of an aft-most stage of low pressure turbine rotor blades within the exemplary turboprop engine of FIG. 13.

FIG. 18 is a table depicting numerical values showing the relationships between various parameters in accordance with various example embodiments of the present disclosure.

FIG. 19 is a graph depicting a range of corrected specific power values and maximum exhaust gas temperature values of gas turbine engines in accordance with various example embodiments of the present disclosure.

FIG. 20 is a schematic view of a turbomachine of a gas turbine engine in accordance with another exemplary aspect of the present disclosure.

FIG. 21 is a schematic view of a turbomachine of a gas turbine engine in accordance with another exemplary aspect of the present disclosure.

FIG. 22 is a schematic view of a turbomachine of a gas turbine engine in accordance with another exemplary aspect of the present disclosure.

FIG. 23 is a schematic view of a turbomachine of a gas turbine engine in accordance with another exemplary aspect of the present disclosure.

FIG. 24 is a schematic view of a turbomachine of a gas turbine engine in accordance with another exemplary aspect of the present disclosure.

FIG. 25 is a schematic view of a turbomachine of a gas turbine engine in accordance with another exemplary aspect of the present disclosure.

FIG. 26 is a schematic view of a turbomachine of a gas turbine engine in accordance with another exemplary aspect of the present disclosure.

FIG. 27 is a schematic view of a turbomachine of a gas turbine engine in accordance with another exemplary aspect of the present disclosure.

FIG. 28 is a schematic view of a turbomachine of a gas turbine engine in accordance with another exemplary aspect of the present disclosure.

FIG. 29 is a schematic view of a turboprop engine in accordance with an exemplary aspect of the present disclosure.

FIG. 30 is a schematic view of a turboprop engine in accordance with an exemplary aspect of the present disclosure.

FIG. 31 is a schematic view of a turbomachine of a gas turbine engine in accordance with another exemplary aspect of the present disclosure.

FIG. 32 is a schematic view of a turbomachine of a gas turbine engine in accordance with another exemplary aspect of the present disclosure.

FIG. 33 is a schematic view of a gas turbine engine in accordance with an exemplary aspect of the present disclosure.

FIG. 34 is a schematic view of a high pressure turbine rotor blade in accordance with an exemplary aspect of the present disclosure.

DETAILED DESCRIPTION

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.

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.

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. Cooled cooling air systems contemplated by the present disclosure may be incorporated into turbofan engines, open rotor engines, turboprop engines (see FIGS. 31 and 32), and/or turboshaft engines.

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.

The term “intercooler assembly” is used herein to mean a system configured to provide cooling to a gas flow through a compressor section of a turbomachine of a gas turbine engine, transferring heat from such gas flow to one or more heat sinks on the gas turbine engine and/or an aircraft incorporating the gas turbine engine. Exemplary intercooler assemblies of the present disclosure may include an intercooler heat exchanger positioned within a working gas flowpath through the compressor section, integrated into (or otherwise in thermal communication with) one or more liners or walls of the working gas flowpath through the compressor section, or located external to a turbomachine of the engine with all or a portion of the airflow through the compressor section being redirected to the intercooler heat exchanger.

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.

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.

The terms “forward” and “aft” refer to relative positions within a gas turbine engine or vehicle, and are based on a normal operational attitude of the gas turbine engine or vehicle. More particularly, forward and aft are used herein with reference to a direction of travel and a direction of propulsive thrust of the gas turbine engine or vehicle.

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.

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.

The singular forms “a”, “an”, and “the” include plural references unless the context clearly dictates otherwise.

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).

It should be appreciated, that as used herein, terms of approximation, such as “about” or “approximately” refer to being within a ten percent (10%) margin.

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.

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.

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.

The term “takeoff power level” refers to a power level of a gas turbine engine used during a maximum steady state permitted power level during a standard day operating condition, as may be documented in a Federal Aviation Administration (“FAA”)-type certificate data sheet (e.g., an FAA certification data sheet, a European Aviation Safety Agency (“EASA”) data sheet, or the like).

The term “output power” of a gas turbine engine, with respect to a turboprop or turboshaft gas turbine engine, refers to a brake horsepower providing to an output shaft (e.g., a propeller shaft of a propeller assembly, or an output drive shaft 1224 of a turboprop engine) when the during operation of the gas turbine engine at a takeoff power level. The output power of a gas turbine engine is sometimes also referred to as an output power of a turbomachine in the context of a turboprop or turboshaft gas turbine engine.

As used herein, the “maximum steady state permitted power level” refers to a maximum permitted power level for any steady state duration of time (e.g., a maximum take off power, a maximum 5 minute take off power, or other lowest duration permitted power). As used herein, the “maximum steady state permitted power level” does not refer to any transient operating conditions, one engine inoperable operating conditions, or the like.

The term “standard day operating condition” refers to ambient conditions of sea level altitude, 59 degrees Fahrenheit, and 60 percent relative humidity.

The term “overall pressure ratio” of a compressor section refers to a ratio of a pressure at an outlet of a last stage of compression (prior to combustion) to a pressure at an inlet of the compressor section (prior to any compression in the compressor section). Unless specified otherwise, the overall pressure ratio is defined when the engine is operated at a takeoff power level.

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.

The term maximum exhaust gas temperature (referred to herein as “maximum EGT”) refers to a maximum permitted takeoff temperature (i.e., when operated at a maximum steady state permitted power level) documented in a Federal Aviation Administration (“FAA”)-type certificate data sheet

(e.g., an FAA certification data sheet, a European Aviation Safety Agency (“EASA”) data sheet, or the like). For example, in certain exemplary embodiments, the term maximum 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 maximum 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 maximum 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).

For example, with reference to the exemplary engine 610 discussed below with reference to FIGS. 13 and 14, the term maximum EGT refers to a maximum permitted takeoff temperature of an airflow after a first stator downstream of a last stage of rotor blades of the intermediate pressure turbine 620 (at a location into a first of the plurality of low pressure turbine rotor blades 630). In embodiments wherein the engine is configured as a two spool engine (as compared to the three spool engine of FIGS. 13 and 14 see, e.g., FIG. 20), the term maximum EGT refers to a maximum permitted takeoff temperature of an airflow after a first stator downstream of the last stage of rotor blades of the high pressure turbine 628 (at a location into a first of the plurality of low pressure turbine rotor blades of the low pressure turbine 630).

The term EGT is sometimes also referred to as an indicated turbine exhaust gas temperature or indicated turbine temperature, and the term maximum EGT is sometimes also referred to as a redline EGT.

The term maximum drive turbine shaft torque (T_{OUT}) refers to a torque on a shaft of a gas turbine engine on a high speed side of a power gearbox of the gas turbine engine (which typically is the shaft coupled to the low pressure turbine) when the gas turbine engine is operated at a maximum steady state permitted power level documented in an FAA-type certificate data sheet (e.g., an FAA certification data sheet, a European Aviation Safety Agency (“EASA”) data sheet, or the like). As will be appreciated, the torque on the drive turbine shaft may be determined using an output power (P_{OUT} , in kilowatts, or “kW”) of a drive turbine (e.g., of a low pressure turbine in the embodiments described herein) coupled to the drive turbine shaft and rotational speed (N; in revolutions per minute, or “rpm”) of the drive turbine, using the equation: $T_{OUT}=9,548.8\times P_{OUT}/N$.

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.

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.

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.

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.

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.

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.

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, maximum 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, maximum 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.

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.

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.

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.

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.

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.

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.

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.

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.

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.

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.

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.

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.

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.

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.

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

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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.

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.

Notably, for the embodiment depicted, the engine 100 includes one or more features to increase an efficiency of a third stream thrust, $F_{N_{3S}}$ (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.

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.

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.

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, $F_{N_{3S}}$, during one or more engine operating conditions. Further, by introducing a variability in the geometry of the inlet guide vanes 186 and the

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fan exhaust nozzle 178, the engine 100 may be capable of generating more efficient third stream thrust, $F_{N_{3S}}$, across a relatively wide array of engine operating conditions, including takeoff and climb as well as cruise.

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.

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.

As will be appreciated, the engine 100 defines a total sea level static thrust output $F_{N_{Total}}$, 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.

The total sea level static thrust output $F_{N_{Total}}$ may generally be equal to a sum of: a fan stream thrust $F_{N_{Fan}}$ (i.e., an amount of thrust generated by the fan 152 through the bypass passage 194), the third stream thrust $F_{N_{3S}}$ (i.e., an amount of thrust generated through the fan duct 172), and a turbomachine thrust $F_{N_{TM}}$ (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 $F_{N_{Total}}$ 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.

As will be appreciated, the engine 100 defines a maximum 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).

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

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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 alternately 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 alternately 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.

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.

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 ($A_{HPCExit}$) within the reference plane 230. More specifically, the HP compressor 128 defines an inner radius (R_{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 (R_{OUTER}) 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):

$$A_{HPCExit} = \pi(R_{OUTER}^2 - R_{INNER}^2). \quad \text{Expression (1):}$$

The inventors of the present disclosure have found that for a given total thrust output (Fn_{Total}), a decrease in size of the high pressure compressor exit area ($A_{HPCExit}$) 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 maximum exhaust gas temperature (EGT). In particular, the inventors of the present disclosure have found that the high pressure compressor exit area ($A_{HPCExit}$) 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_{Total}) of the engine 100.

Referring back to FIG. 2, the exemplary engine 100 depicted includes one or more technologies to accommodate

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the relatively small high pressure compressor exit area ($A_{HPCExit}$) for the total thrust output (Fn_{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.

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.

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.

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.

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.

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.

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**.

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**.

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.

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.

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**.

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**.

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%.

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**.

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.

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 CO₂.

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**.

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.

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.

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.

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.

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.

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.

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).

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.

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, maximum 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 maximum 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.

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, maximum exhaust gas temperature, and compressor exit area, respectively.

The inventors of the present disclosure discovered bounding the relationship between a product of total thrust output and maximum 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 be 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.

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.

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

$$CST = Fn_{Total} \times EGT / (A_{HPCExit}^2 \times 1000), \quad \text{Expression (2)}$$

where CST is corrected specific thrust; Fn_{Total} is a total sea level static thrust output of the gas turbine engine in pounds; EGT is maximum exhaust gas temperature in degrees Celsius; and $A_{HPCExit}$ is a high pressure compressor exit area in square inches.

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.

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.

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 degrees Celsius and less than 1250 degrees Celsius, such as greater than 1150 degrees Celsius and less than 1250 degrees Celsius, such as greater than 1000 degrees Celsius and less than 1300 degrees Celsius.

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.

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.

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_{Total} of the engine 100 may generally be equal to a sum of: a fan stream thrust Fn_{Fan} (i.e., an amount of thrust generated by a fan 152 through a bypass passage 194) and a turbomachine thrust Fn_{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.

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.

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).

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.

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.

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.

However, referring particularly 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.

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.

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.

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.

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 252A and a low-pressure duct assembly 252B, and the CCA heat exchanger 254 includes a high-pressure CCA heat exchanger 254A and a low-pressure CCA heat exchanger 254B.

The high-pressure duct assembly 252A 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 252A may be configured to receive a high-pressure cooling airflow from the HP compressor 128 through the high-pressure duct assembly 252A and provide such high-pressure cooling airflow to the first stage 214 of HP turbine rotor blades 206. The high-pressure CCA heat exchanger 254A may be configured to reduce a temperature of the high-pressure cooling airflow through the high-pressure duct assembly 252A at a location upstream of the first stage 214 of HP turbine rotor blades 206.

The low-pressure duct assembly 252B 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 252B 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 252B may be configured to receive a low-pressure cooling airflow from the LP compressor 126 through the low-pressure duct assembly 252B and provide such low-pressure cooling airflow to the second stage of HP turbine rotor blades 206. The low-pressure CCA heat exchanger 254B may be configured to reduce a temperature of the

low-pressure cooling airflow through the low-pressure duct assembly 252B upstream of the second stage of HP turbine rotor blades 206.

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.

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.

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.

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.

For the embodiment of FIG. 12, the term maximum 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).

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 252A may be positioned inwardly of the working gas flow path 142 along the radial direction R and the low-pressure duct assembly 252B may be positioned outwardly of the working gas flow path 142 along the radial direction R).

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.

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.).

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 (Al₂O₃), silicon dioxide (SiO₂), 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.

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 (Al₂O₃), silicon dioxide (SiO₂), aluminosilicates such as mullite, or mixtures thereof), or mixtures thereof.

