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Thrust control method for an airbreathing jet engine

Abstract

A method of thrust control for an airbreathing jet engine includes obtaining a demanded thrust setting, a value of static pressure at a first axial location of a combustion chamber of the airbreathing jet engine and a value of static pressure at a second axial location of the combustion chamber of the airbreathing jet engine. The second axial location is downstream of the first axial location. The method also includes obtaining a ratio of the value of static pressure at the first axial location of the combustion chamber to the value of static pressure at the second axial location of the combustion chamber; and controlling a fuel flow rate of the airbreathing engine based at least in part on the demanded thrust setting and the ratio of the value of the static pressure at the first axial location to the value of the static pressure at the second axial location.

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Background/Summary

CROSS-REFERENCE TO RELATED APPLICATIONS
(1) This specification is based upon and claims the benefit of priority from United Kingdom patent application number GB 2302422.7 filed on Feb. 21, 2023, the entire contents of which is incorporated herein by reference.
BACKGROUND
1. Field of the Disclosure
(2) The present disclosure concerns a thrust control method for an airbreathing jet engine.
(3) More particularly, the present disclosure concerns a thrust control method for a reheat system based at least in part on pressure measurement at multiple axial locations within the combustion chamber of the reheat system.
(4) The present disclosure also concerns modifications to an airbreathing jet engine to enable to the performance of the method for thrust control.

2. Description of the Related Art

(5) Airbreathing jet engines such as reheat systems are a known means of augmenting the thrust produced by a gas turbine engine for aircraft propulsion.

(6) Other forms of airbreathing jet engine include ramjets.

(7) Although mostly used for military applications, reheat systems have also been used for civil applications, such as the Aérospatiale/British Aircraft Corporation Concorde.

(8) The degree of thrust augmentation produced by a reheat system is typically controlled by closed-loop feedback control of the position or orientation of a component of a reheat system fuel metering valve, such as a flow-metering orifice.

(9) However, as this form of control does not account for differences between the demanded and delivered fuel flow, or the combustion efficiency of the delivered fuel within the reheat duct, this form of control may be less accurate than required, producing more thrust than required and unnecessarily increasing fuel consumption.

(10) There therefore exists a desire to provide a method that overcome these problems or at least provides a useful alternative to known thrust control methods.

SUMMARY OF THE DISCLOSURE

(11) According to a first aspect there is provided a method of thrust control for an airbreathing jet engine that has a combustion chamber. The method comprises obtaining a demanded thrust setting, obtaining a value of static pressure at a first axial location of the combustion chamber of the airbreathing jet engine and obtaining a value of static pressure at a second axial location of the combustion chamber of the airbreathing jet engine. The second axial location is downstream of the first axial location. The method also comprises obtaining a ratio of the value of static pressure at the first axial location of the combustion chamber to the value of static pressure at the second axial location of the combustion chamber and controlling a fuel flow rate of the airbreathing engine based at least in part on the demanded thrust setting and the ratio of the value of the static pressure at the first axial location to the value of the static pressure at the second axial location.

(12) An advantage of the above method is that a more accurate method of control is obtained, relative to thrust control based on demanding a fuel flow to be delivered. This is because differences in static pressure at different locations within the combustion chamber equate to a change in momentum of a working fluid within the combustion chamber, and consequently, the thrust developed by the airbreathing engine. This method also accommodates for potential variables that may influence a fuel flow demand based thrust control method, such as the density of the fuel combusted within the combustion chamber and the combustion efficiency of the fuel when burnt within the combustion chamber.

(13) Optionally, the first axial location is upstream of a fuel injection apparatus within the combustion chamber, and the second axial location is downstream of the fuel injection apparatus of the combustion chamber.

(14) The value of static pressure at the first axial location may be obtained by measuring a static pressure at at least one static pressure offtake in a wall of the combustion chamber at the first axial location.

(15) The value of static pressure at the second axial location may be obtained by measuring a static pressure at at least one static pressure offtake in a wall of the combustion chamber at the second axial location.

(16) The value of static pressure at the first axial location may be an average of the static pressure measured at the at least one static pressure offtake at the first axial location.

(17) The value of static pressure at the second axial location may be an average of the static pressure measured at the at least one static pressure offtake at the second axial location.

(18) An advantage of using averages static pressures is that the stability of the thrust control method may be improved.

