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**Minelli**

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(54) **COMBINATION OF A GAS TURBINE  
ENGINE AND A POWER ELECTRONICS**

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(57) **ABSTRACT**

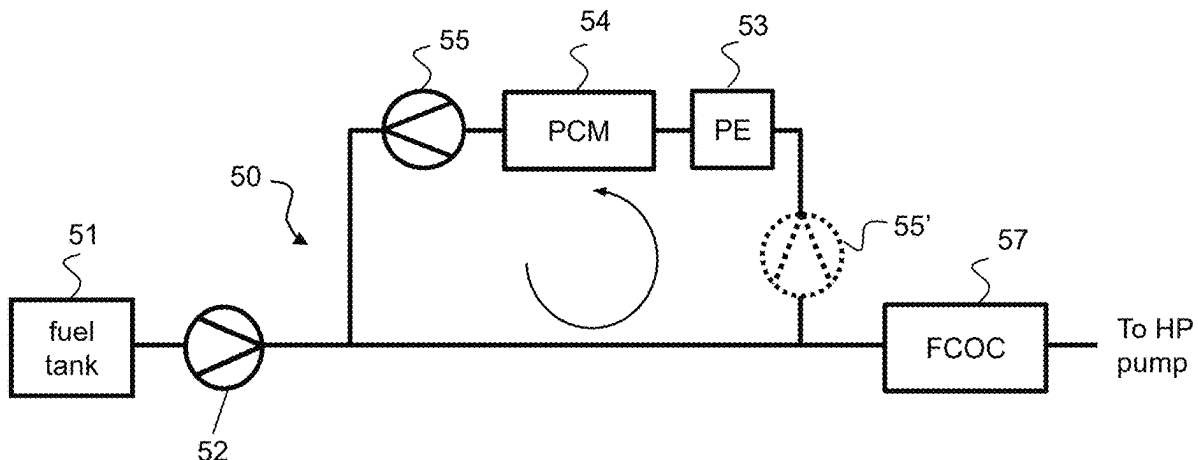
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A combination of a gas turbine engine and a power electronics for powering aircraft and/or engine systems. The engine includes an engine core comprising a turbine, a combustor, a compressor, and a core shaft connecting the turbine to the compressor, and a fuel circuit for supplying a fuel flow to the combustor. The power electronics is configured to transfer heat produced by the power electronics to a cooling flow formed by a portion of the fuel flow. The fuel circuit is configured to circulate the cooling flow in a loop during selected engine conditions such that the cooling flow transfers heat from the power electronics to a phase change material located on the loop. The phase change material has a phase change temperature at a predetermined limiting temperature whereby the phase change material stores heat from the cooling flow to prevent the power electronics exceeding the limiting temperature.

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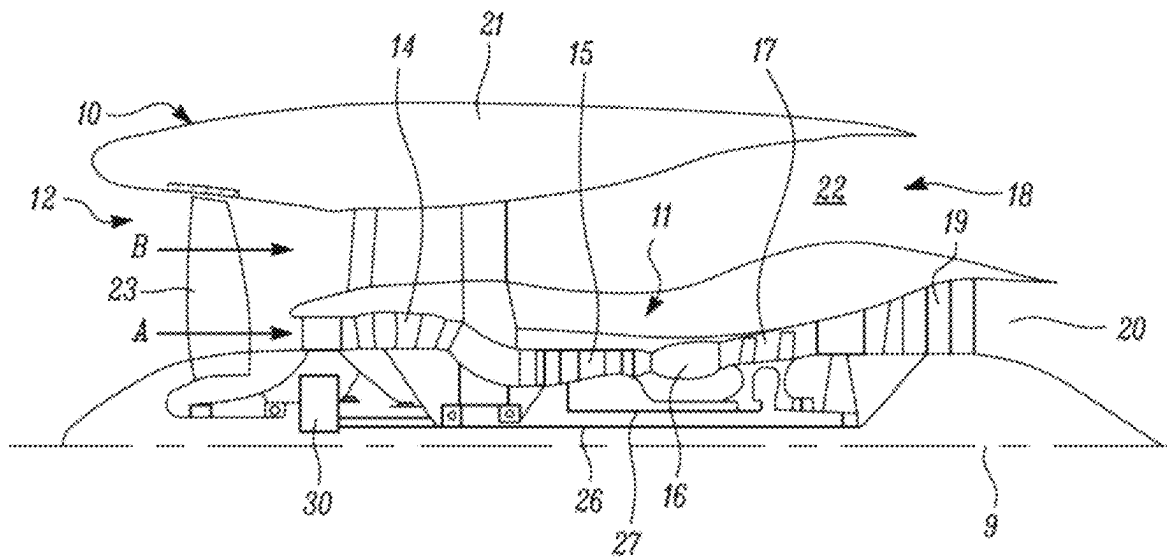


FIG. 1

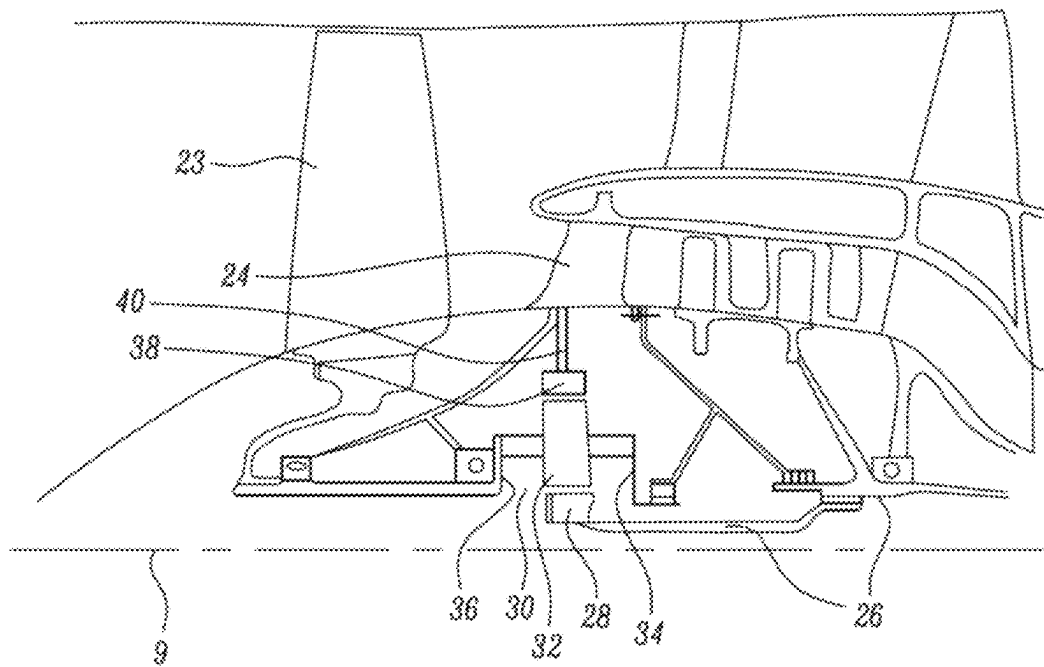
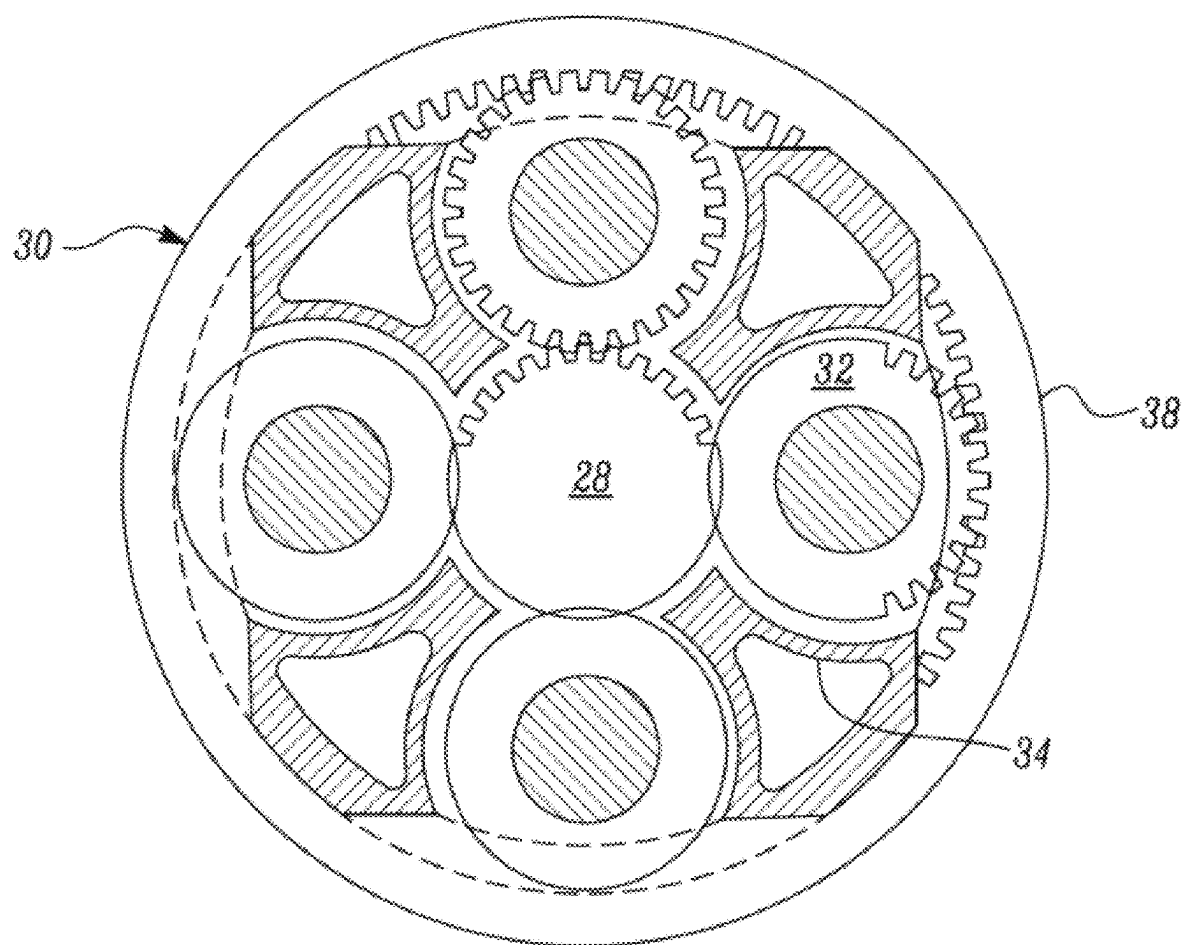