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_2O_3), silicon dioxide (SiO_2), aluminosilicates, and mixtures thereof. Aluminosilicates can include crystalline materials such as mullite ($3\text{Al}_2\text{O}_3 \ 2\text{SiO}_2$), as well as glassy aluminosilicates.

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.

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.

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.

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.

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.

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 maximum 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: $Fn_{Total} \times EGT / (A_{HPCExit}^2 \times 1000)$.

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.

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.

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).

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^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^2 and 160 hp/ft^2 or higher at cruise altitude during a cruise operating mode, depending on whether the engine is an open rotor or ducted engine.

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.

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.

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.

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.

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.

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.

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.

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.

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.

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.

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 described 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.

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.

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.

The discussion above is primarily directed to gas turbine engines configured as a turbofan engine or open rotor engine. After additional research and testing, it was discovered that the concepts outlined above can similarly apply to produce improvements in gas turbine engines configured to function as a turboprop or turboshaft engine.

Turboshaft and turboprop engines generally include a turbomachine, the turbomachine including a compressor section, a combustion section, a turbine section, and defining a working gas flowpath therethrough. The power generated by the turbomachine is transmitted to a load, e.g., a propeller in the case of a turboprop engine, through an output shaft. In such a manner, for turboprop engines, output shaft causes the propeller rotor blades to rotate and generate a thrust output. A turboshaft engine is configured in a similar manner, but without the propeller assembly. With a turboshaft engine, the load driven by the output shaft may be a number of different aeronautical vehicle loads, including a vertical thrust propeller (driven through one or more gears), an electric machine, etc. Efficiency and power density of the gas turbine engine (or simply "engine") are important factors in the performance of an aeronautical vehicle incorporating the engine.

Conventional turboprop and turboshaft engine design has been constrained by the thermal efficiency limits imposed by the temperatures and pressures at an exit of a high pressure compressor (HPC) of the compressor section and as well as an exhaust gas temperature (EGT). For example, for a desired engine power output produced from an increased pressure ratio across the HPC, there is an increase in the gas temperature at the exit of the HPC, at a combustor inlet, at the turbine section inlet, through the turbine section, and through an exhaust section of the engine. These constraints have historically dictated a size of a core (the core being, e.g., the HPC, combustor, and a high pressure turbine) of the turbomachine of the engine and, consequently, an overall power output and efficiency of the engine.

The inventors found that there are approaches to making an engine capable of operating at higher temperatures while providing a net benefit to engine performance: intercooling a compressor section of a turbomachine of the engine, reducing the temperature of a gas used to cool core components, utilizing advanced materials capable of withstanding higher operating temperature conditions, or combinations thereof. As thermal efficiency increases, the size of the core can be reduced for a given power output, resulting in a turbomachine that is physically smaller for a given power output. This improved power density is of significant value.

The inventors discovered that improvements in power density may be achieved without having to pay unacceptable costs in other aspects of the engine design, such as complexity, reliability, diminished engine cycles requiring part replacement or integration with airframes having different requirements for engine size, shape and power transfers.

In the context of a turboprop or turboshaft engine that utilizes these advanced technologies, the inventors discovered, unexpectedly and contrary to conventional expectations, that the benefits of operating at higher temperatures and pressures, while maintaining or reducing the core size, can indeed outweigh the associated costs of incorporation these advanced technologies. This discovery was made during the course of designing various engine architectures with different power classes and mission requirements. In particular, the inventors discovered a significant relationship between engine power output, core size, and the conditions exiting the HPC and through the turbine section, whereby including the noted technologies produces a net benefit. Previously it was thought that the cost for including one or more of these advanced technologies was too prohibitive, as compared to the benefits of increasing the temperatures through the core (e.g., EGT).

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 engine power output, based on this discovery. The cooled cooling air system may receive an airflow from, e.g., 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, the first stage of high pressure turbine rotor blades may be capable of withstanding increased temperatures by using the cooled cooling air, providing a net benefit to the engine, i.e., providing an increase in thermal efficiency of the engine, taking into consideration the costs associated with accommodations made for the system used to cool the cooling air.

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 external to the turbomachine (e.g., downstream of a propeller in a propeller stream), 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.

Similarly, the inventors of the present disclosure found that an intercooler assembly could additionally or alternatively be included for engines within the bounds disclosed herein while maintaining or even increasing the maximum engine power output, based on this discovery. The intercooler assembly can enable higher overall pressure ratios by reducing a temperature of a gas flow through one or more

stages of the compressor section, such that for a given overall pressure ratio, the gas temperature at the exit of the HPC is reduced. The inventors reached this conclusion after similarly evaluating potentially negative impacts to engine performance brought on by introduction of an intercooler assembly. For example, in order to provide increases in temperature reduction of an airflow through the compressor section an intercooler assembly may generally require corresponding increases in pressure drop across a heat exchanger of the intercooler assembly. Therefore, in order to achieve a higher temperature reduction by the intercooler assembly, the intercooler assembly traditionally causes a reduction in efficiency and overall compressor pressure ratio.

With a goal of arriving at an improved turboprop or turboshaft engine capable of operating at higher temperatures, e.g., through the turbine section, the inventors have proceeded in the manner of designing turboprop and turboshaft engines having an overall pressure ratio (and associated HPC exit area), a maximum output power, maximum exhaust gas temperature, and the supporting technology characteristics; checking the propulsive/thermal efficiency and qualitative engine characteristics of the designed engine; redesigning the engine to have higher or lower compression ratios (and associated HPC exit areas) based on the impact on other aspects of the architecture, total power output, maximum exhaust gas temperature, and supporting technology characteristics; rechecking the propulsive/thermal efficiency and qualitative engine characteristics of the redesigned engine; etc. during the design of several different types of turboprop or turboshaft engines, including the turboprop and turboshaft engines described below with reference to FIGS. 13 through 17, FIGS. 20 through 34, and the embodiments listed in the Table of FIG. 18.

Referring now to the drawings, wherein identical numerals indicate the same elements throughout the figures, FIG. 13 is a schematic cross-sectional view of a gas turbine engine in accordance with an exemplary embodiment of the present disclosure. More particularly, for the embodiment of FIG. 13, the gas turbine engine is a turboprop engine 610. As shown in FIG. 13, turboprop engine 610 defines an axial direction A (extending parallel to a longitudinal centerline or central axis 612 provided for reference), a radial direction R, and a circumferential direction C (not shown) disposed about the axial direction A. Turboprop engine 610 generally includes a propeller section 614 and a turbomachine 616 disposed aft of the propeller section 614 from an aircraft perspective, the propeller section 614 being operable with, and driven by, turbomachine 616.

The turbomachine 616 includes, in a serial flow relationship, a booster compressor 618, a high pressure (HP) compressor 622, a combustion section 626, a high pressure (HP) turbine 628, an intermediate pressure (IP) turbine 620, a low pressure (LP) turbine 630, and an exhaust section 632. An air flow path generally extends through booster compressor 618, HP compressor 622, combustion section 626, HP turbine 628, IP turbine 620, LP turbine 630, and exhaust section 632 which are in fluid communication with each other.

In at least certain exemplary embodiments, the HP turbine 628 can include at least two stages of HP turbine rotor blades. Such a configuration may ensure a sufficient amount of power is provided to the HP compressor 622.

An HP shaft or spool 634 drivingly connects the HP turbine 628 to the HP compressor 622. An IP shaft 635 drivingly connects the IP turbine 620 to the booster compressor 618. An LP shaft or spool 636 drivingly connects the

LP turbine 630 to propeller section 614 of the turboprop engine 610. The turbomachine 616 includes a drive turbine drivingly coupled to a drive turbine shaft configured to provide an output torque to, e.g., the fan assembly 614 in the embodiment shown. For the embodiment depicted, the drive turbine is the LP turbine 630 and the drive turbine shaft is the LP shaft 636.

For the embodiment depicted, propeller section 614 includes a variable pitch propeller 638 having a plurality of propeller blades 640 coupled to a disk 642 in a spaced apart manner. As depicted, the propeller blades 640 extend outwardly from disk 642 generally along the radial direction R. Each propeller blade 640 is rotatable relative to the disk 642 about a pitch axis P by virtue of the propeller blades 640 being operatively coupled to a suitable actuation member 644 configured to collectively vary the pitch of the propeller blades 640, e.g., in unison. The propeller blades 640, disk 642, and actuation member 644 are together rotatable about a fan centerline 645 by LP shaft 636 across a power gear box 646. The power gear box 646 includes a plurality of gears for stepping down the rotational speed of the LP shaft 636 to a more efficient rotational fan speed and is attached to one or both of a core frame or a fan frame through one or more coupling systems. Additionally, for the embodiment shown, the power gear box 646 is an offset gear box, such that the fan axis 645 is offset from the longitudinal centerline axis 612 of the turbomachine 616.

During operation of the turboprop engine 610, a volume 30 of air 650 (also referred to as a free stream flow of air prior to its encounter with the propeller 638) passes through blades 640 of propeller 638 and is urged toward an inlet 652 of turbomachine 616. More specifically, turboprop engine 610 includes an intake channel 654 that extends from the 35 inlet 652, which is non-axisymmetric with respect to longitudinal centerline axis 612, to the booster compressor 618, where the channel 654 is axisymmetric with respect to longitudinal centerline axis 612.

The booster compressor 618 and HP compressor 622 each 40 include one or more sequential stages of compressor stator vanes, one or more sequential stages of compressor rotor blades, an impeller, or combinations thereof. In particular, the booster compressor 618 is depicted as an axial compressor (having multiple stages of compressor stator vanes and 45 rotor blades) and the HP compressor 622 is depicted as a centrifugal compressor (having an impeller).

Though the illustrated embodiment includes both axial and centrifugal flow compressors, in some forms the turboprop engine 610 can include just an axial flow compressor(s) or centrifugal flow compressor(s).

The HP compressor 622 directs compressed air into combustion section 626 where the air mixes with fuel. Combustion section 626 includes a combustor which combusts the air/fuel mixture to provide combustion gases. The combustion gases flow through HP turbine 628, IP turbine 620, and LP turbine 630. Each of these HP, IP, and LP turbines 628, 620, 630 includes one or more sequential stages of turbine stator vanes and one or more sequential stages of turbine rotor blades. The turbine rotor blades are 55 coupled to a respective one of the HP shaft 634, IP shaft 635, or LP shaft 636 to extract thermal and/or kinetic energy from the combustion gasses flowing therethrough. The energy extraction from HP turbine 628 supports operation of HP compressor 622 through HP shaft 634, the energy extraction from IP turbine 620 supports operation of booster compressor 618 through IP shaft 635, and the energy extraction from LP turbine 630 supports operation of propeller section 614

through LP shaft 636 (across the power gear box 646). Combustion gases exit turboprop engine 610 through exhaust section 632.

In other exemplary embodiments, the turbine engine may include any suitable number of compressors, turbines, shafts, etc. For example, as will be appreciated, HP shaft 634 and LP shaft 636 may further be coupled to any suitable device for any suitable purpose. For example, in certain exemplary embodiments, turboprop engine 610 of FIG. 13 may be utilized in aeroderivative applications. Additionally, in other exemplary embodiments, turboprop engine 610 may include any other suitable type of combustor, such as a reverse flow combustor.

The embodiment of turboprop engine 610 illustrated in FIG. 13 further includes an intercooler assembly 700. In particular, referring now to FIG. 14, providing a close-up view of a portion of the turbomachine 616 of FIG. 13, the intercooler assembly 700 includes an intercooler heat exchanger 702 in thermal communication with the compressor section of the turbomachine 616. For the embodiment depicted, the intercooler heat exchanger 702 is positioned in thermal communication with the compressor section at a location downstream of the booster compressor 618 and upstream of the HP compressor 622.

It will be appreciated, however, that in other exemplary embodiments, the intercooler heat exchanger 702 may additionally or alternatively be positioned in thermal communication with the compressor section at a location within the booster compressor 618 and/or within the HP compressor 622 (e.g., an inter-stage heat exchanger).

Referring still to the embodiment of FIG. 14, it will be appreciated that the turbomachine 616 defines a working gas flowpath 704 extending from the inlet 652 (see FIG. 13), through the compressor section, combustion section 626, turbine section, and exhaust section 632 (see FIG. 13). More specifically, for the embodiment depicted, the intercooler heat exchanger 702 is positioned within the working gas flowpath 704 through the compressor section at a location downstream of the booster compressor 618 and upstream of the HP compressor 622.