(19) The average of static pressure measured at the first axial location may be a fluidic average, or

a numerical average.

(20) The average of static pressure measured at the second axial location may be a fluidic average, or a numerical average.

(21) Fluidic averaging, by coupling a plurality of pressure offtakes at the same axial location together (e.g., a plurality of offtakes circumferentially disposed around a cross-section of the combustion chamber at a fixed axial location), provides a simple averaging means.

(22) Numerical averaging may be performed within a controller and may be combined with other numerical processing functions such as filtering. This may increase the stability of an input to the control method, and consequently, the stability of the control method.

(23) Controlling fuel flow rate may be based at least in part on the demanded thrust setting and a ratio of the value of the static pressure at the first axial location to the value of the static pressure at the second axial location. Controlling fuel flow rate may comprise referencing a look-up table or equation having the demanded thrust setting as an input to determine a demanded ratio of the value of the static pressure at the first axial location to the value of the static pressure at the second axial location from an output of the look-up table, and varying a fuel flow rate to the airbreathing jet engine such that the ratio of the value of the static pressure at the first axial location of the combustion chamber to the value of static pressure at the second axial location of the combustion chamber is substantially equal to the demanded ratio of the value of the static pressure at the first axial location to the value of the static pressure at the second axial location.

(24) Controlling to a pressure ratio has the advantage that the control system method broadly takes account of the flight condition of an aircraft in which the airbreathing engine is comprised and generating thrust.

(25) The demanded ratio of the value of the static pressure at the first axial location to the value of the static pressure at the second axial location may be additionally based upon a flight condition of an aircraft comprising the airbreathing jet engine.

(26) This may be advantageous if the aircraft is operating at or close to a boundary of its operational envelope (forward speed, altitude). This is because when the aircraft is operating at or close to a boundary of its operational envelope, the propulsion system of the aircraft may be subject to additional constraints (for example, a rotational speed limit or maximum pressure limit of a related gas turbine engine), making it desirable to modify the static pressure-ratio based control. Alternatively or additionally, a customer of the aircraft comprising the air breathing engine may have specific thrust requirements for portions of the operational envelope, corresponding to the aircraft flight condition.

(27) The flight condition of the airbreathing engine may comprise at least one of a velocity of the airbreathing jet engine, a static temperature of an atmosphere through which the airbreathing jet engine is travelling, a static pressure of the atmosphere through which the airbreathing jet engine is travelling.

(28) The demanded ratio of the value of the static pressure at the first axial location to the value of the static pressure at the second axial location may be additionally based upon a demanded discharge angle of a jet through a thrust-vectoring propelling nozzle and/or a demanded cross-sectional area of a variable area propelling nozzle.

(29) This is advantageous because the method may additionally compensate for changes in a configuration of the airbreathing engine which may cause a change in static pressure drop between different axial locations of the combustion chamber and/or a change in the static pressure distribution at the first axial location and/or second axial location. For example, actuating a thrust-vectoring nozzle to change the discharge angle may change the static pressure distribution at the second axial location in particular, due to its proximity to the thrust vectoring nozzle.

(30) Controlling the fuel flow rate of the airbreathing jet engine may additionally comprise varying a configuration of a fuel distribution valve to vary the distribution of fuel to different portions of the fuel injection apparatus such that the fuel flow rate provided to the airbreathing jet engine is

minimised for the demanded thrust setting.

(31) This has the advantage that the thrust control method may also reduce specific fuel consumption of the airbreathing engine, by enabling optimisation of fuel distribution within the combustion chamber.

(32) According to a second aspect, there is provided an airbreathing jet engine. The airbreathing engine comprises a combustion chamber and at least one fuel injection apparatus configured to inject fuel into the combustion chamber. The airbreathing jet engine additionally comprises at least one static pressure offtake at a first axial location within the combustion chamber and at least one static pressure offtake at a second axial location within the combustion chamber. The airbreathing jet engine also comprises at least one pressure transducer, the at least one pressure transducer configured to obtain a value of static pressure at the first axial location, and to obtain a value of static pressure at the second axial location. The airbreathing jet engine also comprises a fuel metering valve and a fuel flow controller. The fuel flow controller controls a flow rate of fuel provided by the fuel metering valve to the combustion chamber via the at least one fuel injection apparatus based at least in part on a demanded thrust setting for the airbreathing jet engine and a ratio of a value of the static pressure at the first axial location to a value of the static pressure at the second axial location.