FIG. 2

*FIG. 3*

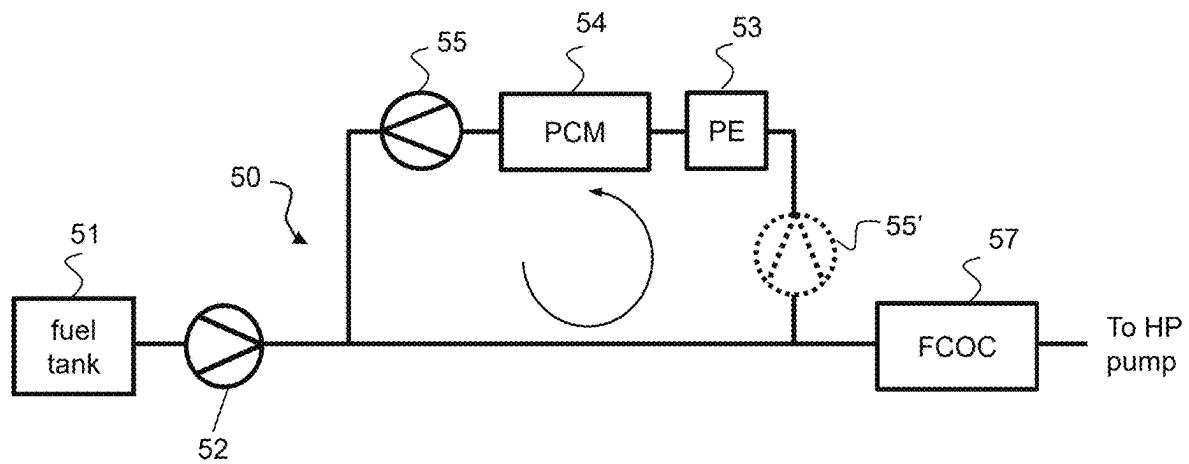


Fig. 4

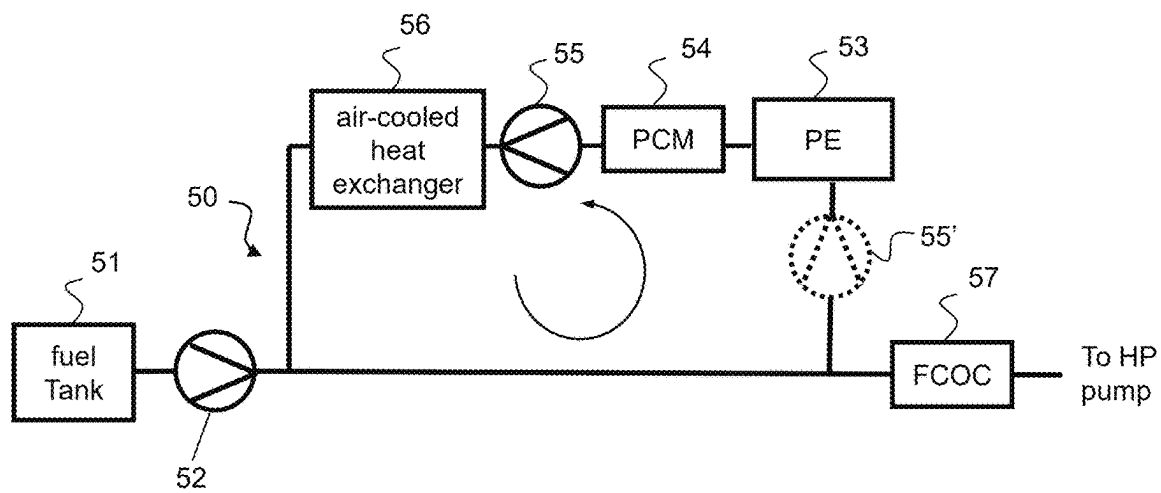


Fig. 5

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## COMBINATION OF A GAS TURBINE ENGINE AND A POWER ELECTRONICS

### CROSS-REFERENCE TO RELATED APPLICATIONS

This specification is based upon and claims the benefit of priority from UK Patent Application Number 2201313.0 filed on Feb. 2, 2022, the entire contents of which are incorporated herein by reference.

### BACKGROUND

#### 1. Field of the Disclosure

The present invention relates to a combination of a gas turbine engine and a power electronics for powering aircraft and/or engine systems.

#### 2. Description of the Related Art

Gas turbine engines include sophisticated thermal management systems to control the temperatures of components. In particular, heat is rejected into the oil of the engine oil system used for cooling and lubricating engine components. The oil in the oil system is cooled in turn by transferring heat to engine fuel and/or airflows.

A further source of cooling demand can derive from the thermal management of electrical components, such as power electronics, which form an increasingly important part of aircraft and/or engine systems. Failure to meet increased cooling demands can result in less reliable or worsened performance of such systems. In particular, the performance of power electronics for powering aircraft and/or engine systems (e.g. the cabin blower system) can be affected by temperature changes and therefore reliably controlling its temperature and rates of temperature change during all phases of aircraft operation is important.