The intercooler assembly 700 is configured to cool the airflow through the compressor section, allowing the compressor section to define a higher overall pressure ratio without a compressor exit temperature (and downstream temperatures such as turbine inlet temperature and exhaust gas temperature) exceeding operability thresholds. The intercooler heat exchanger 702 is accordingly in fluid communication with a cooling fluid source through an inlet line 706 and an outlet line 708. The inlet line 706 may provide a cooling fluid to the intercooler heat exchanger 702, whereby the cooling fluid may accept heat from the airflow through the working gas flowpath 704 between the booster compressor 618 and HP compressor 622. The outlet line 708 may provide the heated cooling fluid away from the intercooler heat exchanger 702.

The cooling fluid source may be a thermal transport bus (see thermal transport bus 800 of FIG. 15, below) utilizing a supercritical thermal fluid (such as supercritical CO₂, supercritical N₂, or a commercial refrigerant/transfer fluid).

Additionally, or alternatively, the cooling fluid source may be any other suitable cooling fluid source, such as a cooled cooling air, a pressurized bypass or freestream air-flow, a fuel flow, or the like.

For example, in at least certain exemplary embodiments, the cooling fluid source may be a fuel flow. For example, the turboprop engine 610 may include a fuel system that is a cryogenic fuel system, such as a cryogenic hydrogen fuel

(i.e., configured to store a liquid hydrogen). With such an exemplary embodiment, the intercooler heat exchanger 702 may be in thermal communication with a liquid hydrogen, creating a large temperature differential between the cooling fluid (liquid hydrogen) and the airflow through the compressor section to be cooled. Such may allow for a relatively compact intercooler heat exchanger 702 with a relatively low pressure drop for a given amount of desired temperature reduction of the airflow through the compressor section. At the same time, the heat extracted from the airflow through the compressor section may be used to help vaporize the liquid hydrogen prior to combustion.

In particular, in certain exemplary embodiments, the cooling fluid source for the intercooler heat exchanger 702 can be a thermal transport bus having a conduit with a flow of thermal transport fluid therethrough. More specifically, referring now briefly to FIG. 15, a schematic view of a thermal transport bus 800 as may be utilized with the turboprop engine 610 described above with reference to FIGS. 13 and 14 is provided.

The thermal transport bus 800 includes an intermediary heat exchange fluid flowing therethrough and is formed of one or more suitable fluid conduits 804. 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 N₂, or a supercritical CO₂. Utilizing a supercritical fluid may allow for an intercooler assembly 700 to include an intercooler heat exchanger 702 that has a relatively low pressure drop for a given amount of heat transfer, allowing in a more efficient compressor section while still achieving a desirably low compressor exit temperature.

The exemplary thermal transport bus 800 includes a pump 806 in fluid communication with the heat exchange fluid in the thermal transport bus 800 for generating a flow of the heat exchange fluid in/through the thermal transport bus 800.

Moreover, the exemplary thermal transport bus 800 includes one or more heat source exchangers 808 in thermal communication with the heat exchange fluid in the thermal transport bus 800. Specifically, the thermal transport bus 800 depicted includes a plurality of heat source exchangers 808. The plurality of heat source exchangers 808 are configured to transfer heat from, e.g., one or more of accessory systems of the turboprop engine 610 to the heat exchange fluid in the thermal transport bus 800. For example, in certain exemplary embodiments, the plurality of heat source exchangers 808 may include one or more of: an intercooler heat exchanger (such as the exemplary intercooler heat exchanger 702 of FIG. 14); 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.

For the embodiment depicted, there are three heat source exchangers 808. The heat source exchangers 808 are each

arranged in series flow along the thermal transport bus 800. However, in other exemplary embodiments, any other suitable number of heat source exchangers 808 may be included and one or more of the heat source exchangers 808 may be arranged in parallel flow along the thermal transport bus 800 (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 808 in thermal communication with the heat exchange fluid in the thermal transport bus 800, or alternatively, there may be at least two heat source exchangers 808, at least four heat source exchangers 808, at least five heat source exchangers 808, or at least six heat source exchangers 808, and up to twenty heat source exchangers 808 in thermal communication with heat exchange fluid in the thermal transport bus 800.

Additionally, the exemplary thermal transport bus 800 of FIG. 17 further includes one or more heat sink exchangers 810 permanently or selectively in thermal communication with the heat exchange fluid in the thermal transport bus 800. The one or more heat sink exchangers 810 are located downstream of the plurality of heat source exchangers 808 and are configured for transferring heat from the heat exchange fluid in the thermal transport bus 800, e.g., to atmosphere, to fuel, to a fan stream, etc. For example, in certain embodiments the one or more heat sink exchangers 810 may include at least one of a RAM heat sink exchanger, a fuel heat sink exchanger, a bypass passage heat sink exchanger, a bleed air 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. As noted above, the fuel may be a cryogenic fuel, such as a liquid hydrogen fuel.

Moreover, the other heat sink exchanger may generally be an “air to heat exchange fluid” heat exchanger which transfers heat from the heat exchange fluid to an airflow.

For the embodiment of FIG. 15, the one or more heat sink exchangers 810 of the thermal transport bus 800 depicted includes a plurality of individual heat sink exchangers 810. More particularly, for the embodiment of FIG. 17, the one or more heat sink exchangers 810 include three heat sink exchangers 810 arranged in series. The three heat sink exchangers 810 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 810 may include any other suitable number and/or type of heat sink exchangers 810. For example, in other exemplary embodiments, a single heat sink exchanger 810 may be provided, at least two heat sink exchangers 810 may be provided, at least four heat sink exchangers 810 may be provided, at least five heat sink exchangers 810 may be provided, or up to twenty heat sink exchangers 810 may be provided. Additionally, in still other exemplary embodiments, two or more of the one or more heat sink exchangers 810 may alternatively be arranged in parallel flow with one another.

Referring still to the exemplary embodiment depicted in FIG. 15, one or more of the plurality of heat sink exchangers 810 and one or more of the plurality of heat source exchangers 808 are selectively in thermal communication with the heat exchange fluid in the thermal transport bus 800. More particularly, the thermal transport bus 800 depicted includes a plurality of bypass lines 812 for selectively bypassing each heat source exchanger 808 and each heat sink exchanger 810 in the plurality of heat sink exchangers 810. Each bypass line 812 extends between an upstream juncture 814 and a downstream juncture 816—the upstream juncture 814

located just upstream of a respective heat source exchanger 808 or heat sink exchanger 810, and the downstream juncture 816 located just downstream of the respective heat source exchanger 808 or heat sink exchanger 810.

Additionally, each bypass line 812 meets at the respective upstream juncture 814 with the thermal transport bus 800 via a three-way valve 818. The three-way valves 818 each include an inlet fluidly connected with the thermal transport bus 800, a first outlet fluidly connected with the thermal transport bus 800, and a second outlet fluidly connected with the bypass line 812. The three-way valves 818 may each be a variable throughput three-way valve, such that the three-way valves 818 may vary a throughput from the inlet to the first and/or second outlets. For example, the three-way valves 818 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 818 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.

Notably, the three-way valves 818 may be in operable communication with a controller of an engine including the thermal transport bus 800 (e.g., turboprop engine 610 of FIGS. 13 and 14).

Further, each bypass line 812 also meets at the respective downstream juncture 816 with the thermal transport bus 800. Between each heat source exchanger 808 or heat sink exchanger 810 and downstream juncture 816, the thermal transport bus 800 includes a check valve 820 for ensuring a proper flow direction of the heat exchange fluid. More particularly, the check valve 820 prevents a flow of heat exchange fluid from the downstream juncture 816 towards the respective heat source exchanger 808 or heat sink exchanger 810.

Referring now to FIG. 16, a close-up view of an aft-most compression stage of the exemplary HP compressor 622 of FIGS. 13 and 14 is provided.

As will be appreciated, the HP compressor 622 is a centrifugal compressor, such that the aft-most compression stage is an impeller 902. The impeller 902 includes a leading edge 904 and a trailing edge 906 and a base 908 to which the impeller 902 is coupled. The base 908 may be coupled to the HP shaft 634 to drive rotation of the impeller 902.

The impeller 902 extends into the working gas flowpath 704. At the trailing edge 906 of the impeller 902, the working gas flowpath 704 is defined between the base 908 and a liner 910.

The turboprop engine 610 further defines a reference plane 912 intersecting with an aft-most point of the trailing edge 906 of the impeller 902 depicted, the reference plane 230 being orthogonal to an airflow direction 914 out of the HP compressor 622. In the embodiment depicted, the airflow direction 914 is along the radial direction R.

Further, the HP compressor 622 defines a high pressure compressor exit area (A_{HPCExi}) within the reference plane 912. More specifically, the HP compressor 622 defines a flowpath height (H_F) in a direction orthogonal to the airflow direction 914, and more specifically within the reference plane 912 and parallel to the axial direction A for the embodiment shown. The flowpath height (H_F) is defined between the liner 910 and the base 908 at the downstream-most portion of the trailing edge 906. Notably, the downstream-most portion of the trailing edge 906 further defines

a radius R_{TE} . The HP compressor 622 exit area is defined for the embodiment depicted according to Expression (3):

$$A_{HPCExit} = 2\pi R_{TE} \times H_F \quad \text{Expression (3)}$$

Briefly, it will be appreciated that in certain exemplary embodiments, the HP compressor 622 may be configured as an axial compressor, such that a compressor rotor blade is positioned at an exit of the HP compressor 622. In such an embodiment the high pressure compressor exit area ($A_{HPCExit}$) may be define according to an equation similar to Expression (4), below.

The inventors of the present disclosure have found that for a given total output power of the turboprop engine 610, a decrease in size of the high pressure compressor exit area ($A_{HPCExit}$) may generally relate in an increase in a compressor exit temperature (i.e., a temperature of the airflow through the working gas flowpath 704 at the reference plane 912), a turbine inlet temperature (i.e., a temperature of the airflow through the working gas flowpath 704 provided to a first stage of rotor blades of the HP turbine 628; see FIG. 13), and the maximum exhaust gas temperature (EGT). In particular, the inventors of the present disclosure have found that the high pressure compressor exit area ($A_{HPCExit}$) may generally be used as an indicator of the above temperatures to be achieved by the engine 700 during operation for a given power output of the engine 700.

As will further be appreciated, a total, or rather a maximum, power output of the turboprop engine 610 can generally be determined as a function of a maximum drive turbine shaft torque (T_{OUT}) in Newton meters (N-m) and a maximum rotational speed of the LP turbine 630, in revolutions per minute (rpm). The maximum rotational speed of the LP turbine 630 is limited by a drive turbine exit area (A_{DTExit}) in square inches, as the higher drive turbine exit areas (A_{DTExit}) push the bounds of the strength to weight properties forming the LP turbine 630 (measured as a function of area (A) times speed (N) squared; AN^2).

In particular, referring now to FIG. 17, it will be appreciated that the LP turbine 630 defines the drive turbine exit areas (A_{DTExit}). FIG. 17 provides a close-up view of an LP turbine rotor blade 920 in an aftmost stage 922 of LP turbine rotor blades 920. As will be appreciated, the LP turbine rotor blade 920 includes a trailing edge 924 and the aftmost stage 922 of LP turbine rotor blades 920 includes a rotor 926 having a base 928 to which the LP turbine rotor blade 920 is coupled. The base 928 includes a flowpath surface 930 defining in part the working gas flow path 704 through the LP turbine 630. Moreover, the LP turbine 630 includes a shroud or liner 932 located outward of the LP turbine rotor blade 920 along the radial direction R. The shroud or liner 932 also includes a flowpath surface 934 defining in part the working gas flow path 704 through the LP turbine 630.