(33) The fuel flow controller may additionally control the flow rate of fuel, based upon a flight condition of an aircraft comprising the airbreathing jet engine and/or a configuration of the airbreathing jet engine.

(34) The airbreathing jet engine may be a reheat system (also known as an afterburner) or a ramjet.

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

(36) The term "airbreathing jet engine" as used herein is a jet engine that generates thrust by ejecting a propelling jet of exhaust through a nozzle. For example, an airbreathing jet engine may be a gas turbine engine, a ramjet, or a reheat system.

(37) The term "reheat system" as used herein means a combustion chamber and propelling nozzle, placed downstream of a gas turbine, Fuel is burnt within the combustion chamber of the reheat system to augment the thrust produced by an upstream gas turbine engine. A reheat system is also known as an afterburner.

(38) Reheat systems and ramjets are forms of athodyd as they are aero thermodynamic dynamic ducts. An athodyd is here defined as a form of jet engine comprising a continuous duct or tube of constant or varying cross-sectional area, into which air is admitted at an upstream entrance, fuel is added and combusted within the engine, and the resulting exhaust gas discharged at a downstream exit.

(39) The term "% augmented thrust" as used herein is defined as the % increase in thrust due to operation of a reheat system relative to the dry thrust produced by a gas turbine engine. Thus, at for example 30% augmented thrust, the total thrust developed by the combination of the reheat system and engine is 130% of the dry thrust developed by the engine in isolation.

(40) The term "fuel:air ratio" as used herein is defined as a gravimetric (mass) ratio of fuel to air.

(41) Throughout this specification and in the claims that follow, unless the context requires otherwise, the word "comprise" or variations such as "comprises" and "comprising", will be understood to imply the inclusion of a stated integer or group of integers but not the exclusion of any other stated integer or group of integers.

Description

BRIEF DESCRIPTION OF THE DRAWINGS

(1) Embodiments will now be described by way of example only, with reference to the drawings, in which:

(2) FIG. **1** is a sectional side view of a gas turbine engine;

(3) FIG. **2** is a sectional side view of a known reheat system;

(4) FIG. **3** is a schematic relationship between % augmented thrust produced by a reheat system, and a fuel:air ratio within the reheat duct of a reheat system;

(5) FIG. **4** is a schematic representation of a sectional side view of a reheat system, comprising jet pipe pressure measurement;

(6) FIG. **5** is a schematic relationship between the pressure ratio at two axial locations within the duct of a reheat system and the % augmented thrust obtained; and,

(7) FIG. **6** is an example method of thrust control for a reheat system.

(8) The following table lists the reference numerals used in the drawings with the features to which they refer:

(9) TABLE-US-00001 Ref no. Feature FIG. 10 Gas turbine engine 1 11 Principal and rotational axis (of engine) 1 4 12 Air intake 1 14 Intermediate pressure compressor 1 15 High pressure compressor 1 16 Combustion equipment 1 17 High pressure turbine 1 18 Intermediate pressure turbine 1 19 Low pressure turbine 1 20 Exhaust nozzle 1 21 Nacelle 1 22 Bypass duct 1 30 Combustion chamber 2 4 31 Fuel injection apparatus 2 4 32 Fuel metering valve 2 4 33 Ignitor 2 4 34 Flame stabiliser 2 4 35 First axial location 4 36 Second axial location 4 37 Pressure transducer 4 38 Static pressure offtake 4 39 Tubing 4 40 Propelling nozzle 2 4 50 Reheat system/Airbreathing jet engine 2 4 60 Thrust to fuel:air ratio relationship 3 61 Lean extinction limit 3 62 Rich extinction limit 3 63 Fuel:air ratio upper permissible operational limit 3 63a Fuel:air ratio at maximum thrust 3 5 64 Upper nominal fuel:air ratio limit 3 65 Offset 3 70 Pressure ratio to thrust relationship 5 71 Direction of increasing fuel:air ratio 5 80 Fuel flow controller 4

DETAILED DESCRIPTION OF THE DISCLOSURE

(10) Aspects and embodiments of the present disclosure will now be discussed with reference to the accompanying drawings. Further aspects and embodiments will be apparent to those skilled in the art.