The cooling demand on the thermal management system of the engine can be even greater when the gas turbine engine includes a power gear box (PGB) to drive an engine fan, as the PGB also requires cooling to reliably perform its function.

Conventional heat management systems meet the power electronics' cooling demands by rejecting heat into engine fluid heat sinks (e.g. engine fuel and/or airflows). However, during sub-idle engine operation conditions (which may for example occur at engine start-up and windmill relight), other low-power conditions (e.g. idle and descent), and some transient operations (e.g. from end of cruise to top of descent) conventional engine fluid heat sinks may be unavailable or insufficient to meet the cooling demands of the power electronics. Sub-idle engine operation conditions typically apply from 0 rpm to idle, which is the steady state engine operating condition with no load applied.

Similar problems may also occur during post-shutdown heat soak back conditions. The present invention has been devised in light of the above considerations.

### SUMMARY

The present invention is at least partly based on a recognition that the cooling demand of power electronics for powering aircraft and/or engine systems can be met by making innovative use of the engine fuel circuit, even during engine operation conditions when a rate of fuel flow through

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the fuel circuit may be inadequate to meet the cooling demand of the power electronics.

Thus, in a first aspect, the present invention provides a combination of a gas turbine engine and a power electronics for powering aircraft and/or engine systems,

wherein the engine includes:

an engine core comprising a turbine, a combustor, a compressor, and a core shaft connecting the turbine to the compressor; and

a fuel circuit for supplying a fuel flow to the combustor; wherein the power electronics is configured to transfer heat produced by the power electronics to a cooling flow formed by a portion of the fuel flow for performing heat management of the power electronics; and

wherein the fuel circuit is configured to circulate the cooling flow in a loop during selected engine conditions such that the cooling flow transfers heat from the power electronics to a phase change material located on the loop, the phase change material having a phase change temperature at a predetermined limiting temperature whereby the phase change material stores heat from the cooling flow to prevent the power electronics exceeding the limiting temperature.

Thus, by adopting this arrangement, despite unavailability or insufficiency of conventional heat sinks during selected engine conditions, the cooling demand of the power electronics can be met by implementing a fuel-cooled solution. The selected engine conditions typically include any one or more of: sub-idle engine operation (e.g. at engine start-up and windmill relight), low-power engine operation (e.g. idle and descent), transition from end of cruise to top of descent operation, and post-shutdown heat soak back. During the sub-idle, low-power and post-shutdown conditions, the rate of fuel flow through the circuit can be inadequate (non-existent in the case of post-shutdown) to meet the cooling demand of the power electronics. During the end of cruise to top of descent transition, thermal inertia in the engine oil system combined with a reduction of fuel flow can cause a transient fuel temperature increase which temporarily reduces the availability of the fuel as a heat sink to achieve a cooling demand of an engine oil circuit. Thus, during the selected engine conditions, the power electronics can transfer heat to a phase change material via the cooling flow. This flow circulates in the loop such that it continuously transfers heat from the power electronics to the phase change material. The phase change material transitions from one thermodynamic phase (e.g. solid) to a different, second thermodynamic phase (e.g. liquid) at the phase change temperature to absorb the heat rejected from the power electronics. Such a cooling solution can achieve a heat transfer rate of up to around 10 kW from the power electronics to the phase change material, which is sufficient to meet the cooling demand of the power electronics when alternative heat sinks are not available.

Optional features of the invention will now be set out. These are applicable singly or in any combination with any aspect of the invention.

As mentioned above, the selected engine conditions may include any one or more of: sub-idle engine operation (e.g. at engine start-up and windmill relight), low-power engine operation conditions (e.g. idle and descent), operation conditions from end of cruise to top of descent (i.e. during aircraft deceleration), and post-shutdown heat soak back. The fuel circuit may be configured to direct the cooling flow into the fuel flow to the combustor during operation of the engine under other conditions, such as engine operation from idle through cruise to maximum power. Thus, the heat

stored in the phase change material can be transferred to the fuel flow and then eventually to the combustor during engine operations conditions when the fuel flow to the combustor provides an adequate heat sink. In particular, the phase change material can thus be regenerated by transitioning back from the second thermodynamic phase (e.g. liquid) to the first (e.g. solid) so that it is available for heat storage again at a later time. During the other conditions, the cooling flow of the fuel circuit can also continue to remove heat produced by the power electronics.

An air-cooled heat exchanger may be located on the loop to transfer heat from the cooling flow to an airflow produced by the engine. For example, the air-cooled heat exchanger may be an air-cooled surface cooler and the airflow may be a bypass duct airflow produced by a fan of a turbofan engine, or an airflow in an air intake or around a nacelle of the engine. An air-cooled surface cooler typically has a plate-like structure, containing a flow channel or channels for flow of cooling fluid, and an array of fins which project therefrom into the airflow. Advantageously, this type of air-cooled heat exchanger is generally able to provide convective air cooling even when the produced airflow is at or close to zero and may be integrated in an aircraft structure to perform a structural function as well as a cooling function. This can increase the cooling capacity of the engine and/or decrease the amount of phase change material required to meet the cooling demand of the power electronics. However, other types of air-cooled heat exchangers, which can transfer sufficient heat from the power electronics via natural convection, may be used.

The phase change temperature of the phase change material may be 100° C. or less, and preferably 80° C. or less. In this way, the power electronics can be prevented from exceeding such a temperature.

The phase change material may be configured to store at least 0.3 MJ of heat (e.g. 1 kW for 5 min), and preferably at least 1 MJ of heat. Advantageously, such a heat storage capacity ensures that the cooling demand of typical aircraft and/or engine power electronics can be met during the selected engine conditions.

The phase change material may conveniently be a silicate salt or paraffin. The phase change material may be doped to enhance its phase change, heat storage, and heat transfer properties.

The power electronics may be for powering an aircraft cabin blower system.

The power electronics may further include internal passages for flow therethrough of the cooling flow. Alternatively or additionally, the power electronics may further include one or more cold plates cooled by the cooling flow for extracting heat produced by the power electronics. Such internal passages and/or cold plates can enhance the transfer of heat from the power electronics to the cooling flow, thereby more reliably and effectively controlling the power electronics' temperature.

The fuel circuit may further include: a low pressure pump configured to pressurize fuel to a low pressure; and a high pressure pump configured to receive the low pressure fuel and increase the pressure to a high pressure for onward supply to a fuel metering system. In this case, the fuel circuit can be configured such that the loop is formed between an outlet of the low pressure pump and an inlet of the high pressure pump. Relative to using a cooling flow from fuel upstream of the low pressure pump, forming the loop between an outlet of the low pressure pump and an inlet of the high pressure pump avoids compromising the suction capability of the low pressure pump, as the cooling flow

(having an increased temperature after cooling the power electronics) is not returned to the inlet of the low pressure pump.

The engine may further include a fuel-oil heat exchanger having a fuel side on the fuel circuit between the low pressure pump and the high pressure pump, and an oil side on an oil circuit for lubricating and cooling components of the engine, the fuel-oil heat exchanger transferring heat from the oil circuit to the fuel circuit; the fuel circuit being configured such that the loop is formed between an outlet of the low pressure pump and the fuel side of the fuel-oil heat exchanger. Thus, the heat transfer from the power electronics into the fuel flow is performed before the generally greater heat transfer into the fuel flow from the fuel-oil heat exchanger. Were these two heat transfer processes performed in the opposite order, the cooling of the power electronics might be compromised. On the other hand, the relatively small amount of heat eventually transferred into the fuel circuit from the PE compared to that transferred in from the oil circuit does not significantly affect the ability of the fuel circuit to cool the oil from the oil circuit in the fuel-oil heat exchanger.