The turboprop engine 610 (FIG. 13) defines a reference plane 936 intersecting with an aft-most point of the trailing edge 924 of the LP turbine rotor blade 920 depicted, the reference plane 936 being orthogonal to an airflow direction 938 out the LP turbine 630. The airflow direction 938 out the LP turbine 630 is parallel to the axial direction A in the embodiment depicted. Further, the LP turbine 630 defines the drive turbine exit area (A_{DTExit}) within the reference plane 936. More specifically, the LP turbine 630 defines an inner radius (R_{INNER}) extending along the radial direction R within the reference plane 936 from the longitudinal centerline axis 612 to the flowpath surface 930 of the base 928 of the rotor 926 of the aftmost stage 922 of LP turbine rotor blades 920, as well as an outer radius (R_{OUTER}) extending along the radial direction R within the reference plane 936

from the longitudinal centerline axis 612 to the flowpath surface 934 of the shroud or liner 932. The LP turbine 630 exit area is defined according to Expression (4):

$$A_{DTExit} = \pi(R_{OUTER}^2 - R_{INNER}^2) \quad \text{Expression (4)}$$

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 power outputs (which is a function of a maximum drive turbine shaft torque and maximum rotational speed; the maximum rotational speed being limited by a drive turbine exit area), maximum exhaust gas temperatures, and supporting technology characteristics and evaluating an overall engine performance and other qualitative turbofan engine characteristics—a significant relationship between a high pressure compressor exit area, total power output (and drive turbine exit area and maximum low pressure torque), and maximum exhaust gas temperature that enables increased engine core operating temperatures and overall engine propulsive/thermal efficiency. The relationship can be thought of as an indicator of the ability of a turboprop or turboshaft 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 power output as represented by drive turbine exit area and maximum drive turbine shaft torque, and without overly detrimentally affecting overall engine performance and other qualitative engine characteristics. The relationship applies to an engine that incorporates an intercooler assembly, incorporates a cooled cooling air system, builds portions of the core using material capable of operating at higher temperatures, or a combinations thereof. Significantly, the relationship ties the core size (as represented by the exit area of the higher pressure compressor) to the desired output power and exhaust gas temperature associated with the desired propulsive/thermal efficiency and practical limitations of the engine design, as described below.

During the design and evaluation of various turboprop and turboshaft engines, the inventors unexpectedly discovered that the integration of these advanced technologies could lead to a net benefit in engine performance, despite the anticipated costs. This discovery was made while exploring different engine architectures, power classes, and mission requirements, and assessing the impact on overall engine performance and qualitative characteristics. The inventors found that the benefits of operating at higher temperatures and pressures, which were previously thought to be cost-prohibitive, could indeed outweigh the costs when the core size is maintained or reduced.

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. Similar results were discovered with the introduction of an intercooler assembly. Referring to the case of utilizing more temperature-resistant material, such as a Ceramic 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. Regardless of the technology, however, the relationship now described can apply to identify the interrelated operating conditions and core size—i.e., maximum drive turbine shaft torque, drive

turbine exit area, maximum exhaust gas temperature, and high pressure compressor exit area.

The inventors of the present disclosure discovered bounding the relationship between a product of total power output (which is a function of a maximum drive turbine shaft torque and maximum rotational speed; the maximum rotational speed being limited by a drive turbine exit area) squared and maximum exhaust gas temperature, divided by the high pressure compressor exit area (corrected specific power) 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 output power 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 be 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.

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 into 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.

The desired relationship, referred to herein as a Corrected Specific Power (CSP) of a gas turbine engine, providing for the improved gas turbine engine, discovered by the inventors, is expressed as:

$$\left(\frac{T_{OUT}}{\sqrt{A_{DTExit}}} \right)^2 * \frac{EGT}{A_{HPCExit}} * 10^{-11}. \quad \text{Expression (5)}$$

where T_{OUT} is maximum drive turbine shaft torque in Newton meters; EGT is maximum exhaust gas temperature in degrees Celsius; $A_{HPCExit}$ is the high pressure compressor exit area in square meters; and A_{DTExit} is the drive turbine exit area in square meters. As mentioned earlier, the maximum power output of the gas turbine engine is a function of the maximum drive turbine shaft torque (T_{OUT}) and a rotational speed of the LP shaft. The rotational speed of the LP shaft is limited by a size of the LP turbine, as the materials forming the LP turbine have a strength to weight relationship that limits how big the LP turbine can be and how fast the LP turbine can rotate (measured as a function of area (A) times speed (N) squared: AN^2). Expression (5), above, and the limits of CSP identified herein identify a design space where an improved power output is sufficiently high to justify the means for accommodating the increased energy in the flow. The improved power output is more specifically realized through increases in overall pressure ratio and/or EGT, which results in more energy in a flow through the turbine section, allowing for more power extraction by the turbine section for a given engine size. Said

another way, application of Expression (5) yields an engine that allows for more maximum drive turbine shaft torque (T_{OUT}) for a given LP turbine size (represented by the drive turbine exit area (A_{DTExit})).

Referring now to FIGS. 18 and 19, various exemplary gas turbine engines are illustrated in accordance with one or more exemplary embodiments of the present disclosure. In particular, FIG. 18 provides a table including numerical values corresponding to several of the plotted gas turbine engines in FIG. 19. FIG. 19 is a plot 1000 of gas turbine engines in accordance with one or more exemplary embodiments of the present disclosure, showing the CSP on a Y-axis 1002 and the EGT on an X-axis 1004.

As shown, the plot 1000 in FIG. 19 depicts a first range 1006, wherein CSP is greater than $0.0001194 \times EGT^2 - 0.103 \times EGT + 22.14$ and less than $0.0003294 \times EGT^2 - 0.3061 \times EGT + 77.91$. The stated advantages of CSP are valid only when the engine design falls within these upper and lower bounds and the EGT is between 525 and 1250. The units of 10 the CSP values are Newtons squared times degrees Celsius over meters squared. These values represent the optimized range for achieving the desired thermal efficiency and power density. An engine with a CSP within this range is expected to exhibit higher thermal efficiency, as indicated by a lower 20 specific fuel consumption (SFC).

In addition, FIG. 19 depicts a second range 1010 where CSP is greater than 3.3 and less than 101 and EGT is greater than 600 degrees Celsius and less than 1,000 degrees Celsius; and a third range 1008 where CSP is greater than or 30 equal to 4 and less than or equal to 69 and EGT is greater than 700 degrees Celsius and less than 900 degrees Celsius. For this second range 1010 and third range 1008, the gas 35 turbine engines can be of a lower thrust class (e.g., a turbomachine of the gas turbine engine providing an output power of at least 550 horsepower and up to 2,000 horsepower when operated at a takeoff power level), while also having a compressor section of the turbomachine defining an increased overall pressure ratio (OPR; e.g., greater than 14:1 and less than 22:1). Such an arrangement can allow for the 40 increased power output for a given engine size. For example, an increased torque output of the LP turbine of the turbomachine can be achieved with such an increased OPR by incorporating an intercooling assembly. Designing a gas 45 turbine engine having a CSP and EGT within the second range 1010 or third range 1008, can allow the gas turbine engines in the noted thrust class and defining the noted OPRs to provide improved overall performance and thermal efficiencies.

It will be appreciated that although the discussion above 50 is generally related to a turboprop engine having a particular intercooler assembly and a particular engine architecture, in various embodiments of the present disclosure, the relationship outlined above with respect to Expression (5) may be applied to any other suitable engine architecture, including 55 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 and high pressure turbine exit area, while maintaining or even increasing the maximum engine power output without, e.g., prematurely wearing various components within the turbomachine exposed to the working gas flowpath.

For example, reference will now be made to FIG. 20. FIG. 20 provides a schematic view of a turbomachine 616 in accordance with another exemplary embodiment of the 60 present disclosure. The turbomachine 616 of FIG. 20 may be configured in substantially the same manner as the exemplary turbomachine 616 described above with reference to

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FIGS. 13 and 14. Accordingly, the same or similar numbers may refer to the same or similar parts.

For example, the turbomachine 616 generally includes a compressor section, a combustion section 626, and a turbine section arranged in serial flow order. However, for the embodiment of FIG. 20, the compressor section includes a single, HP compressor 622 (i.e., no booster compressor 618) and the turbine section does not include an intermediate pressure turbine 620. Notably, with the exemplary embodiment of FIG. 20, the HP compressor 622 is configured as an axial-centrifugal compressor, including stages of compressor rotor blades and stator vanes and an impeller.

Further by way of example, reference will now be made to FIGS. 21 through 23. FIGS. 21 through 23 provide schematic views of three additional turbomachines 616 in accordance with exemplary embodiments of the present disclosure. Each of these turbomachines 616 may be configured in substantially the same manner as the exemplary turbomachine 616 described above with reference to FIGS. 13 and 14. Accordingly, the same or similar numbers may refer to the same or similar parts.

For example, each turbomachine 616 generally includes a compressor section, a combustion section 626, and a turbine section arranged in serial flow order. However, referring particularly to the embodiment of FIG. 21, the compressor section again includes a single, HP compressor 622 (i.e., no booster compressor 618) and the turbine section does not include an intermediate pressure turbine 620. Notably, with the exemplary embodiment of FIG. 21, the HP compressor 622 is configured as a multi-stage, centrifugal compressor, including two stages of impellers (each driven by an HP turbine 628).

Referring to FIGS. 22 and 23, the compressor sections each include a booster compressor 618 in addition to the HP compressor 622, with both the booster compressor 618 and HP compressor 622 each configured as a centrifugal compressor. The turbine section of each of the turbomachines 616 of FIGS. 22 and 23 further includes an intermediate pressure turbine 620 drivingly coupled to the respective booster compressor 618 through a respective intermediate pressure shaft 635.

Further, for the embodiment of FIG. 23, the gas turbine engine 610 further includes an intercooler assembly 700, the intercooler assembly 700 having an intercooler heat exchanger 702 in thermal communication with the compressor section. For the embodiment of FIG. 23, the compressor section defines a transition zone 1050 downstream of the booster compressor 618 and upstream of the HP compressor 622. The transition zone 1050 defines a portion of a working gas flowpath 704 of the turbomachine 616 extending between the booster compressor 618 and the HP compressor 622. The intercooler heat exchanger 702 may be incorporated into the transition zone 1050, such as into one or more of the lines, frames, or the like of the transition zone 1050. The intercooler assembly 700 of FIG. 23, and in particular the intercooler heat exchanger 702, may include or otherwise be in thermal communication with a cooling fluid source.

However, in other embodiments, the intercooler heat exchanger 702 may be positioned within the working gas flowpath 704, e.g., having one or more fins, plates, or other heat transfer structures within the working gas flowpath 704. The intercooler heat exchanger 702 may extend in the circumferential direction, arranged as a 360 degree, axisymmetric heat exchanger.

Further by way of example, reference will now be made to FIGS. 24 through 26. FIGS. 24 through 26 provide

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schematic views of three additional turbomachines 616 in accordance with exemplary embodiments of the present disclosure. Each of these turbomachines 616 may be configured in substantially the same manner as the exemplary turbomachine 616 described above with reference to FIGS. 21 through 23, respectively. Accordingly, the same or similar numbers may refer to the same or similar parts.

For example, each turbomachine 616 generally includes a compressor section, a combustion section 626, and a turbine section arranged in serial flow order. However, by contrast to the exemplary turbomachines 616 of FIGS. 21 through 23, for the embodiments of FIGS. 24 through 26, the HP compressor 622 is an axial-centrifugal compressor, including stages of compressor rotor blades and stator vanes and an impeller. Briefly, referring particularly to FIG. 24, the HP compressor 622 includes a first stage impeller, one or more intermediate stages of compressor rotor blades and stator vanes, and an aft-most stage impeller.

Further by way of example, reference will now be made to FIGS. 27 and 28. FIGS. 27 and 28 provide schematic views of two additional turbomachines 616 in accordance with exemplary embodiments of the present disclosure. Each of these turbomachines 616 may be configured in substantially the same manner as the exemplary turbomachines 616 described above with reference to FIGS. 22 and 23, respectively. Accordingly, the same or similar numbers may refer to the same or similar parts.

For example, each turbomachine 616 generally includes a compressor section, a combustion section 626, and a turbine section arranged in serial flow order. However, by contrast to the exemplary turbomachines 616 of FIGS. 22 through 23, for the embodiments of FIGS. 27 and 28, the booster compressor 618 is an axial-centrifugal compressor, including stages of compressor rotor blades and stator vanes and an impeller.

Referring now to FIG. 29, a turboprop engine 610 in accordance with another exemplary embodiment of the present disclosure is provided. The exemplary turboprop engine 610 of FIG. 29 may be configured in a similar manner as the exemplary turboprop engine 610 of FIGS. 13 and 14, and accordingly, the same or similar numbers may refer to the same or similar parts.