(11) With reference to FIG. **1**, a gas turbine engine is generally indicated at **10**, having a principal and rotational axis **11**. The engine **10** comprises, in axial flow series, an air intake **12**, a propulsive fan **13**, an intermediate pressure compressor **14**, a high-pressure compressor **15**, combustion equipment **16**, a high-pressure turbine **17**, an intermediate pressure turbine **18**, a low-pressure turbine **19** and an exhaust nozzle **20**. A nacelle **21** generally surrounds the engine **10** and defines both the intake **12** and the exhaust nozzle **20**.

(12) The gas turbine engine **10** works in the conventional manner so that air entering the intake **12** is accelerated by the fan **13** to produce two air flows: a first air flow into the intermediate pressure compressor **14** and a second air flow which passes through a bypass duct **22** to provide propulsive thrust. The intermediate pressure compressor **14** compresses the air flow directed into it before delivering that air to the high pressure compressor **15** where further compression takes place.

(13) 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 combusted. The resultant hot combustion products then expand through, and thereby drive the high, intermediate and low-pressure turbines **17**, **18**, **19** before being exhausted through the nozzle **20** to provide additional propulsive thrust. The high **17**, intermediate **18** and low **19** pressure turbines drive respectively the high pressure compressor **15**, intermediate pressure compressor **14** and fan **13**, each by suitable interconnecting shaft.

(14) Other gas turbine engines to which the present disclosure may be applied may have alternative configurations. By way of example such engines may have an alternative number of

interconnecting shafts (e.g. two) and/or an alternative number of compressors and/or turbines. Further the engine may comprise a gearbox provided in the drive train from a turbine to a compressor and/or fan.

(15) The thrust from the gas turbine engine **10** may be augmented by connecting a reheat system, as an example of air airbreathing jet engine, to the gas turbine engine **10**, downstream of the gas turbine engine **10**. Thus, gas which has passed through the core, and optionally, the bypass duct of the gas turbine engine, subsequently passes through the reheat system **50**.

(16) The following disclosure relates to a method of thrust control of a reheat system **50**, as a form of airbreathing jet engine. The disclosure is also applicable to the control of other forms of airbreathing jet engine, such as a ramjet or scramjet.

(17) As illustrated in FIG. **2**, a known reheat system **50** is substantially tubular, comprising a duct (jet pipe) **30** that is connected to a propelling nozzle **40**. The duct **30** is a combustion chamber, as fuel may be combusted within it, A principal axis of duct **30** may be co-axial or substantially co-axial with principal axis **11** of gas turbine engine **10**.

(18) The propelling nozzle **40** may be a variable area propelling nozzle **40**. The propelling nozzle **40** may be a thrust-vectoring propelling nozzle **40**, configured to control the angle at which a propelling jet is discharged from the propelling nozzle **40** relative to principal axis **11**.

(19) The reheat system **50** comprises at least one fuel injection apparatus **31**, that is axially located between an upstream interface with the gas turbine engine **10**, and a downstream interface with the propelling nozzle **40**. Fuel is provided to the at least one fuel injection apparatus **31** by a reheat system fuel metering valve **32**.

(20) In some but not necessarily all embodiments, the fuel injection apparatus **31** may comprise at least one fuel distribution valve (not shown). The at least one fuel distribution valve may distribute fuel to different portions of the fuel injection apparatus **31** and/or, may vary the proportion of fuel that is provided to different portions of the fuel injection apparatus **31**. By so doing, fuel may be distributed to different portions of the combustion chamber **30**, within which the fuel injection apparatus is present.

(21) In other embodiments, the reheat system **50** may comprise a plurality of independent fuel injection apparatus, each fuel injection apparatus providing fuel to a different portion of the combustion chamber **30**. In these embodiments, fuel may be distributed to different portions of the combustion chamber **30** by independently controlling the fuel provided to each independent fuel injection apparatus.

(22) At least one ignitor **33** is located proximal to the at least one fuel injection apparatus **31**. The at least one ignitor **33** is configured to ignite fuel injected via the at least one fuel injection apparatus **31** into the jet pipe **30** of the reheat system **50**.

(23) At least one flame stabiliser **34**, such as a v-gutter, may be located within the combustion chamber, as is known in the art.