The gas turbine engine may further include a cooling pump on the loop configured to drive the cooling flow. For example, the cooling pump may conveniently be an electric pump powered directly by the engine or by an aircraft system (e.g. a cabin blower system). Such a cooling pump can help to more reliably control the flow rate of the cooling flow, and thus more reliably and effectively control the power electronics' temperature.

As noted elsewhere herein, the present disclosure relates to a gas turbine engine. Such a gas turbine engine comprises an engine core comprising a turbine, a combustor, a compressor, and a core shaft connecting the turbine to the compressor. Such a gas turbine engine may comprise a fan (having fan blades) located upstream of the engine core.

Arrangements of the present disclosure may be particularly, although not exclusively, beneficial for fans that are driven via a gearbox. Accordingly, the gas turbine engine may comprise a gearbox that receives an input from the core shaft and outputs drive to the fan so as to drive the fan at a lower rotational speed than the core shaft. The input to the gearbox may be directly from the core shaft, or indirectly from the core shaft, for example via a spur shaft and/or gear. The core shaft may rigidly connect the turbine and the compressor, such that the turbine and compressor rotate at the same speed (with the fan rotating at a lower speed).

The gas turbine engine as described and/or claimed herein may have any suitable general architecture. For example, the gas turbine engine may have any desired number of shafts that connect turbines and compressors, for example one, two or three shafts. Purely by way of example, the turbine connected to the core shaft may be a first turbine, the compressor connected to the core shaft may be a first compressor, and the core shaft may be a first core shaft. The engine core may further comprise a second turbine, a second compressor, and a second core shaft connecting the second turbine to the second compressor. The second turbine, second compressor, and second core shaft may be arranged to rotate at a higher rotational speed than the first core shaft.

In such an arrangement, the second compressor may be positioned axially downstream of the first compressor. The second compressor may be arranged to receive (for example directly receive, for example via a generally annular duct) flow from the first compressor.

The gearbox may be arranged to be driven by the core shaft that is configured to rotate (for example in use) at the

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lowest rotational speed (for example the first core shaft in the example above). For example, the gearbox may be arranged to be driven only by the core shaft that is configured to rotate (for example in use) at the lowest rotational speed (for example only be the first core shaft, and not the second core shaft, in the example above). Alternatively, the gearbox may be arranged to be driven by any one or more shafts, for example the first and/or second shafts in the example above.

The gearbox may be a reduction gearbox (in that the output to the fan is a lower rotational rate than the input from the core shaft). Any type of gearbox may be used. For example, the gearbox may be a “planetary” or “star” gearbox, as described in more detail elsewhere herein. The gearbox may have any desired reduction ratio (defined as the rotational speed of the input shaft divided by the rotational speed of the output shaft), for example greater than 2.5, for example in the range of from 3 to 4.2, or 3.2 to 3.8, for example on the order of or at least 3, 3.1, 3.2, 3.3, 3.4, 3.5, 3.6, 3.7, 3.8, 3.9, 4, 4.1 or 4.2. The gear ratio may be, for example, between any two of the values in the previous sentence. Purely by way of example, the gearbox may be a “star” gearbox having a ratio in the range of from 3.1 or 3.2 to 3.8. In some arrangements, the gear ratio may be outside these ranges.

In any gas turbine engine as described and/or claimed herein, the combustor may be provided axially downstream of the fan and compressor(s). For example, the combustor may be directly downstream of (for example at the exit of) the second compressor, where a second compressor is provided. By way of further example, the flow at the exit to the combustor may be provided to the inlet of the second turbine, where a second turbine is provided. The combustor may be provided upstream of the turbine(s).

The or each compressor (for example the first compressor and second compressor as described above) may comprise any number of stages, for example multiple stages. Each stage may comprise a row of rotor blades and a row of stator vanes, which may be variable stator vanes (in that their angle of incidence may be variable). The row of rotor blades and the row of stator vanes may be axially offset from each other.

The or each turbine (for example the first turbine and second turbine as described above) may comprise any number of stages, for example multiple stages. Each stage may comprise a row of rotor blades and a row of stator vanes. The row of rotor blades and the row of stator vanes may be axially offset from each other.

Each fan blade may be defined as having a radial span extending from a root (or hub) at a radially inner gas-washed location, or 0% span position, to a tip at a 100% span position. The ratio of the radius of the fan blade at the hub to the radius of the fan blade at the tip may be less than (or on the order of) any of: 0.4, 0.39, 0.38, 0.37, 0.36, 0.35, 0.34, 0.33, 0.32, 0.31, 0.3, 0.29, 0.28, 0.27, 0.26, or 0.25. The ratio of the radius of the fan blade at the hub to the radius of the fan blade at the tip may be in an inclusive range bounded by any two of the values in the previous sentence (i.e. the values may form upper or lower bounds), for example in the range of from 0.28 to 0.32. These ratios may commonly be referred to as the hub-to-tip ratio. The radius at the hub and the radius at the tip may both be measured at the leading edge (or axially forwardmost) part of the blade. The hub-to-tip ratio refers, of course, to the gas-washed portion of the fan blade, i.e. the portion radially outside any platform.

The radius of the fan may be measured between the engine centreline and the tip of a fan blade at its leading edge. The fan diameter (which may simply be twice the

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radius of the fan) may be greater than (or on the order of) any of: 220 cm, 230 cm, 240 cm, 250 cm (around 100 inches), 260 cm, 270 cm (around 105 inches), 280 cm (around 110 inches), 290 cm (around 115 inches), 300 cm (around 120 inches), 310 cm, 320 cm (around 125 inches), 330 cm (around 130 inches), 340 cm (around 135 inches), 350 cm, 360 cm (around 140 inches), 370 cm (around 145 inches), 380 (around 150 inches) cm, 390 cm (around 155 inches), 400 cm, 410 cm (around 160 inches) or 420 cm (around 165 inches). The fan diameter may be in an inclusive range bounded by any two of the values in the previous sentence (i.e. the values may form upper or lower bounds), for example in the range of from 240 cm to 280 cm or 330 cm to 380 cm.

The rotational speed of the fan may vary in use. Generally, the rotational speed is lower for fans with a higher diameter. Purely by way of non-limitative example, the rotational speed of the fan at cruise conditions may be less than 2500 rpm, for example less than 2300 rpm. Purely by way of further non-limitative example, the rotational speed of the fan at cruise conditions for an engine having a fan diameter in the range of from 220 cm to 300 cm (for example 240 cm to 280 cm or 250 cm to 270 cm) may be in the range of from 1700 rpm to 2500 rpm, for example in the range of from 1800 rpm to 2300 rpm, for example in the range of from 1900 rpm to 2100 rpm. Purely by way of further non-limitative example, the rotational speed of the fan at cruise conditions for an engine having a fan diameter in the range of from 330 cm to 380 cm may be in the range of from 1200 rpm to 2000 rpm, for example in the range of from 1300 rpm to 1800 rpm, for example in the range of from 1400 rpm to 1800 rpm.

In use of the gas turbine engine, the fan (with associated fan blades) rotates about a rotational axis. This rotation results in the tip of the fan blade moving with a velocity  $U_{tip}$ . The work done by the fan blades on the flow results in an enthalpy rise  $dH$  of the flow. A fan tip loading may be defined as  $dH/U_{tip}^2$ , where  $dH$  is the enthalpy rise (for example the 1-D average enthalpy rise) across the fan and  $U_{tip}$  is the (translational) velocity of the fan tip, for example at the leading edge of the tip (which may be defined as fan tip radius at leading edge multiplied by angular speed). The fan tip loading at cruise conditions may be greater than (or on the order of) any of: 0.28, 0.29, 0.30, 0.31, 0.32, 0.33, 0.34, 0.35, 0.36, 0.37, 0.38, 0.39 or 0.4 (all values being dimensionless). The fan tip loading may be in an inclusive range bounded by any two of the values in the previous sentence (i.e. the values may form upper or lower bounds), for example in the range of from 0.28 to 0.31, or 0.29 to 0.3.