For example, the turboprop engine 610 of FIG. 29 generally includes a fan assembly 614 and a turbomachine 616, with the fan assembly 614 being driven by the turbomachine 616 across a power gear box 646. However, for the embodiment of FIG. 29, the turboprop engine 610 is configured as a reverse flow engine. In such a manner, it will be appreciated that the turboprop engine 610 is characterized by a general relationship between the direction of the flow of incoming air 650 (such direction can be used to characterize the relative motion of air during a mode of operation of the engine 610 such as a forward thrust mode) and that of a flow of air axially through the turboprop engine 610. The flow of air through the turbomachine 616 is generally reverse to that of the flow of incoming air 650. Turning the flow from the direction of the incoming flow of air 650 to the axial direction through the turbomachine 616 is usually performed by an intake channel 654. The change of direction is reversed in that the bulk direction of the flow of air 650 (itself having a circumferential swirl component imparted by the propeller blades 640 in addition to a longitudinal component) is opposite, or reverse, to the bulk direction of air flow axially through the turbomachine 616 (which itself also includes a longitudinal component but also include radial and circumferential components owing to the shape of the flow path and swirl induced by rotating turbomachinery

components) during one or more phases of operation of the turbomachine 616. Thus, it will also be appreciated that the term “reverse” is a relative comparison of the longitudinal components of the bulk flow of air 650 and bulk flow of air axially within the engine 610. Though the longitudinal direction of the flow of air 650 may not be perfectly parallel with the axial flow of air through the engine 610, it will be appreciated that the longitudinal components of the directions of the flow of air 650 and the axial flow are reversed.

Moreover, it will be appreciated that for the embodiment depicted, an exhaust section 632 is not axi-symmetric, and instead include one or more exhaust outlets 633 oriented on, e.g., one or more sides of the turboprop engine 610, such as opposing sides as in the embodiment depicted in FIG. 29.

Further, for the embodiment of FIG. 29, the power gear box 646 is not an offset power gear box, such that a fan axis 645 is aligned with a longitudinal centerline axis 612 of the turbomachine 616.

In addition, the exemplary turboprop engine 610 of FIG. 29 includes an intercooler assembly 700, the intercooler assembly 700 having an intercooler heat exchanger 702 in thermal communication with the compressor section. For the embodiment of FIG. 29, the compressor section defines a transition zone 1050 downstream of the booster compressor 618 and upstream of the HP compressor 622. The transition zone 1050 defines a portion of a working gas flowpath 704 of the turbomachine 616 extending between the booster compressor 618 and the HP compressor 622. The intercooler heat exchanger 702 is, however, for the embodiment shown located externally from the turbomachine 616 (e.g., in a pylon, wing, or fuselage of an aircraft incorporating the turboprop engine 610). The intercooler assembly 700 includes an outflow duct 720 extending from the compressor section to the externally-located intercooler heat exchanger 702, and an inflow duct 722 extending from the intercooler heat exchanger 702 back to the compressor section. In the embodiment depicted, the outflow duct 720 and the inflow duct 722 are each in airflow communication with the compressor section via a transition zone 1050.

The intercooler assembly 700 of FIG. 29, and in particular the intercooler heat exchanger 702, may include or otherwise be in thermal communication with a cooling fluid source (e.g., through inlet and outlet lines 706, 708). Including the intercooler heat exchanger 702 at the location external to the turbomachine 616 may allow the intercooler heat exchanger 702 to be positioned closer to the cooling fluid source. In situations where the cooling fluid source is a flow of cryogenic fuel, such as liquid hydrogen, it can be more efficient to redirect the airflow with less redirecting of the fuel.

Further, still, referring now to FIG. 30, a turboprop engine 610 in accordance with yet another exemplary embodiment of the present disclosure is provided. The exemplary turboprop engine 610 of FIG. 30 may be configured in a similar manner as the exemplary turboprop engine 610 of FIGS. 13 and 14, and accordingly, the same or similar numbers may refer to the same or similar parts.

However, for the embodiment of FIG. 30, the turboprop engine 610 is arranged in a “pusher” configuration, such that a fan assembly 614 is positioned aft of a turbomachine 616 of the turboprop engine 610. With such an exemplary aspect, a low pressure shaft 636 of the turbomachine 616 need not extend concentrically with, e.g., a high pressure shaft 634, and instead may extend aft directly to a power gear box 646 and the fan assembly 614. Such a configuration can allow for more desirably mechanical properties of the turboprop engine 610.

Referring now to FIG. 31, a turboprop engine 610 in accordance with still another exemplary embodiment of the present disclosure is provided. The turbomachine 616 of FIG. 31 may be configured in a similar manner as the exemplary turboprop engine 610 of FIGS. 13 and 14, and accordingly, the same or similar numbers may refer to the same or similar parts.

However, it will be appreciated that for the embodiment of FIG. 31, the turbomachine 616 does not include an intercooler assembly 700, and instead the turbomachine 616 includes a Cooled Cooling Air (“CCA”) system 1100. The CCA system 1100 generally includes a CCA heat exchanger 1102, a cold fluid delivery assembly 1104, and a hot air bleed assembly 1106.

The cold fluid delivery assembly 1104 generally includes a cold fluid inlet duct 1108 and a cold fluid outlet duct 1110. The cold fluid inlet duct 1108 is in airflow communication with a cold fluid source 1112 and the cold fluid outlet duct 1110 is in airflow communication with a cold fluid sink 1114. The cold fluid inlet duct 1108 is configured to provide a cooling fluid, such as a cooling airflow, from the cold fluid source 1112 to the CCA heat exchanger 1102, and the cold fluid outlet duct 1110 is configured to receive the cooling fluid, such as the cooling airflow, from the CCA heat exchanger 1102 and exhaust it to the cold fluid sink 1114. In certain exemplary embodiments, the cold fluid source 1112 may be an airflow over the turbomachine 616 (e.g., a propeller stream), a bleed airflow from the compressor section, a fuel flow, etc.

It will be appreciated that as used herein, the term “cold fluid” in the context of the cold fluid delivery assembly 1104 refers to a fluid at a temperature lower than a temperature of an airflow received through the hot air bleed assembly 1106. Accordingly, the term is a relative term and does not imply or require any absolute temperature.

The hot air bleed assembly 1106 includes a hot air bleed duct 1116 in airflow communication with a working gas flowpath 704 through the turbomachine 616 at a downstream end of the compressor section, or a location downstream of the compressor section and upstream of a combustor of the combustion section 626. The hot air bleed assembly 1106 further includes a CCA delivery duct 1118 in thermal communication with a hot component of the turbomachine 616. In particular, in the embodiment depicted, the CCA delivery duct 1118 is in thermal communication with a first stage of turbine rotor blades of the HP turbine 628 of the turbomachine 616.

Additionally, or alternatively, in other exemplary embodiments, the CCA delivery duct 1118 may be in thermal communication with an aft-most stage of the HP compressor 622, a rotor at the aft-most stage of the HP compressor 622, a sump within the turbine section, a rotor of the HP turbine 628, one or more airfoils through the HP turbine 628, or other hot components of the turbomachine 616.

During operation, high pressure airflow is bled through the hot air bleed duct 1116 and provided to the CCA heat exchanger 1102, where heat from the high pressure airflow is transferred to the cooling airflow through the cold fluid delivery assembly 1104. The cooled high pressure airflow from the CCA heat exchanger 1102 is then provided through the CCA delivery duct 1118 to the hot component, to cool the hot component.

As is depicted in phantom, in certain exemplary embodiments, the hot air bleed assembly 1106, and in particular the hot air bleed duct 1116, may be configured to receive bleed air from one or more locations upstream of the location between the HP compressor 622 and the combustor.

In at least certain exemplary embodiments, when the gas turbine engine 610 is operated at a takeoff power level, the CCA system 1100 is configured to provide a temperature reduction of the cooling airflow (i.e., the airflow through the hot air bleed assembly 1106) equal to at least 15% of the EGT and up to 45% of the EGT. Further, when the gas turbine engine 610 is operated at the takeoff power level, the CCA system 1100 is configured to receive between 2.5% and 35% of an airflow through a working gas flowpath 704 of the turbomachine 616 at an inlet to a first compressor of the compressor section (the HP compressor 622 in the embodiment depicted).

Inclusion of the CCA system 1100 may allow for the turbomachine 616 to operate with a higher EGT, higher overall pressure ratios through the compressor section, or both, to define a CSP within one or more of the ranges described above.

Notably, however, in other exemplary embodiments, one or more turbomachines 616 of the present disclosure can have other suitable configurations. For example, referring now to FIG. 32, a CCA system 1100 is provided where a hot air bleed duct 1116 of the CCA system 1100 is in airflow communication with a booster compressor 618 upstream of an HP compressor 622. The hot air bleed duct 1116 includes a pump 1120 to increase a pressure of the airflow through the hot air bleed duct 1116 to enable the airflow to be provided, e.g., to the first stage of turbine rotor blades of the HP turbine 628.

Moreover, it will be appreciated that the exemplary turbomachine architectures described herein are by way of example only, and that in other embodiments, other suitable architectures may be provided. Further, in other exemplary embodiments, the CCA systems 1100, intercooler assemblies 700, etc. may be incorporated into any suitable turbomachine 616 architecture of the present disclosure. Moreover, although the turbomachines 616 described herein have been described in the context of turboprop engines 610, in other exemplary embodiments, the gas turbine engines may not include a propeller assembly 614 and accordingly may instead be configured as a turboshaft engine, usable for any suitable turboshaft application.

As described above, improved power output of a gas turbine engine can be realized through increasing the overall pressure ratio, which results in more energy in a flow through the turbine section of the engine, allowing for more power extraction by the turbine section for a given engine size. As discussed above, the inventors of the present disclosure determined that certain multiple variable compressor geometry increases the overall pressure ratio. For example, the inventors determined that inlet guide vanes can be used to increase the overall pressure ratio of the engine, such as by directing an incoming airflow into the compressor of the engine to improve power output and efficiency of the engine.

A turboshaft engine or a turboprop engine (e.g., a turboshaft engine connected to a propeller) configured in accordance with the present disclosure, having a compressor section defining an overall pressure ratio as described herein, for the power class of engine described herein, results in an overall more efficient gas turbine engine for the power class.

Referring now to FIG. 33, a turbomachine 616 in accordance with still another exemplary embodiment of the present disclosure is provided. The turbomachine 616 of FIG. 33 may be configured in a similar manner as the exemplary turboprop engine 610 of FIGS. 13 and 14, and, accordingly, the same or similar numbers may refer to the same or similar parts. As shown in FIG. 33, the turboshaft

or turboprop engine 610 defines an axial direction A (extending parallel to a longitudinal centerline axis 612 provided for reference), a radial direction R, and a circumferential direction (i.e., a direction extending about the axial direction A; not depicted). In general, the turboprop engine 610 includes a core turbine engine 616, also referred to herein as the turbomachine 161, and an output shaft assembly 617.

The exemplary turbomachine 616 depicted in FIG. 33 generally includes a substantially tubular outer casing 1202 that partially encloses an annular inlet duct 654. The radial inlet duct 654 includes at least a portion extending generally along the radial direction R, and is further configured to turn a direction of an air flow therethrough, such that the resulting airflow is generally along the axial direction A. Additionally, the outer casing 1202 encases, in serial flow relationship, a compressor section including a single compressor 622; a combustion section 626 including a reverse flow combustor; a turbine section including a high pressure (HP) turbine 628 and a low pressure (LP) turbine 630; and an exhaust section 634. Moreover, the turboshaft or turboprop engine 610 depicted is a dual-spool engine, including a first, high pressure (HP) shaft or spool 634 coupling the HP turbine 628 to the compressor 622, and a low pressure (LP) shaft or spool 636 coupled to the LP turbine 630, and drivingly connecting the LP turbine 630 to an output shaft assembly 617, a gearbox 646 (which may drive, e.g., a fan assembly (not shown)), etc.

The compressor section 622, combustion section 626, turbine section 628-630, and exhaust section 634, referred together as the core engine or turbomachine 616, together define a core air flowpath 704, also referred to as a working gas flowpath 704, through the core turbine engine 616. Notably, for the embodiment depicted, the turbomachine 616 further includes a stage of inlet guide vanes (IGVs) 1204 at a forward end of the working gas flowpath 704. Specifically, the IGVs 1204 are positioned at least partially within the radial inlet duct 654, the radial inlet duct 654 located upstream of the compressor section, including the compressor 622. More specifically, for the embodiment depicted, the compressor section, including the compressor 622, is located downstream of the IGVs 1204. Further, the exemplary IGVs 1204 of FIG. 33 are configured as variable IGVs 1204. The variable IGVs 1204 are each rotatable about a pitch axis 1206, allowing for the IGVs 1204 to direct (and modify) an airflow through the inlet duct 654 into the compressor 622 of the compressor section in a desired direction. The IGVs 1204 are used to control the airflow to the first stage of rotor blades of the compressor 622. In particular, the IGVs 1204 can be angled to at least partially swirl the air. Adjusting the airflow helps to reduce turbulence as well as increases the compression ratio of the engine 610, which helps to improve engine efficiency and increase output power. In certain embodiments, each of the variable IGVs 1204 may be configured to rotate completely about the respective pitch axis 1206, or alternatively, each of the plurality of variable IGVs 1204 may include a flap or tail configured to rotate about a respective pitch axis 1206. It should be appreciated, however, that in still other exemplary embodiments, each of the IGVs 1204 may not be configured to rotate about a respective pitch axis 1206, and instead may include any other suitable geometry or configuration allowing for a variance in a direction of the airflow over the variable IGVs 1204. Additionally, in other exemplary embodiments, the stage of IGVs 1204 may instead be located at any other suitable location within the radial inlet duct 654.