(24) The operation and thrust control of the reheat system **50** is now disclosed.

(25) It is known that a reheat system **50** augments the thrust produced by the gas turbine engine **10**, by burning fuel within the duct (jet pipe) **30**. The duct (jet pipe) **30** is therefore a combustion chamber **30**.

(26) Burning fuel in the jet pipe increases the temperature and reduces the density of the working fluid within the jet pipe. This reduction in density increases the velocity of the working fluid within the jet pipe, thereby increasing the momentum of the working fluid when discharged from the propelling nozzle **40**. This increase in momentum of the working fluid augments the thrust produced by the gas turbine engine **10**.

(27) The effective area of propelling nozzle **40** may be varied in response to combustion within the combustion chamber **30**, so as not to adversely impact operation of gas turbine engine **10**.

(28) Thus, by controlling the rate at which fuel is injected into the jet pipe **30**, it is possible to control the degree of temperature rise of the working fluid within the jet pipe **30**, thereby

controlling the degree of thrust augmentation. Many reheat systems therefore control reheat system thrust augmentation by controlling the position or orientation of a component of a fuel metering valve **32**, such as a metering orifice, to control the fuel flow rate provided to the reheat system **50**. The demanded position or orientation of the component of the fuel metering valve **32** may be determined by a fuel flow controller.

(29) Controlling the thrust augmentation of a reheat system **50** by controlling the position or orientation of a component of a fuel metering valve **32** has several disadvantages.

(30) For example, tolerancing differences between different fuel metering valves **32** made to the same design mean that different metering valves **32** may deliver different fuel flow rates for the same demanded fuel flow rate (defined by the demanded position or orientation of a component of a fuel metering valve **32**).

(31) Furthermore, the fuel flow rate delivered by a specific fuel metering valve **32** may change with use of that fuel metering valve **32**, due to wear or general degradation of the components within the fuel metering valve **32**.

(32) For example, the fuel flow rate delivered by the fuel metering valve **32** may decrease with use, due to degradation of seals within the fuel metering valve **32**. This degradation may increase internal leakages within the fuel metering valve, leading to a reduction in the delivered fuel flow for a given position or orientation of a component of the fuel metering valve **32** relative to the delivered fuel flow when the metering valve was assembled or (if appropriate), last overhauled.

(33) To ensure that, across of fleet of aircraft fitted with gas turbine engines and reheat systems, each engine and reheat system combination delivers the required thrust when both new and deteriorated, the relationship between demanded thrust and fuel metering valve **32** setting may be set to accommodate for the variability between fuel metering valves **32**, and deterioration of these fuel metering valves **32** due to use.

(34) This means that a portion of the fleet of reheat systems, fitted with higher flowing fuel metering valves, will deliver more fuel than demanded (overfuelling).

(35) Overfuelling has several potential disadvantages.

(36) For example, overfuelling may lead to a reduction in the operational range of an aircraft comprising the reheat system, as more fuel is supplied to the reheat system than required, increasing fuel consumption. This may also have a detrimental commercial impact for the manufacturer of the reheat system, as increased fuel consumption due to overfuelling may trigger penalty clauses in a commercial contract between the manufacturer and their customers, due to failure to meet a contractual operational range requirement, due to this increased fuel consumption.

(37) Overfuelling may also further elevate temperatures within the combustion chamber **30**, leading to a reduction in life of the combustion chamber or its components.

(38) Incorrect fuelling may also lead to other control issues, as is now explained with reference to FIG. **3**. This FIG. provides an example illustration of how the % augmented thrust produced by a heat system may vary with reheat system fuel air ratio (thrust to fuel:air ratio relationship **60**)

(39) FIG. **3** illustrates that a reheat system produces thrust over a range of fuel:air ratios, the range of fuel:air ratios having a lean extinction limit **61** and a rich extinction limit **62**.

(40) The lean extinction limit **61**, at which the minimum % augmented thrust increase occurs, may correlate to a lean combustion stability limit, below which stable operation of the reheat system is not possible due to a propensity for flame out, due to excess air and correspondingly low combustion temperatures.

(41) The rich extinction limit **62** corresponds to higher combustion temperatures than the lean extinction limit, but does not correspond to the maximum % thrust augmentation produced by the reheat system.