Gas turbine engines in accordance with the present disclosure may have any desired bypass ratio, where the bypass ratio is defined as the ratio of the mass flow rate of the flow through the bypass duct to the mass flow rate of the flow through the core at cruise conditions. In some arrangements the bypass ratio may be greater than (or on the order of) any of the following: 10, 10.5, 11, 11.5, 12, 12.5, 13, 13.5, 14, 14.5, 15, 15.5, 16, 16.5, 17, 17.5, 18, 18.5, 19, 19.5 or 20. The bypass ratio may be in an inclusive range bounded by any two of the values in the previous sentence (i.e. the values may form upper or lower bounds), for example in the range of from 12 to 16, 13 to 15, or 13 to 14. The bypass duct may be substantially annular. The bypass duct may be radially outside the core engine. The radially outer surface of the bypass duct may be defined by a nacelle and/or a fan case.

The overall pressure ratio of a gas turbine engine as described and/or claimed herein may be defined as the ratio of the stagnation pressure upstream of the fan to the stag-



nation pressure at the exit of the highest pressure compressor (before entry into the combustor). By way of non-limitative example, the overall pressure ratio of a gas turbine engine as described and/or claimed herein at cruise may be greater than (or on the order of) any of the following: 35, 40, 45, 50, 55, 60, 65, 70, 75. The overall pressure ratio may be in an inclusive range bounded by any two of the values in the previous sentence (i.e. the values may form upper or lower bounds), for example in the range of from 50 to 70.

Specific thrust of an engine may be defined as the net thrust of the engine divided by the total mass flow through the engine. At cruise conditions, the specific thrust of an engine described and/or claimed herein may be less than (or on the order of) any of the following:  $110 \text{ Nkg}^{-1} \text{ s}$ ,  $105 \text{ Nkg}^{-1} \text{ s}$ ,  $100 \text{ Nkg}^{-1} \text{ s}$ ,  $95 \text{ Nkg}^{-1} \text{ s}$ ,  $90 \text{ Nkg}^{-1} \text{ s}$ ,  $85 \text{ Nkg}^{-1} \text{ s}$  or  $80 \text{ Nkg}^{-1} \text{ s}$ . The specific thrust may be in an inclusive range bounded by any two of the values in the previous sentence (i.e. the values may form upper or lower bounds), for example in the range of from  $80 \text{ Nkg}^{-1} \text{ s}$  to  $100 \text{ Nkg}^{-1} \text{ s}$ , or  $85 \text{ Nkg}^{-1} \text{ s}$  to  $95 \text{ Nkg}^{-1} \text{ s}$ . Such engines may be particularly efficient in comparison with conventional gas turbine engines.

A gas turbine engine as described and/or claimed herein may have any desired maximum thrust. Purely by way of non-limitative example, a gas turbine as described and/or claimed herein may be capable of producing a maximum thrust of at least (or on the order of) any of the following: 160 kN, 170 kN, 180 kN, 190 kN, 200 kN, 250 kN, 300 kN, 350 kN, 400 kN, 450 kN, 500 kN, or 550 kN. The maximum thrust may be in an inclusive range bounded by any two of the values in the previous sentence (i.e. the values may form upper or lower bounds). Purely by way of example, a gas turbine as described and/or claimed herein may be capable of producing a maximum thrust in the range of from 330 kN to 420 kN, for example 350 kN to 400 kN. The thrust referred to above may be the maximum net thrust at standard atmospheric conditions at sea level plus 15 degrees C. (ambient pressure 101.3 kPa, temperature 30 degrees C.), with the engine static.

In use, the temperature of the flow at the entry to the high pressure turbine may be particularly high. This temperature, which may be referred to as TET, may be measured at the exit to the combustor, for example immediately upstream of the first turbine vane, which itself may be referred to as a nozzle guide vane. At cruise, the TET may be at least (or on the order of) any of the following: 1400K, 1450K, 1500K, 1550K, 1600K or 1650K. The TET at cruise may be in an inclusive range bounded by any two of the values in the previous sentence (i.e. the values may form upper or lower bounds). The maximum TET in use of the engine may be, for example, at least (or on the order of) any of the following: 1700K, 1750K, 1800K, 1850K, 1900K, 1950K or 2000K. The maximum TET may be in an inclusive range bounded by any two of the values in the previous sentence (i.e. the values may form upper or lower bounds), for example in the range of from 1800K to 1950K. The maximum TET may occur, for example, at a high thrust condition, for example at a maximum take-off (MTO) condition.

A fan blade and/or aerofoil portion of a fan blade described and/or claimed herein may be manufactured from any suitable material or combination of materials. For example at least a part of the fan blade and/or aerofoil may be manufactured at least in part from a composite, for example a metal matrix composite and/or an organic matrix composite, such as carbon fibre. By way of further example at least a part of the fan blade and/or aerofoil may be manufactured at least in part from a metal, such as a titanium

based metal or an aluminium based material (such as an aluminium-lithium alloy) or a steel based material. The fan blade may comprise at least two regions manufactured using different materials. For example, the fan blade may have a protective leading edge, which may be manufactured using a material that is better able to resist impact (for example from birds, ice or other material) than the rest of the blade. Such a leading edge may, for example, be manufactured using titanium or a titanium-based alloy. Thus, purely by way of example, the fan blade may have a carbon-fibre or aluminium based body (such as an aluminium lithium alloy) with a titanium leading edge.

A fan as described and/or claimed herein may comprise a central portion, from which the fan blades may extend, for example in a radial direction. The fan blades may be attached to the central portion in any desired manner. For example, each fan blade may comprise a fixture which may engage a corresponding slot in the hub (or disc). Purely by way of example, such a fixture may be in the form of a dovetail that may slot into and/or engage a corresponding slot in the hub/disc in order to fix the fan blade to the hub/disc. By way of further example, the fan blades may be formed integrally with a central portion. Such an arrangement may be referred to as a bladed disc or a bladed ring. Any suitable method may be used to manufacture such a bladed disc or bladed ring. For example, at least a part of the fan blades may be machined from a block and/or at least part of the fan blades may be attached to the hub/disc by welding, such as linear friction welding.

The gas turbine engines described and/or claimed herein may or may not be provided with a variable area nozzle (VAN). Such a variable area nozzle may allow the exit area of the bypass duct to be varied in use. The general principles of the present disclosure may apply to engines with or without a VAN.

The fan of a gas turbine as described and/or claimed herein may have any desired number of fan blades, for example 14, 16, 18, 20, 22, 24 or 26 fan blades.

As used herein, cruise conditions have the conventional meaning and would be readily understood by the skilled person. Thus, for a given gas turbine engine for an aircraft, the skilled person would immediately recognise cruise conditions to mean the operating point of the engine at mid-cruise of a given mission (which may be referred to in the industry as the "economic mission") of an aircraft to which the gas turbine engine is designed to be attached. In this regard, mid-cruise is the point in an aircraft flight cycle at which 50% of the total fuel that is burned between top of climb and start of descent has been burned (which may be approximated by the midpoint—in terms of time and/or distance—between top of climb and start of descent. Cruise conditions thus define an operating point of, the gas turbine engine that provides a thrust that would ensure steady state operation (i.e. maintaining a constant altitude and constant Mach Number) at mid-cruise of an aircraft to which it is designed to be attached, taking into account the number of engines provided to that aircraft. For example where an engine is designed to be attached to an aircraft that has two engines of the same type, at cruise conditions the engine provides half of the total thrust that would be required for steady state operation of that aircraft at mid-cruise.