Furthermore, the compressor 622 of the compressor section includes at least three stages of compressor rotor blades.

More specifically, for the embodiment depicted, the compressor 622 of the compressor section includes at least four stages of compressor rotor blades. More specifically still, for the embodiment depicted, the compressor 622 of the compressor section includes four stages of radially oriented compressor rotor blades 1208, and an additional stage of centrifugal compressor rotor blades 1210. As is depicted, the core turbine engine 616 further includes a transition duct 627 immediately downstream of the compressor 622, the transition duct 627 having at least a portion extending generally along the radial direction R to provide a compressed air flow from the compressor 622 to the reverse flow combustor 626. The stage of centrifugal compressor rotor blades 1210 are configured to assist with turning the compressed air within the compressor section radially outward into the transition duct 627. Notably, however, in other exemplary embodiments, the combustion section may not include the reverse flow combustor 626. With such an exemplary embodiment, the compressor 622 may not include the stage of centrifugal compressor rotor blades 1210.

Additionally, between each stage of compressor rotor blades 1208, 1210, the compressor section includes a stage of compressor stator vanes. Notably, the first stage of compressor stator vanes is configured as a stage of variable compressor stator vanes 1212, such that each of the variable compressor stator vanes 1212 may rotate about a respective pitch axis 1214. By contrast, the remaining stages of compressor stator vanes are configured as fixed compressor stator vanes 1216. Such a configuration may assist with increasing an overall pressure ratio of the compressor 622. For example, the compressor 622 having the multiple number of stages of compressor rotor blades 1208, 1210, and optionally including a stage of variable compressor stator vanes 1212, in addition to being located downstream of a stage of variable inlet guide vanes 1204, may allow for the compressor 622 of the compressor section to operate in a more efficient manner. More specifically, for the embodiment depicted, the compressor section configured in accordance with one or more exemplary aspects of the present disclosure defines an overall pressure ratio of at least 14:1, such as at least 15:1. For example, in certain exemplary embodiments, the overall pressure ratio of the compressor section may be at least 16:1, and up to 22:1, such as up to 20:1. As used herein, the term "overall pressure ratio" refers to a pressure ratio of the compressor section during operation of the turboshaft or turboprop engine 610 at a rated speed.

It will be appreciated that during operation of the turboshaft/turboprop engine 610, a volume of air 1254 enters the turboshaft/turboprop engine 610 through the radial inlet duct 654, and flows across the variable inlet guide vanes 1204 and into the compressor 622 of the compressor section. A pressure of the air is increased as it is routed through the compressor 622 and is then provided to the reverse flow combustor 626 of the combustion section, where the air is mixed with fuel and burned to provide combustion gases. The combustion gases are routed through the HP turbine 628 where a portion of thermal and/or kinetic energy from the combustion gases is extracted via sequential stages of HP turbine stator vanes 1256 that are coupled to the outer casing 1202 and HP turbine rotor blades 1258 that are coupled to the HP shaft 634, thus causing the HP shaft 634 to rotate, which supports operation of the compressor 622. The combustion gases are then routed through the LP turbine 630 where a second portion of thermal and kinetic energy is extracted from the combustion gases via sequential stages of

LP turbine stator vanes 1260 that are coupled to the outer casing 1202 and LP turbine rotor blades 1262 that are coupled to the LP shaft 636, thus causing the LP shaft 636 to rotate, which supports operation of the output shaft assembly 617. The combustion gases are subsequently routed through the exhaust section 632 of the core turbine engine 616.

Notably, for the embodiment depicted, the HP turbine 628 includes at least two stages of HP turbine rotor blades 1218 and up to three stages of HP turbine rotor blades 1218. In particular, the HP turbine 628 includes two stages of HP turbine rotor blades 1218. Such a configuration may ensure a sufficient amount of power is provided to the compressor 622 through the HP shaft 634. For the embodiment depicted, the HP turbine rotor blades 58 of the at least two stages of HP turbine rotor blades 58 are formed of a ceramic matrix composite material. Accordingly, the HP turbine rotor blades 58 may be capable of withstanding the relatively elevated temperatures within the HP turbine 628 without requiring a flow of cooling air to cool the HP turbine rotor blades 58.

It should be appreciated, however, that in other exemplary embodiments, the HP turbine rotor blades 1218 may be air cooled HP turbine rotor blades. For example, referring briefly to FIG. 34 providing a perspective view of an HP turbine rotor blade 1218 in accordance with an exemplary embodiment of the present disclosure, the HP turbine rotor blade 1218 may include a plurality of cooling holes 1220 through which a cooling airflow 1222 is provided during operation of the turboshaft engine 610 to maintain a temperature of the HP turbine rotor blade 1218 below a predetermined temperature threshold. The cooling airflow 1222 may be received from, e.g., one or more of the CCA systems described herein above.

Referring again to FIG. 33, as described above, the LP shaft 636 is coupled to the LP turbine 630, and is further mechanically coupled to the drive shaft assembly 617. More specifically, the drive shaft assembly 616 includes a gearbox 646 and a driveshaft 1224. The LP shaft 636 is mechanically coupled to the drive shaft 1224 of the drive shaft assembly 617 through the gearbox 646. As will be appreciated, the driveshaft 1124 may be coupled to any suitable device. For example, in certain exemplary embodiments, the turboshaft engine 610 of FIG. 33 may be utilized to drive a propeller of a helicopter, may be utilized in aeroderivative applications, or may be attached to a propeller for an airplane, in which case the turboshaft engine may instead be referred to as a turboprop engine.

Moreover, it will be appreciated that the turboshaft engine 610 depicted in FIG. 33 is a relatively small turboshaft engine 610. For example, the turboshaft engine 610 depicted may be rated to generate less than about 2,000 horsepower at maximum rated power (also referred to as "peak operation") and the compressor section (including the compressor 622) may have a nominal design of less than about 10.5 pounds per second of airflow. Notably, as used herein, "horsepower" refers to brake horsepower during standard day operating conditions, i.e., a horsepower delivered to the drive shaft assembly 617 by the LP shaft 636 during operation of the turboshaft engine 610 at sea level and with an ambient temperature of around seventy (70) degrees Fahrenheit.

A turboshaft engine configured in accordance with the present disclosure, having a compressor section defining an overall compressor ratio as described herein, for the power class of engine described herein, results in an overall more efficient gas turbine engine for the power class.

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.

For example, as discussed in more detail above, in additional or alternative embodiments, a gas turbine engine, such as a turboprop or turboshaft engine, may incorporate advanced materials capable of withstanding the relatively high temperatures at downstream stages of a high pressure compressor (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.). 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."

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.

As used herein, ceramic matrix composite (CMC) material refers to a non-metallic material having high temperature capability. Exemplary CMC materials utilized for such components may include silicon carbide (SiC), silicon nitride, or alumina matrix materials and combinations thereof. Ceramic fibers may be embedded within the matrix, such as oxidation stable reinforcing fibers including monofilaments like sapphire and silicon carbide (e.g., Textron's SCS-6), as well as roving and yarn including silicon carbide (e.g., Nippon Carbon's NICALON®, Ube Industries' TYRANNO®, and Dow Corning's SYLRAMIC®), alumina silicates (e.g., Nextel's 440 and 480), and chopped whiskers and fibers (e.g., Nextel's 440 and SAFFIL®), and 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). For example, in certain embodiments, bundles of the fibers, which may include a ceramic refractory material coating, are formed as a reinforced tape, such as a unidirectional reinforced tape. A plurality of the tapes may be laid up together (e.g., as plies) 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. In other embodiments, the CMC material may be formed as, e.g., a carbon fiber cloth rather than as a tape.

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. 13 through 17 and 20 through 32. The method includes operating the gas turbine engine at a takeoff power level, wherein operating the gas turbine engine at the takeoff power level includes driving a propeller of a propeller assembly across a propeller shaft of the propeller assembly, the gas turbine engine further including a turbomachine with a high pressure compressor defining a high pressure compressor exit area ($A_{HPCExit}$) in square inches and a low pressure turbine defining a drive turbine exit area (A_{DTExit}) in square inches, the gas turbine engine defining a maximum exhaust gas temperature (EGT) in degrees Celsius, a maximum drive turbine shaft torque (T_{OUT}) in Newton meters, and a corrected specific power in Newtons squared times degrees Celsius over meters squared. The corrected specific power is determined in accordance with Expression (3), and is greater than $0.0001194 \times EGT^2 - 0.103 \times EGT + 22.14$ and less than $0.0003294 \times EGT^2 - 0.3061 \times EGT + 77.91$.

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. Additionally, or alternatively, operating the gas turbine engine at the takeoff power level further includes reducing a temperature of an airflow through the compressor section with an intercooler assembly.

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.

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

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 ($A_{HPCExit}$) in square inches; wherein the gas turbine engine defines a maximum exhaust gas temperature (EGT) in degrees Celsius, a total sea level static thrust output (F_{nTotal}) 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: $F_{nTotal} \times EGT / (A_{HPCExit}^2 \times 1000)$.

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.

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

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

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

The gas turbine engine of any preceding clause, wherein the EGT is greater than 1000 degrees Celsius and less than 1300 degrees Celsius, and wherein the corrected specific thrust is greater than or equal to 45.

The gas turbine engine of any preceding clause, wherein the EGT is greater than 1000 degrees Celsius and less than 1300 degrees Celsius, and wherein the corrected specific thrust is greater than or equal to 50.

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.

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.

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.

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.

The gas turbine engine of any preceding clause, further comprising a primary fan driven by the turbomachine.

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.

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.

The gas turbine engine of any preceding clause, wherein the secondary fan is a single stage secondary fan.

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 ($A_{HPCExit}$) in square inches, the gas turbine engine defining a maximum exhaust gas temperature (EGT) in degrees Celsius, a total sea level static thrust output (Fn_{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: $Fn_{Total} \times EGT / (A_{HPCExit}^2 \times 1000)$.

The method of any preceding clause, wherein the EGT defined by the gas turbine engine is greater than 1000 degrees Celsius and less than 1300 degrees Celsius.

The method of any preceding clause, wherein the EGT defined by the gas turbine engine is greater than 1100 degrees Celsius and less than 1300 degrees Celsius.

The method of any preceding clause, wherein the EGT defined by the gas turbine engine is greater than 1000

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degrees 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.

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.

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.

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).

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).

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).

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).

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).

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). or a combination thereof.

In one or more of the exemplary cooled cooling air systems described herein, the 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.

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.

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.

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.

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.

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.

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 ($A_{HPCExit}$) in square inches and the turbine section having a drive turbine defining a drive turbine exit area (A_{DTExit}) in square inches, the turbomachine further comprising a drive turbine shaft coupled to the drive turbine; wherein the gas turbine engine defines a maximum exhaust gas temperature (EGT) in degrees Celsius, a maximum drive turbine shaft torque (T_{OUT}) in Newton meters, and a corrected specific power (CSP) in Newtons squared times degrees Celsius over meters squared, wherein the corrected specific power is determined as follows:

$$\left(\frac{T_{OUT}}{\sqrt{A_{DTExit}}} \right)^2 * \frac{EGT}{A_{HPCExit}} * 10^{-11};$$

wherein CSP is greater than $0.0001194 \times EGT^2 - 0.103 \times EGT + 22.14$ and less than $0.0003294 \times EGT^2 - 0.306 \times EGT + 77.91$; and wherein EGT is greater than 525 degrees Celsius and less than 1250 degrees Celsius.

The gas turbine engine of any preceding clause, wherein CSP is less than 210 and greater than 1.

The gas turbine engine of any preceding clause, wherein the EGT is greater than 750 degrees Celsius and less than 1100 degrees Celsius, and wherein CSP is less than 140 and greater than 12.1.