(42) This is because as the fuel:air ratio approaches the rich extinction limit **62**, thrust reaches a maximum **63***a*, before falling as fuel air ratio increases. This reduction in thrust with increasing fuel flow occurs as the fuel air ratio approaches the stoichiometric ratio, corresponding to the

amount of air necessary to completely combust the fuel, but leaving no oxygen remaining post combustion.

(43) In practical reheat systems, it is desirable to avoid operation at or close to the rich extinction limit **62**, yet for operational reasons (operating at high % augmented thrust in combat), it is desirable to be capable of operation at or close to the maximum **63***a*. This is because if the reheat system is operated at a fuel air ratio richer than the maximum **63***a*, additional fuel is injected into the combustion chamber for no additional increase in thrust. This worsens the operational range of an aircraft comprising the reheat system yet provides no additional benefit in aircraft performance.

(44) Practical reheat systems may therefore operate to an upper permissible operational limit **63**, representing an afterburner fuel:air ratio that is sufficiently high to provide a high % thrust augmentation, but is close to the maximum **63***a*.

(45) Although in FIG. **3**, the upper permissible operational limit **63** is shown to be at a lower (leaner) fuel:air ratio than the fuel:air ratio at which maximum thrust occurs **63***a*, in some embodiments, the upper permissible operational limit **63** and maximum **63***a* may be the same.

(46) From the previous discussion, it will be appreciated that variability between individual fuel metering valves **32**, and degradation of these fuel metering valves **32** with use, causes some metering valves to overfuel, particularly when new.

(47) Consequently, if these higher-flow fuel metering valves **32** are demanded to deliver a fuel flow corresponding to the reheat system fuel air ratio upper permissible operational limit **63**, these fuel metering valves would in fact overfuel, and deliver a fuel flow equivalent to a reheat system fuel air-ratio that is greater than the upper operational limit **63**. As illustrated in FIG. **3**, this could lead to a decrease in thrust, despite the increase in fuelling, if the actual fuel:air ratio exceeded the fuel:air ratio at which maximum **63***a* occurs.

(48) To ensure no individual reheat system fuel metering valve provides a maximum demanded fuel flow at a flight condition that exceeds the upper permissible operational limit **63** fuel:air ratio, it may be necessary to define an upper nominal fuel:air ratio **64**, defining the maximum nominal demanded fuel flow at any given flight condition. In fuel:air ratio terms, the upper nominal fuel:air ratio is smaller than the upper permissible operation limit **63**, by an offset **65**. The magnitude of the offset **65** incorporates an allowance made for variability between different fuel metering valves **32** and degradation of individual metering valves **32** with time or use.

(49) From FIG. **3**, it will be appreciated that a downside of this approach is that the maximum % augmented thrust obtained by operation to the upper nominal fuel:air ratio **64** is reduced relative to operation at the upper permissible operational limit **63**, and therefore, is lower than the maximum thrust that could be achieved at any given flight condition with perfect control (zero variability) of fuel air ratio. It will also be appreciated that the magnitude of this reduction is dependent on the magnitude of offset **65**, allowing for fuel metering valve variation and degradation.

(50) Reducing the magnitude of offset **65** is therefore desirable, as the magnitude of offset **65** may be correlated to a derate of the maximum thrust achievable by a reheat system. This is likely to be undesirable for a combat aircraft that comprises reheat system **50**, as the derate in maximum thrust may also reduce the combat effectiveness of the combat aircraft.

(51) An improved means of reheat system control is now disclosed.

(52) The improved means of control reduces the required magnitude of offset **65**. This is because the alternative means of control uses a static pressure-based control parameter that is more directly linked to the actual thrust produced by a reheat system, than the fuel metering valve based control means known in the art.

(53) The improved means of control additionally reduces the magnitude of offset **65** because, unlike traditional control means based upon a position or orientation of a flow metering component within fuel metering valve **32** (volumetric based control), the alternative means of control is not affected by the changes in density of the fuel provided to the airbreathing jet engine. This may be an advantage as different standards of fuel, each having a different nominal specified density, may

only be available for refuelling an aircraft comprising the airbreathing jet engine, at different geographical locations, from which the aircraft is operated.

(54) The improved means of control therefore prevents the reheat system from operation at a fuel air ratio that is richer than the fuel air ratio **63***a* at which maximum % thrust augmentation is obtained.