In other words, for a given gas turbine engine for an aircraft, cruise conditions are defined as the operating point of the engine that provides a specified thrust (required to provide—in combination with any other engines on the aircraft—steady state operation of the aircraft to which it is designed to be attached at a given mid-cruise Mach Number)

at the mid-cruise atmospheric conditions (defined by the International Standard Atmosphere according to ISO 2533 at the mid-cruise altitude). For any given gas turbine engine for an aircraft, the mid-cruise thrust, atmospheric conditions and Mach Number are known, and thus the operating point of the engine at cruise conditions is clearly defined.

Purely by way of example, the forward speed at the cruise condition may be any point in the range of from Mach 0.7 to 0.9, for example 0.75 to 0.85, for example 0.76 to 0.84, for example 0.77 to 0.83, for example 0.78 to 0.82, for example 0.79 to 0.81, for example on the order of Mach 0.8, on the order of Mach 0.85 or in the range of from 0.8 to 0.85. Any single speed within these ranges may be part of the cruise condition. For some aircraft, the cruise conditions may be outside these ranges, for example below Mach 0.7 or above Mach 0.9.

Purely by way of example, the cruise conditions may correspond to standard atmospheric conditions (according to the International Standard Atmosphere, ISA) at an altitude that is in the range of from 10000 m to 15000 m, for example in the range of from 10000 m to 12000 m, for example in the range of from 10400 m to 11600 m (around 38000 ft), for example in the range of from 10500 m to 11500 m, for example in the range of from 10600 m to 11400 m, for example in the range of from 10700 m (around 35000 ft) to 11300 m, for example in the range of from 10800 m to 11200 m, for example in the range of from 10900 m to 11100 m, for example on the order of 11000 m. The cruise conditions may correspond to standard atmospheric conditions at any given altitude in these ranges.

Purely by way of example, the cruise conditions may correspond to an operating point of the engine that provides a known required thrust level (for example a value in the range of from 30 kN to 35 kN) at a forward Mach number of 0.8 and standard atmospheric conditions (according to the International Standard Atmosphere) at an altitude of 38000 ft (11582 m). Purely by way of further example, the cruise conditions may correspond to an operating point of the engine that provides a known required thrust level (for example a value in the range of from 50 kN to 65 kN) at a forward Mach number of 0.85 and standard atmospheric conditions (according to the International Standard Atmosphere) at an altitude of 35000 ft (10668 m).

In use, a gas turbine engine described and/or claimed herein may operate at the cruise conditions defined elsewhere herein. Such cruise conditions may be determined by the cruise conditions (for example the mid-cruise conditions) of an aircraft to which at least one (for example 2 or 4) gas turbine engine may be mounted in order to provide propulsive thrust.

According to an aspect, there is provided an aircraft comprising a gas turbine engine as described and/or claimed herein. The aircraft according to this aspect is the aircraft for which the gas turbine engine has been designed to be attached. Accordingly, the cruise conditions according to this aspect correspond to the mid-cruise of the aircraft, as defined elsewhere herein.

According to an aspect, there is provided a method of operating a gas turbine engine as described and/or claimed herein. The operation may be at the cruise conditions as defined elsewhere herein (for example in terms of the thrust, atmospheric conditions and Mach Number).

According to an aspect, there is provided a method of operating an aircraft comprising a gas turbine engine as described and/or claimed herein. The operation according to this aspect may include (or may be) operation at the mid-cruise of the aircraft, as defined elsewhere herein.

The skilled person will appreciate that except where mutually exclusive, a feature or parameter described in relation to any one of the above aspects may be applied to any other aspect. Furthermore, except where mutually exclusive, any feature or parameter described herein may be applied to any aspect and/or combined with any other feature or parameter described herein.

## BRIEF DESCRIPTION OF THE DRAWINGS

Embodiments illustrating the principles of the invention will now be discussed with reference to the accompanying figures in which:

FIG. 1 is a sectional side view of a gas turbine engine;

FIG. 2 is a close-up sectional side view of an upstream portion of a gas turbine engine;

FIG. 3 is a partially cut-away view of a gearbox for a gas turbine engine; and

FIGS. 4 and 5 are schematic diagrams of respective variants of a part of a fuel circuit of the gas turbine engine.

## DETAILED DESCRIPTION OF THE DISCLOSURE

Aspects and embodiments of the present invention will now be discussed with reference to the accompanying figures. Further aspects and embodiments will be apparent to those skilled in the art. All documents mentioned in this text are incorporated herein by reference.

FIG. 1 illustrates a gas turbine engine 10 having a principal rotational axis 9. The engine 10 comprises an air intake 12 and a propulsive fan 23 that generates two airflows: a core airflow A and a bypass airflow B. The gas turbine engine 10 comprises a core 11 that receives the core airflow A. The engine core 11 comprises, in axial flow series, a low pressure compressor 14, a high pressure compressor 15, combustion equipment 16, a high pressure turbine 17, a low pressure turbine 19 and a core exhaust nozzle 20. A nacelle 21 surrounds the gas turbine engine 10 and defines a bypass duct 22 and a bypass exhaust nozzle 18. The bypass airflow B flows through the bypass duct 22. The fan 23 is attached to and driven by the low pressure turbine 19 via a shaft 26 and an epicyclic gearbox 30.

In use, the core airflow A is accelerated and compressed by the low pressure compressor 14 and directed into the high pressure compressor 15 where further compression takes place. The compressed air exhausted from the high pressure compressor 15 is directed into the combustion equipment 16 where it is mixed with fuel and the mixture is combusted. The resultant hot combustion products then expand through, and thereby drive, the high pressure and low pressure turbines 17, 19 before being exhausted through the nozzle 20 to provide some propulsive thrust. The high pressure turbine 17 drives the high pressure compressor 15 by a suitable interconnecting shaft 27. The fan 23 generally provides the majority of the propulsive thrust. The epicyclic gearbox 30 is a reduction gearbox.

An exemplary arrangement for a geared fan gas turbine engine 10 is shown in FIG. 2. The low pressure turbine 19 (see FIG. 1) drives the shaft 26, which is coupled to a sun wheel, or sun gear, 28 of the epicyclic gear arrangement 30. Radially outwardly of the sun gear 28 and intermeshing therewith is a plurality of planet gears 32 that are coupled together by a planet carrier 34. The planet carrier 34 constrains the planet gears 32 to precess around the sun gear 28 in synchronicity whilst enabling each planet gear 32 to rotate about its own axis. The planet carrier 34 is coupled via

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linkages 36 to the fan 23 in order to drive its rotation about the engine axis 9. Radially outwardly of the planet gears 32 and intermeshing therewith is an annulus or ring gear 38 that is coupled, via linkages 40, to a stationary supporting structure 24.

Note that the terms “low pressure turbine” and “low pressure compressor” as used herein may be taken to mean the lowest pressure turbine stages and lowest pressure compressor stages (i.e. not including the fan 23) respectively and/or the turbine and compressor stages that are connected together by the interconnecting shaft 26 with the lowest rotational speed in the engine (i.e. not including the gearbox output shaft that drives the fan 23). In some literature, the “low pressure turbine” and “low pressure compressor” referred to herein may alternatively be known as the “intermediate pressure turbine” and “intermediate pressure compressor”. Where such alternative nomenclature is used, the fan 23 may be referred to as a first, or lowest pressure, compression stage.

The epicyclic gearbox 30 is shown by way of example in greater detail in FIG. 3. Each of the sun gear 28, planet gears 32 and ring gear 38 comprise teeth about their periphery to intermesh with the other gears. However, for clarity only exemplary portions of the teeth are illustrated in FIG. 3. There are four planet gears 32 illustrated, although it will be apparent to the skilled reader that more or fewer planet gears 32 may be provided within the scope of the claimed invention. Practical applications of a planetary epicyclic gearbox 30 generally comprise at least three planet gears 32.