The gas turbine engine of any preceding clause, wherein the T_{OUT} is greater than 530 Newton-meters and less than 4740 Newton-meters.

The gas turbine engine of any preceding clause, further comprising: an intercooler assembly comprising a heat exchanger, the heat exchanger in thermal communication with the compressor section.

The gas turbine engine of any preceding clause, wherein the compressor section defines in part a working gas flowpath through the turbomachine, and wherein heat exchanger is in direct thermal communication with the working gas flowpath through the compressor section.

The gas turbine engine of any preceding clause, wherein the compressor section defines in part a working gas flowpath through the turbomachine, wherein the compressor section comprises a first compressor and a second compressor, and wherein the heat exchanger is in thermal communication with the working gas flowpath through the compressor section at a location between the first compressor and the second compressor.

The gas turbine engine of any preceding clause, wherein the compressor section defines in part a working gas flowpath through the turbomachine, wherein the compressor section comprises a compressor defining an upstream end and a downstream end, and wherein the heat exchanger is in thermal communication with the working gas flowpath through the compressor at a location between the upstream end and the downstream end of the compressor.

The gas turbine engine of any preceding clause, wherein the heat exchanger is located externally of the turbomachine.

The gas turbine engine of any preceding clause, further comprising a fuel system configured as a liquid hydrogen fuel system, and wherein the heat exchanger is in thermal communication with the liquid hydrogen fuel system.

The gas turbine engine of any preceding clause, wherein the compressor section comprises a first compressor, and

wherein the first compressor is configured as an axial compressor, a centrifugal compressor, or an axial-centrifugal compressor.

The gas turbine engine of any preceding clause, wherein the compressor section further comprises a second compressor, and wherein the second compressor is configured as an axial compressor, a centrifugal compressor, or an axial-centrifugal compressor.

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.

The gas turbine engine of any 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.

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.

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.

The gas turbine engine of any preceding clause, wherein the gas turbine engine is a turboprop engine further comprising: a propeller assembly, wherein the propeller assembly comprises a propeller driven by the drive turbine shaft.

The gas turbine engine of any preceding clause, further comprising: a power gearbox, wherein the drive turbine shaft is driven by the turbomachine across the power gearbox.

The gas turbine engine of any preceding clause, wherein the gas turbine engine is a turboshaft engine.

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 ($A_{HPCExit}$) in square inches and the turbine section having a drive turbine defining a drive turbine exit area (A_{DTExit}) in square inches, the turbomachine further comprising a drive turbine shaft coupled to the drive turbine and defining an overall pressure ratio greater than 14:1 and less than or equal to 22:1; and wherein the gas turbine engine defines a maximum exhaust gas temperature (EGT) greater than 600 degrees Celsius and less than 1000 degrees Celsius, an output power of at least 550 horsepower and up to 2,000 horsepower when operated at a rated speed, a maximum drive turbine shaft torque (T_{OUT}) in Newton meters, and a corrected specific power (CSP) in Newtons squared times degrees Celsius over meters squared, wherein CSP is greater than 3.3 and less than 101 and is determined as follows:

$$\left(\frac{T_{OUT}}{\sqrt{A_{DTExit}}} \right)^2 * \frac{EGT}{A_{HPCExit}} * 10^{-11}.$$

The gas turbine engine of any preceding clause, wherein the turbomachine defines an overall pressure ratio greater than 15:1 and less than or equal to 20:1.

The gas turbine engine of any preceding clause, wherein the turbomachine comprises a stage of variable inlet guide vanes upstream of the compressor section and a stage of variable stator vanes within the compressor section.

The gas turbine engine of any preceding clause, wherein the turbine section further comprises a high pressure turbine having a first stage of high pressure turbine rotor blades and a second stage of high pressure turbine rotor blades, wherein the first and second stages of high pressure turbine rotor blades each include air cooled high pressure turbine rotor blades.

The gas turbine engine of any preceding clause, wherein the drive turbine is a low pressure turbine comprising three stages of low pressure turbine rotor blades.

The gas turbine engine of any preceding clause, wherein the compressor section includes an airflow of less than 10.5 pounds per second therethrough when the gas turbine engine is operated at a takeoff power level.

The gas turbine engine of any preceding clause, wherein the gas turbine engine defines a maximum exhaust gas temperature (EGT) greater than 700 degrees Celsius and less than 900 degrees Celsius, and wherein CSP is greater than or equal to 4 and less than or equal to 69.

A method of operating a gas turbine engine, comprising: operating the gas turbine engine at a takeoff power level, wherein operating the gas turbine engine at the takeoff power level comprises driving a propeller of a propeller assembly across a propeller shaft of the propeller assembly, the gas turbine engine further comprising a turbomachine with a high pressure compressor defining a high pressure compressor exit area ($A_{HPCExit}$) in square inches and a drive turbine defining a drive turbine exit area (A_{DTExit}) in square inches, the gas turbine engine defining a maximum exhaust gas temperature (EGT) in degrees Celsius, a maximum drive turbine shaft torque (T_{OUT}) in Newton meters, and a corrected specific power in Newtons squared times degrees Celsius over meters squared; wherein the corrected specific power (CSP) is determined as follows:

$$\left(\frac{T_{OUT}}{\sqrt{A_{DTExit}}} \right)^2 * \frac{EGT}{A_{HPCExit}} * 10^{-11};$$

wherein CSP is greater than $0.0001194 \times EGT^2 - 0.103 \times EGT + 22.14$ and less than $0.0003294 \times EGT^2 - 0.3061 \times EGT + 77.91$; and wherein EGT is greater than 525 degrees Celsius and less than 1250 degrees Celsius.

A gas turbine engine comprising: a turbomachine comprising a stage of inlet guide vanes, 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 ($A_{HPCExit}$) in square inches and the turbine section having a drive turbine defining a drive turbine exit area (A_{DTExit}) in square inches, the turbomachine further comprising a drive turbine shaft coupled to the drive turbine; wherein the gas turbine engine defines a maximum exhaust gas temperature (EGT) in degrees Celsius, a maximum drive turbine shaft torque (T_{OUT}) in Newton meters, and a corrected specific power (CSP) in Newtons squared times degrees Celsius over meters squared, wherein the corrected specific power is determined as follows:

$$\left(\frac{T_{OUT}}{\sqrt{A_{DTExit}}} \right)^2 * \frac{EGT}{A_{HPCExit}} * 10^{-11};$$

5 wherein CSP is greater than $0.0001194 \times EGT^2 - 0.103 \times EGT + 22.14$ and less than $0.0003294 \times EGT^2 - 0.3061 \times EGT + 77.91$; and wherein EGT is greater than 525 degrees Celsius and less than 1250 degrees Celsius; wherein an overall pressure ratio for the compressor section is at least 15:1; and wherein the gas turbine engine generates less than 2,000 horsepower at maximum rated power.

15 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 ($A_{HPCExit}$) in square inches and the turbine section having a drive turbine defining a drive turbine exit area (A_{DTExit}) in square inches, the turbomachine further comprising a drive turbine shaft coupled to the drive turbine and defining an overall pressure ratio greater than 15:1 and less than or equal to 22:1; and wherein the gas turbine engine defines a maximum exhaust gas temperature (EGT) greater than 600 degrees Celsius and less than 1000 degrees Celsius, an output power of at least 550 horsepower and less than 2,000 horsepower when operated at a maximum rated power, a maximum drive turbine shaft torque (T_{OUT}) in Newton meters, and a corrected specific power (CSP) in Newtons squared times degrees Celsius over meters squared, wherein CSP is greater than 3.3 and less than 101 and is determined as follows:

$$35 \left(\frac{T_{OUT}}{\sqrt{A_{DTExit}}} \right)^2 * \frac{EGT}{A_{HPCExit}} * 10^{-11}.$$

The gas turbine engine of any preceding clause, wherein the turbomachine defines an overall pressure ratio greater than 15:1 and less than or equal to 20:1.

40 The gas turbine engine of any preceding clause, wherein the turbine section includes a high pressure turbine and a low pressure turbine, the high pressure turbine comprising at least two stages of turbine rotor blades.

45 The gas turbine engine of any preceding clause, further comprising: a first spool coupling the compressor section to the high pressure turbine; and a second spool coupled to the low pressure turbine.

The gas turbine engine of any preceding clause, wherein 50 the second spool is mechanically coupled to a drive shaft.

The gas turbine engine of any preceding clause, further comprising: a gearbox, wherein the second spool is mechanically coupled to a drive shaft through the gearbox.

The gas turbine engine of any preceding clause, wherein 55 the turbine rotor blades of the at least two stages of turbine rotor blades of the high pressure turbine are air cooled turbine rotor blades.

The gas turbine engine of any preceding clause, wherein the turbine rotor blades of the at least two stages of turbine rotor blades of the high pressure turbine are formed of a ceramic matrix composite material.

The gas turbine engine of any preceding clause, wherein the compressor section further comprises a stage of variable stator vanes.

60 The gas turbine engine of any preceding clause, wherein the compressor of the compressor section includes at least three stages of compressor rotor blades.

The gas turbine engine of any preceding clause, wherein the compressor of the compressor section includes at least four stages of compressor rotor blades.

The gas turbine engine of any preceding clause, wherein the overall compressor ratio of the compressor is at least 16:1.

The gas turbine engine of any preceding clause, further comprising: a combustion section comprising a reverse flow combustor.

The gas turbine engine of any preceding clause, further comprising: a radial inlet duct located upstream of the compressor section.

The gas turbine engine of any preceding clause, wherein the EGT is greater than 750 degrees Celsius and less than 1100 degrees Celsius, and wherein CSP is less than 140 and greater than 121.

The gas turbine engine of any preceding clause, wherein the T_{OUT} is greater than 530 Newton-meters and less than 4740 Newton-meters.

The gas turbine engine of any preceding clause, further comprising: an intercooler assembly comprising a heat exchanger, the heat exchanger in thermal communication with the compressor section.

The gas turbine engine of any preceding clause, wherein the compressor section defines in part a working gas flowpath through the turbomachine, and wherein the heat exchanger is in direct thermal communication with the working gas flowpath through the compressor section.

The gas turbine engine of any preceding clause, wherein the compressor section defines in part a working gas flowpath through the turbomachine, wherein the compressor section comprises a first compressor and a second compressor, and wherein the heat exchanger is in thermal communication with the working gas flowpath through the compressor section at a location between the first compressor and the second compressor.

The gas turbine engine of any preceding clause, wherein the compressor section defines in part a working gas flowpath through the turbomachine, wherein the compressor section comprises a compressor defining an upstream end and a downstream end, and wherein the heat exchanger is in thermal communication with the working gas flowpath through the compressor at a location between the upstream end and the downstream end of the compressor.

A method of operating a gas turbine engine, comprising: operating the gas turbine engine at a takeoff power level, wherein operating the gas turbine engine at the takeoff power level comprises driving a propeller of a propeller assembly across a propeller shaft of the propeller assembly, the gas turbine engine further comprising a turbomachine with a high pressure compressor defining a high pressure compressor exit area ($A_{HPCExit}$) in square inches and a drive turbine defining a drive turbine exit area (A_{DTExit}) in square inches, the gas turbine engine defining a maximum exhaust gas temperature (EGT) in degrees Celsius, a maximum drive turbine shaft torque (T_{OUT}) in Newton meters, and a corrected specific power in Newtons squared times degrees Celsius over meters squared; wherein the corrected specific power (CSP) is determined as follows:

$$\left(\frac{T_{OUT}}{\sqrt{A_{DTExit}}} \right)^2 * \frac{EGT}{A_{HPCExit}} * 10^{-11};$$

wherein CSP is greater than $0.0001194 \times EGT^2 - 0.103 \times EGT + 22.14$ and less than $0.0003294 \times EGT^2 - 0.3061 \times EGT + 57.791$; wherein EGT is greater than 525 degrees Celsius and less than 1250 degrees Celsius; wherein an overall pressure ratio is at least 15:1; and wherein the gas turbine engine generates less than 2,000 horsepower at maximum rated power.

77.91; wherein EGT is greater than 525 degrees Celsius and less than 1250 degrees Celsius; wherein an overall pressure ratio is at least 15:1; and wherein the gas turbine engine generates less than 2,000 horsepower at maximum rated power.

A gas turbine engine defining an axial direction and a radial direction, the gas turbine engine comprising: a stage of variable inlet guide vanes; a compressor section comprising a compressor located downstream of the stage of variable inlet guide vanes and defining an overall compressor ratio of at least 15:1; and a turbine section located downstream of the compressor section comprising a high pressure turbine and a low pressure turbine, the high pressure turbine comprising at least two stages of turbine rotor blades; wherein the gas turbine engine generates less than 2,000 horsepower at maximum rated power.