(55) It is known that as a working fluid passes through the jet pipe of a reheat system, as the working fluid has a non-zero viscosity, a reduction in static pressure occurs.

(56) Combustion of fuel within the jet pipe **30** causes a further drop in static pressure along the length of the jet pipe, the magnitude of the pressure drop related to the working fluid temperature rise and velocity increase due to combustion. As the velocity increase of the working fluid is in turn related to a momentum change in working fluid, and hence the % augmented thrust, it may now be appreciated that a change in static pressure along the length of the jet pipe **30** of the reheat system **50** may be correlated to the % augmented thrust produced by the reheat system **50**.

(57) A control method based on static pressure drop along the reheat jet pipe (combustion chamber) may therefore be used to control the thrust developed by the reheat system.

(58) The improved means of control requires an amended form of reheat system, an example of which is illustrated in FIG. **4**.

(59) The reheat system of FIG. **4** differs from that illustrated in FIG. **2** in that it additionally comprises at least one static pressure offtake **38** at a first axial location **35** and at least one static pressure offtake **38** at a second axial location **36**, different to the first axial location **35**. Static pressure at the at least one static pressure offtake **38** of the first axial location **35** and second axial location **36** is obtained by pressure measurements made with at least one pressure transducer **38**.

(60) The first axial location **35** is upstream of the second axial location **36**.

(61) The first axial location **35** may be upstream of the at least one fuel injection apparatus **31**. In other words, the first axial location **35** is at an axial location within the combustion chamber which is upstream of a portion of the combustion chamber in which combustion may take place.

(62) The first axial location **35** may be at or proximal to the interface between an exit of the gas turbine engine **10** and an inlet to the reheat system **50**. For example, the first axial location **35** may be at an exit plane of the bypass duct **22**.

(63) The second axial location **36** may be downstream of the at least one fuel injection apparatus **31**. In other words, the second axial location **36** is at an axial location within the combustion chamber which is downstream of a portion of the combustion chamber in which combustion may take place.

(64) The second axial location **36** may be proximal to the interface of the combustion chamber **30** and propelling nozzle **40**.

(65) In some but not necessarily all embodiments, the first axial location **35** may be a plane, such as a cross-sectional plane of the combustion chamber **30**, at a constant axial location along the length of the combustion chamber. Similarly, the second axial location **36** may be plane, such as a cross-sectional plane of the combustion chamber **30**, at a constant axial location along the length of the combustion chamber.

(66) Alternatively, in other embodiments, the first axial location **35** may be an inclined plane across the width of the combustion chamber and the second axial location **36** may also be an inclined plane across the width of the combustion chamber.

(67) Other arrangements of first axial location **35** and second axial location **36** may also be envisaged without departing from the scope of the invention.

(68) In some but not necessarily all embodiments, a plurality of static pressure measurements may be made at the first axial location **35** and/or the second axial location **36**.

(69) For example, in embodiments in which the first axial location **35** is a cross-sectional plane through the combustion chamber **30**, the at least one pressure offtake **38** at the first axial location may comprise a plurality of static pressure offtakes **38**. The plurality of pressure offtakes **38** may be

circumferentially disposed around a perimeter of the cross-sectional plane. In some but not necessarily all these embodiments, the plurality of pressure offtakes **38** at the first axial location may be fluidically connected to each other.

(70) Similarly, the at least one pressure offtake **38** at the second axial location may comprise a plurality of static pressure offtakes **38**. The plurality of pressure offtakes **38** may be circumferentially disposed around a perimeter of a cross-sectional plane. In some but not necessarily all these embodiments, the plurality of pressure offtakes **38** at the second axial location may be fluidically connected to each other.

(71) Fluidically connecting a plurality of pressure offtakes **38** at the same axial location May be desirable because the static pressure formed by the fluidic connection of the plurality of offtakes is an average of the static pressures local to each static pressure offtake of the plurality of static pressure offtakes **38**. This form of averaging is a fluidic average. Fluidically connecting a plurality of pressure offtakes **38** at the same axial location may also be desirable because it reduces the number of pressure transducers **39** required.

(72) The at least one static pressure offtake **38** at the first axial location **35** is configured to enable a measurement of the static pressure at the first axial location to be obtained.

(73) The at least one static pressure offtake at the second axial location is configured to enable a measurement of the static pressure at the second axial location to be obtained.