The epicyclic gearbox 30 illustrated by way of example in FIGS. 2 and 3 is of the planetary type, in that the planet carrier 34 is coupled to an output shaft via linkages 36, with the ring gear 38 fixed. However, any other suitable type of epicyclic gearbox 30 may be used. By way of further example, the epicyclic gearbox 30 may be a star arrangement, in which the planet carrier 34 is held fixed, with the ring (or annulus) gear 38 allowed to rotate. In such an arrangement the fan 23 is driven by the ring gear 38. By way of further alternative example, the gearbox 30 may be a differential gearbox in which the ring gear 38 and the planet carrier 34 are both allowed to rotate.

It will be appreciated that the arrangement shown in FIGS. 2 and 3 is by way of example only, and various alternatives are within the scope of the present disclosure. Purely by way of example, any suitable arrangement may be used for locating the gearbox 30 in the engine 10 and/or for connecting the gearbox 30 to the engine 10. By way of further example, the connections (such as the linkages 36, 40 in the FIG. 2 example) between the gearbox 30 and other parts of the engine 10 (such as the input shaft 26, the output shaft and the fixed structure 24) may have any desired degree of stiffness or flexibility. By way of further example, any suitable arrangement of the bearings between rotating and stationary parts of the engine (for example between the input and output shafts from the gearbox and the fixed structures, such as the gearbox casing) may be used, and the disclosure is not limited to the exemplary arrangement of FIG. 2. For example, where the gearbox 30 has a star arrangement (described above), the skilled person would readily understand that the arrangement of output and support linkages and bearing locations would typically be different to that shown by way of example in FIG. 2.

Accordingly, the present disclosure extends to a gas turbine engine having any arrangement of gearbox styles (for example star or planetary), support structures, input and output shaft arrangement, and bearing locations.

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Optionally, the gearbox may drive additional and/or alternative components (e.g. the intermediate pressure compressor and/or a booster compressor).

Other gas turbine engines to which the present disclosure may be applied may have alternative configurations. For example, such engines may have an alternative number of compressors and/or turbines and/or an alternative number of interconnecting shafts. By way of further example, the gas turbine engine shown in FIG. 1 has a split flow nozzle 18, 20 meaning that the flow through the bypass duct 22 has its own nozzle 18 that is separate to and radially outside the core engine nozzle 20. However, this is not limiting, and any aspect of the present disclosure may also apply to engines in which the flow through the bypass duct 22 and the flow through the core 11 are mixed, or combined, before (or upstream of) a single nozzle, which may be referred to as a mixed flow nozzle. One or both nozzles (whether mixed or split flow) may have a fixed or variable area. Whilst the described example relates to a turbofan engine, the disclosure may apply, for example, to any type of gas turbine engine, such as an open rotor (in which the fan stage is not surrounded by a nacelle) or turboprop engine, for example. In some arrangements, the gas turbine engine 10 may not comprise a gearbox 30.

The geometry of the gas turbine engine 10, and components thereof, is defined by a conventional axis system, comprising an axial direction (which is aligned with the rotational axis 9), a radial direction (in the bottom-to-top direction in FIG. 1), and a circumferential direction (perpendicular to the page in the FIG. 1 view). The axial, radial and circumferential directions are mutually perpendicular.

The gas turbine engine 10 of FIGS. 1 and 2 further has: an oil circuit configured to cool and lubricate bearings of the engine core 11, and gears and bearings of the power gear box (PGB) 30, and a fuel circuit 50 for supplying fuel to the combustor 16. Respective variants of a part of the fuel circuit 50 are shown in the schematic diagrams of FIGS. 4 and 5, the variant of FIG. 5 differing from that of FIG. 4 in that it also has an air-cooled heat exchanger 56 (discussed in more detail below).

During engine operation, oil from an engine oil tank provides a supply of pressurised oil for the oil circuit (not shown). Typically, the oil is drawn from the tank through a strainer, and then pumped through a filter by an oil pump having a pressure-limiting valve (none of which is shown in FIG. 4). The oil is used to cool and lubricate the bearings of the engine core 11 and the PGB 30. Having performed its lubricating and cooling function, the oil is directed to a sump and returned to the oil tank by scavenge pumps.

To prevent the oil overheating such that it would fail to adequately perform its cooling and lubricating functions, heat from the oil circuit is transferred to engine fluid heat sinks, i.e. fuel and airflows at various heat exchangers. For example, heat is transferred from the oil circuit to the fuel circuit 50 via the fuel-cooled oil cooler (FCOC) 57 shown in FIGS. 4 and 5. The transfer of heat from the oil circuit also protects the fuel from forming fuel-borne ice particles. Furthermore, the exchange of heat between the oil and fuel 50 circuits helps to reduce thermodynamic losses in the engine 10.

The fuel circuit 50 shown in FIGS. 4 and 5 supplies fuel from an aircraft fuel tank 51 to the combustor 16 of FIG. 1 for combustion. During operation, fuel drawn from the aircraft fuel tank is pressurised to a low pressure by a low pressure (LP) pump 52. The low pressure fuel is then directed to a high pressure (HP) pump which increases the pressure of the fuel to a high pressure and supplies it to a fuel

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metering system (not shown) which controls the engine-consumed flow in response to a fuel demand indicated by an electronic engine controller (EEC, not shown). The fuel side of the FCOC 57 is between the outlet of the LP pump 52 and the inlet of the HP pump.

The aircraft and/or the engine have power electronics (PE) 53, such as for powering a cabin blower (CB) system, or powering aircraft control actuators. The PE generates heat and requires thermal management, in the form of active cooling, to ensure it does not exceed a temperature which could impair its performance. The PE can be mounted to the engine or it may be mounted at a convenient location on the aircraft at a distance from the engine.

For many engine operating conditions, such as from idle through cruise to maximum power, engine fluid heat sinks are available for extracting excess heat from the PE. For example, heat can be transferred into fuel flow in the fuel circuit, the fuel then being burnt in the combustor, or heat can be transferred into airflows such as airflow B through the bypass duct 22. However, under other engine conditions these heat sinks are unavailable or inadequate to meet the cooling demand of the PE, e.g. because a shut off valve in the fuel metering system is closed or the heat load on the fuel circuit is already too high. Examples of such conditions are sub-idle engine operation (e.g. at engine start up and wind-mill relight), low-power engine operation conditions (e.g. idle and descent), transient operation from end of cruise to top of descent, and post-shutdown heat soak back. Accordingly, for these engine conditions the fuel circuit is configured such that an electric cooling pump 55 circulates a portion of the low pressure fuel around a loop to the PE and a phase change material (PCM) 54, the cooling flow thereby transferring heat from the PE to the PCM. The loop is formed in the fuel circuit between an outlet of the LP pump 52 and an inlet of the HP pump. The direction of circulation of the cooling flow around the loop is indicated by the anti-clockwise arrow in FIGS. 4 and 5.

The location of the cooling pump 55 in the direction of flow around the loop is preferably after the PE 53, but alternatively, it can also be before the PE 53, as indicated by the dashed-line pump 55' in FIGS. 4 and 5. The electric cooling pump can be powered directly by the engine or by the CB system of the aircraft. The pump controls the flow rate of the cooling flow to reliably control the PE's temperature.

The PCM 54 has a phase change temperature at a predetermined limiting temperature, which can be 100° C. or less, and is preferably 80° C. or less. Conveniently it can be a doped or undoped silicate salt or paraffin. The PCM transitions from one thermodynamic phase (e.g. solid) to a different, second thermodynamic phase (e.g. liquid) to store the heat rejected from the PE. Preferably, the PCM can store at least 0.3 MJ of heat, and more preferably at least 1 MJ. In this way, it ensures that the temperature of the PE does not exceed approximately the phase change temperature and rates of PE temperature change are reduced, even during engine conditions when an engine fluid heat sink is unavailable or inadequate to meet the cooling demand of the PE.