The gas turbine engine of any preceding clause, further comprising: a first spool coupling the compressor to the high pressure turbine; and a second spool coupled to the low pressure turbine.

The gas turbine engine of any preceding clause, wherein the second spool is mechanically coupled to a drive shaft.

The gas turbine engine of any preceding clause, further comprising: a gearbox, wherein the second spool is mechanically coupled to a drive shaft through the gearbox.

The gas turbine engine of any preceding clause, wherein the compressor section further comprises a stage of variable stator vanes.

The gas turbine engine of any preceding clause, wherein the compressor of the compressor section includes at least three stages of compressor rotor blades.

The gas turbine engine of any preceding clause, wherein the compressor of the compressor section includes at least four stages of compressor rotor blades.

The gas turbine engine of any preceding clause, wherein the overall compressor ratio of the compressor is at least 16:1.

The gas turbine engine of any preceding clause, further comprising: a combustion section comprising a reverse flow combustor.

The gas turbine engine of any preceding clause, wherein the turbine rotor blades of the at least two stages of turbine rotor blades of the high pressure turbine are air cooled turbine rotor blades.

The gas turbine engine of any preceding clause, wherein the turbine rotor blades of the at least two stages of turbine rotor blades of the high pressure turbine are formed of a ceramic matrix composite material.

The gas turbine engine of any preceding clause, further comprising: a radial inlet duct located upstream of the compressor section.

A gas turbine engine defining an axial direction and a radial direction, the gas turbine engine comprising: a compressor section defining an overall compressor ratio of at least 15:1; and a turbine section located downstream of the compressor section comprising a high pressure turbine and a low pressure turbine; wherein the gas turbine engine generates less than 2,000 horsepower at maximum rated power.

The gas turbine engine of any preceding clause, wherein the high pressure turbine comprises at least two stages of turbine rotor blades.

The gas turbine engine of any preceding clause, wherein the turbine rotor blades of the at least two stages of turbine rotor blades of the high pressure turbine are air cooled turbine rotor blades.

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The gas turbine engine of any preceding clause, wherein the turbine rotor blades of the at least two stages of turbine rotor blades of the high pressure turbine are formed of a ceramic matrix composite material.

The gas turbine engine of any preceding clause, wherein the compressor of the compressor section includes at least four stages of compressor rotor blades.

The gas turbine engine of any preceding clause, wherein the overall compressor ratio of the compressor is at least 16:1.

The gas turbine engine of any preceding clause, further comprising: a first spool coupling the compressor to the high pressure turbine; and a second spool coupled to the low pressure turbine, wherein the second spool is further mechanically coupled to a drive shaft.

The gas turbine engine of any preceding clause, further comprising: a stage of variable inlet guide vanes, wherein the compressor section is located downstream of the stage of variable inlet guide vanes.

A gas turbine engine defining an axial direction and a radial direction, the gas turbine engine comprising: a stage of variable inlet guide vanes; a compressor section comprising a compressor located downstream of the stage of variable inlet guide vanes and defining an overall compressor ratio of at least 15:1; and a turbine section located downstream of the compressor section comprising a high pressure turbine and a low pressure turbine, the high pressure turbine comprising at least two stages of turbine rotor blades; wherein the gas turbine engine generates less than 2,000 horsepower at maximum rated power.

The gas turbine engine of any preceding clause, further comprising: a first spool coupling the compressor to the high pressure turbine; and a second spool coupled to the low pressure turbine.

The gas turbine engine of any preceding clause, wherein the second spool is mechanically coupled to a drive shaft.

The gas turbine engine of any preceding clause, further comprising: a gearbox, wherein the second spool is mechanically coupled to a drive shaft through the gearbox.

The gas turbine engine of any preceding clause, wherein the compressor section further comprises a stage of variable stator vanes.

The gas turbine engine of any preceding clause, wherein the compressor of the compressor section includes at least three stages of compressor rotor blades.

The gas turbine engine of any preceding clause, wherein the compressor of the compressor section includes at least four stages of compressor rotor blades.

The gas turbine engine of any preceding clause, wherein the overall compressor ratio of the compressor is at least 16:1.

The gas turbine engine of any preceding clause, further comprising: a combustion section comprising a reverse flow combustor.

The gas turbine engine of any preceding clause, wherein the turbine rotor blades of the at least two stages of turbine rotor blades of the high pressure turbine are air cooled turbine rotor blades.

The gas turbine engine of any preceding clause, wherein the turbine rotor blades of the at least two stages of turbine rotor blades of the high pressure turbine are formed of a ceramic matrix composite material.

The gas turbine engine of any preceding clause, further comprising: a radial inlet duct located upstream of the compressor section.

A gas turbine engine defining an axial direction and a radial direction, the gas turbine engine comprising: a com-

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pressor section defining an overall compressor ratio of at least 15:1; and a turbine section located downstream of the compressor section comprising a high pressure turbine and a low pressure turbine; wherein the gas turbine engine generates less than 2,000 horsepower at maximum rated power.

The gas turbine engine of any preceding clause, wherein the high pressure turbine comprises at least two stages of turbine rotor blades.

10 The gas turbine engine of any preceding clause, wherein the turbine rotor blades of the at least two stages of turbine rotor blades of the high pressure turbine are air cooled turbine rotor blades.

15 The gas turbine engine of any preceding clause, wherein the turbine rotor blades of the at least two stages of turbine rotor blades of the high pressure turbine are formed of a ceramic matrix composite material.

20 The gas turbine engine of any preceding clause, wherein the compressor of the compressor section includes at least four stages of compressor rotor blades.

The gas turbine engine of any preceding clause, wherein the overall compressor ratio of the compressor is at least 16:1.

25 The gas turbine engine of any preceding clause, further comprising: a first spool coupling the compressor to the high pressure turbine; and a second spool coupled to the low pressure turbine, wherein the second spool is further mechanically coupled to a drive shaft.

30 The gas turbine engine of any preceding clause, further comprising: a stage of variable inlet guide vanes, wherein the compressor section is located downstream of the stage of variable inlet guide vanes.

We claim:

1. A gas turbine engine comprising:

a turbomachine comprising a stage of inlet guide vanes, 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 ($A_{HPCExit}$) in square inches and the turbine section having a drive turbine defining a drive turbine exit area (A_{DTExit}) in square inches, the turbomachine further comprising a drive turbine shaft coupled to the drive turbine;

wherein the gas turbine engine defines a maximum exhaust gas temperature (EGT) in degrees Celsius, a maximum drive turbine shaft torque (T_{OUT}) in Newton meters, and a corrected specific power (CSP) in Newtons squared times degrees Celsius over meters squared, wherein the corrected specific power is determined as follows:

$$\left(\frac{T_{OUT}}{\sqrt{A_{DTEexit}}} \right)^2 * \frac{EGT}{A_{HPCExit}} * 10^{-11};$$

wherein CSP is greater than $0.0001194 \times EGT^2 - 0.103 \times EGT + 22.14$ and less than $0.0003294 \times EGT^2 - 0.306 \times EGT + 77.91$;

wherein EGT is greater than 525 degrees Celsius and less than 1250 degrees Celsius;

wherein an overall pressure ratio for the compressor section is at least 15:1; and

wherein the gas turbine engine generates less than 2,000 horsepower at maximum rated power.

2. The gas turbine engine of claim 1, wherein the turbine section includes a high pressure turbine and a low pressure

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turbine, the high pressure turbine comprising at least two stages of turbine rotor blades.

3. The gas turbine engine of claim **2**, further comprising: a first spool coupling the compressor section to the high pressure turbine; and

a second spool coupled to the low pressure turbine.

4. The gas turbine engine of claim **3**, wherein the second spool is mechanically coupled to a drive shaft.

5. The gas turbine engine of claim **4**, further comprising: a gearbox, wherein the second spool is mechanically coupled to a drive shaft through the gearbox. 10

6. The gas turbine engine of claim **2**, wherein the turbine rotor blades of the at least two stages of turbine rotor blades of the high pressure turbine are air cooled turbine rotor blades. 15

7. The gas turbine engine of claim **2**, wherein the turbine rotor blades of the at least two stages of turbine rotor blades of the high pressure turbine are formed of a ceramic matrix composite material.

8. The gas turbine engine of claim **1**, wherein the compressor section further comprises a stage of variable stator vanes. 20

9. The gas turbine engine of claim **1**, wherein the compressor of the compressor section includes at least three stages of compressor rotor blades. 25

10. The gas turbine engine of claim **1**, wherein the compressor of the compressor section includes at least four stages of compressor rotor blades.

11. The gas turbine engine of claim **1**, further comprising: a combustion section comprising a reverse flow combustor. 30

12. The gas turbine engine of claim **1**, further comprising: a radial inlet duct located upstream of the compressor section. 35

13. The gas turbine engine of claim **1**, wherein the EGT is greater than 750 degrees Celsius and less than 1100 degrees Celsius, and wherein CSP is less than 140 and greater than 12.1. 35

14. The gas turbine engine of claim **1**, wherein the T_{OUT} is greater than 530 Newton-meters and less than 4740 Newton-meters. 40

15. The gas turbine engine of claim **1**, further comprising: an intercooler assembly comprising a heat exchanger, the heat exchanger in thermal communication with the compressor section. 45

16. The gas turbine engine of claim **15**, wherein the compressor section defines in part a working gas flowpath through the turbomachine, and wherein the heat exchanger is in direct thermal communication with the working gas flowpath through the compressor section. 50

17. The gas turbine engine of claim **15**, wherein the compressor section defines in part a working gas flowpath through the turbomachine, wherein the compressor section comprises a first compressor and a second compressor, and wherein the heat exchanger is in thermal communication with the working gas flowpath through the compressor section at a location between the first compressor and the second compressor. 55

18. The gas turbine engine of claim **15**, wherein the compressor section defines in part a working gas flowpath through the turbomachine, wherein the compressor section comprises a compressor defining an upstream end and a downstream end, and wherein the heat exchanger is in 60

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thermal communication with the working gas flowpath through the compressor at a location between the upstream end and the downstream end of the compressor.

19. A method of operating a gas turbine engine, comprising:

operating the gas turbine engine at a takeoff power level, wherein operating the gas turbine engine at the takeoff power level comprises driving a propeller of a propeller assembly across a propeller shaft of the propeller assembly, the gas turbine engine further comprising a turbomachine with a high pressure compressor defining a high pressure compressor exit area ($A_{HPCExit}$) in square inches and a drive turbine defining a drive turbine exit area (A_{DTExit}) in square inches, the gas turbine engine defining a maximum exhaust gas temperature (EGT) in degrees Celsius, a maximum drive turbine shaft torque (T_{OUT}) in Newton meters, and a corrected specific power in Newtons squared times degrees Celsius over meters squared;

wherein the corrected specific power (CSP) is determined as follows:

$$\left(\frac{T_{OUT}}{\sqrt{A_{DTExit}}} \right)^2 * \frac{EGT}{A_{HPCExit}} * 10^{-11};$$

wherein CSP is greater than $0.0001194 \times EGT^2 - 0.103 \times EGT + 22.14$ and less than $0.0003294 \times EGT^2 - 0.3061 \times EGT + 77.91$;

wherein EGT is greater than 525 degrees Celsius and less than 1250 degrees Celsius;

wherein an overall pressure ratio is at least 15:1; and wherein the gas turbine engine generates less than 2,000 horsepower at maximum rated power.

20. 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 ($A_{HPCExit}$) in square inches and the turbine section having a drive turbine defining a drive turbine exit area (A_{DTExit}) in square inches, the turbomachine further comprising a drive turbine shaft coupled to the drive turbine and defining an overall pressure ratio greater than 15:1 and less than or equal to 22:1; and wherein the gas turbine engine defines a maximum exhaust gas temperature (EGT) greater than 600 degrees Celsius and less than 1000 degrees Celsius, an output power of at least 550 horsepower and less than 2,000 horsepower when operated at a maximum rated power, a maximum drive turbine shaft torque (T_{OUT}) in Newton meters, and a corrected specific power (CSP) in Newtons squared times degrees Celsius over meters squared, wherein CSP is greater than 3.3 and less than 101 and is determined as follows:

$$\left(\frac{T_{OUT}}{\sqrt{A_{DTExit}}} \right)^2 * \frac{EGT}{A_{HPCExit}} * 10^{-11}.$$

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