During engine operation conditions (i.e. idle through cruise to maximum power) when the fuel flow in the fuel circuit 50 to the combustor 16 is capable of acting as a heat sink sufficient to meet the cooling demand of the PE 53, the PE is cooled by this fuel flow, and moreover the heat stored in the PCM 54 is transferred to the fuel flow. This allows the PCM to be regenerated, i.e. its stored heat is discharged and it transitions from the second thermodynamic phase (e.g.

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liquid) back to its original, first thermodynamic phase (e.g. solid) such that it is available to store heat again at a later time.

To enhance the heat transfer between the PE 53 and the cooling flow, the PE can be provided with internal passages through which the cooling flow can pass and/or with one or more cold plates cooled by the cooling flow. Additionally, as shown in FIG. 5, cooling of the PE can be supplemented by an air-cooled heat exchanger 56 positioned to transfer heat produced by the PE to the bypass duct (BPD) airflow B produced by the fan 23 or to an airflow in the air intake 12 or around the nacelle 21 of the engine. Preferably, the air-cooled heat exchanger is implemented as an air-cooled surface cooler. Advantageously, air-cooled surface coolers may be integrated in an aircraft structure to perform a structural function as well as a cooling function and are generally able to provide convective air cooling even when the produced airflow is at or close to zero. This can increase the cooling capacity of the engine 10 and/or decrease the amount of PCM 54 required to meet the cooling demand of the PE. However, other types of air-cooled heat exchangers, which can transfer sufficient heat from the PE via natural convection may be used.

The cooling arrangements of FIGS. 4 and 5 can achieve an active cooling rate of as much as about 10 kW to meet the cooling demand of the PE 53 during the selected engine conditions when the fuel flow to the combustor is unavailable or inadequate. Furthermore, relative to using a cooling flow from fuel upstream of the low pressure pump, the cooling arrangements of FIGS. 4 and 5 avoid compromising the suction capability of the LP pump 52, as the cooling flow (having an increased temperature after cooling the PE) is not returned to the inlet of the LP pump. In addition, the heat transfer from the PE into the cooling flow is performed before the generally greater heat transfer from the oil of the oil circuit into the fuel at the FCOC 57. Were the FCOC located upstream of or on the circulating loop to the PE, the cooling of the PE could be compromised. On the other hand, the relatively small amount of heat transferred into the fuel circuit from the PE compared to that transferred in from the oil circuit does not significantly affect the ability of the FCOC to cool the oil when located downstream of the circulating loop to the PE.

Modifications of the fuel circuits shown in FIGS. 4 and 5 are possible. For example, the relative arrangement of the cooling pump 55, PE 53, PCM 54, and/or the air-cooled surface cooler 56 can be altered as long as it achieves the same technical effect.

The features disclosed in the foregoing description, or in the following claims, or in the accompanying drawings, expressed in their specific forms or in terms of a means for performing the disclosed function, or a method or process for obtaining the disclosed results, as appropriate, may, separately, or in any combination of such features, be utilised for realising the invention in diverse forms thereof.

While the invention has been described in conjunction with the exemplary embodiments described above, many equivalent modifications and variations will be apparent to those skilled in the art when given this disclosure. Accordingly, the exemplary embodiments of the invention set forth above are considered to be illustrative and not limiting. Various changes to the described embodiments may be made without departing from the spirit and scope of the invention.

For the avoidance of any doubt, any theoretical explanations provided herein are provided for the purposes of

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improving the understanding of a reader. The inventors do not wish to be bound by any of these theoretical explanations.

Any section headings used herein are for organizational purposes only and are not to be construed as limiting the subject matter described.

Throughout this specification, including the claims which follow, unless the context requires otherwise, the word “comprise” and “include”, and variations such as “comprises”, “comprising”, and “including” will be understood to imply the inclusion of a stated integer or step or group of integers or steps but not the exclusion of any other integer or step or group of integers or steps.

It must be noted that, as used in the specification and the appended claims, the singular forms “a,” “an,” and “the” include plural referents unless the context clearly dictates otherwise. Ranges may be expressed herein as from “about” one particular value, and/or to “about” another particular value. When such a range is expressed, another embodiment includes from the one particular value and/or to the other particular value. Similarly, when values are expressed as approximations, by the use of the antecedent “about,” it will be understood that the particular value forms another embodiment. The term “about” in relation to a numerical value is optional and means for example  $\pm 10\%$ .

I claim:

1. A combination of a gas turbine engine and power electronics for powering aircraft and/or engine systems, wherein

the engine includes:

an engine core comprising a turbine, a combustor, a compressor, and a core shaft connecting the turbine to the compressor; and

a fuel circuit for supplying a fuel flow to the combustor; the fuel circuit includes:

a fuel tank;

a low pressure pump arranged downstream of the fuel tank and configured to pressurise fuel to a low pressure;

a high pressure pump configured to receive the low pressure fuel and increase the pressure to a high pressure for onward supply to a fuel metering system;

a loop configured to circulate a portion of the low pressure fuel in the loop during selected engine conditions;

a phase change material located on the loop, such that the portion of the low pressure fuel transfers heat from the power electronics to the phase change material, the phase change material having a phase change temperature at a predetermined limiting temperature whereby the phase change material stores heat from the portion of the low pressure fuel, which prevents the power electronics from exceeding the limiting temperature;

a cooling pump on the loop configured to drive only the portion of the low pressure fuel; and

the loop is formed downstream of an outlet of the low pressure pump and upstream of an inlet of the high pressure pump.

2. The combination of the gas turbine engine and the power electronics according to claim 1, wherein the selected

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engine conditions include any one or more of: sub-idle engine operation, low-power engine operation, transition from end of cruise to top of descent operation, and post-shutdown heat soak back.

3. The combination of the gas turbine engine and the power electronics according to claim 1, wherein the fuel circuit is configured to direct the portion of the low pressure fuel into the combustor during operation of the engine under other conditions.

4. The combination of the gas turbine engine and the power electronics according to claim 1, wherein an air-cooled heat exchanger is located on the loop to transfer heat from the portion of the low pressure fuel to an airflow produced by the engine.

5. The combination of the gas turbine engine and the power electronics according to claim 1, wherein the phase change temperature of the phase change material is  $100^{\circ}\text{C}$ . or less.

6. The combination of the gas turbine engine and the power electronics according to claim 1, wherein the phase change material is configured to store at least 0.3 MJ of heat.

7. The combination of the gas turbine engine and the power electronics according to claim 1, wherein the phase change material is a silicate salt or paraffin.

8. The combination of the gas turbine engine and the power electronics according to claim 1, wherein the power electronics further includes internal passages for flow through of the portion of the low pressure fuel.

9. The combination of the gas turbine engine and the power electronics according to claim 1, wherein the power electronics further includes one or more cold plates cooled by the portion of the low pressure fuel for extracting heat produced by the power electronics.

10. The combination of the gas turbine engine and the power electronics according to claim 1, wherein the cooling pump is an electric pump.

11. The combination of the gas turbine engine and the power electronics according to claim 1, wherein the engine further includes: a fan located upstream of the engine core, the fan comprising a plurality of fan blades; and a gearbox that receives an input from the core shaft and outputs drive to the fan so as to drive the fan at a lower rotational speed than the core shaft.

12. The combination of the gas turbine engine and the power electronics according to claim 11, wherein: the turbine is a first turbine, the compressor is a first compressor, and the core shaft is a first core shaft; the engine core further includes a second turbine, a second compressor, and a second core shaft connecting the second turbine to the second compressor; and the second turbine, second compressor, and second core shaft are arranged to rotate at a higher rotational speed than the first core shaft.

13. The combination of the gas turbine engine and the power electronics according to claim 1, wherein the portion of the low pressure fuel is diverted from a fuel line at an offtake upstream of the inlet of the high pressure pump and returned to the fuel line upstream of the offtake.

14. The combination of the gas turbine engine and the power electronics according to claim 13, wherein the portion of the low pressure fuel is returned to the fuel line downstream of the outlet of the low pressure pump.

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