(74) Each static pressure offtake **38** at the first and second axial locations **35**, **36** comprises an aperture in the wall of the combustion chamber **30**, the aperture being substantially flush with a wall of the combustion chamber and substantially parallel with a direction of flow of the working fluid proximal to the hole. In this way, a pressure measured by the static pressure offtake does not comprise a dynamic component, as there is no "ram" effect to the measurement.

(75) Tubing **39** is mechanically connected at a first end to the wall of the combustion chamber **30** surrounding each aperture to form a seal with the surrounding wall of the combustion chamber **30** and is coupled at a second end to a pressure transducer **37**. The tubing **39** forms a fluidic connection between static pressure offtake **38** and pressure transducer **37**.

(76) The pressure transducer **37** may be remote from the combustion chamber **30** and aperture. This may be advantageous because the temperature of the pressure transducer may be substantially lower than the temperature of the wall of the combustion chamber **30**. This may be desirable because the local temperature of the wall of the combustion chamber **30** is high during operation of the reheat system **50** and may otherwise damage a transducer proximal to the wall or require that the transducer be cooled.

(77) Locating the pressure transducer **37** remote from the combustion chamber **30** via tubing may have an additional advantage in that the tubing **39** may provide attenuation of higher frequency static pressure oscillations that may otherwise be more distinctly measured by a pressure transducer closely coupled to the at least one static pressure offtake on the wall of the combustion chamber **30**.

(78) Various ways of measuring the static pressures at the first axial location **35** and the second axial location **36** are envisaged.

(79) For example, in some but not necessarily all examples, the static pressure at the first axial location **35** may be measured by a first pressure transducer **37**, and the static pressure at the second axial location **36** may be measured by a second pressure transducer **37** (as illustrated in FIG. **4**).

(80) Also in some but not necessarily all examples, the static pressure at the first axial location **35** and the static pressure at the second axial location **36** may be measured by a common pressure transducer. The common pressure transducer is fluidically couplable to at least one static pressure offtake at the first axial location **35** and is fluidically couplable to at least one static pressure offtake at the second axial location. A switching apparatus is between the common pressure transducer and the at least one static pressure offtake at the first axial location **35** and the at least one static pressure offtake at the second axial location **36**. The switching apparatus comprises a first configuration and a second configuration, switching between the first and second configurations

being controlled by a controller.

(81) In embodiments comprising a common pressure transducer, in the first configuration, the aperture at the first axial location is fluidically coupled to the common transducer, but the aperture at the second axial location is not fluidically coupled to the common pressure transducer. Similarly, in embodiments comprising a common pressure transducer, in the second configuration, the aperture at the second axial location is fluidically coupled to the common transducer, but the aperture at the first axial location is not fluidically coupled to the common pressure transducer.

(82) Alternatively, in some but not necessarily all embodiments, a first pressure transducer **37** may be connected to one of the at least one static pressure offtake at the first axial location **35** and the at least one static pressure offtake at the second axial location **36**. In these embodiments, a differential pressure transducer may be fluidically connected to the at least one static pressure offtake at the first axial location **35** at a first input to the differential pressure transducer and to at least one static pressure offtake at the second axial location **36** at a second input to the differential pressure transducer. In these embodiments, the differential pressure transducer measures a static pressure difference between the first axial location **35** and the second axial location **36**. In this arrangement, the static pressure at one of the first axial location **35** and the second axial location **36** is obtained directly by measurement, while the static pressure at the other of the first axial location **35** and the second axial location **36** may be obtained by addition or subtraction of the measured pressure difference between the first axial location **35** and the second axial location **36**, as appropriate. Fluidically connecting a differential pressure transducer between the at least one aperture at the first axial location **35** and the at least one aperture at the second axial location **36** may result in improved accuracy of the determined upstream and downstream static pressures, as for example, it may permit a pressure transducer with a smaller full-scale pressure range to be used, than would otherwise be possible.

(83) In alternative embodiments, static pressure is not measured remotely, but an individual pressure transducer is mounted in each static pressure offtake **38** such that a measurement surface of the static pressure transducer is flush mounted with the wall of the combustion chamber **30**. As previously disclosed, in these embodiments, cooling to the pressure transducer **37** may be additionally provided.

(84) Measuring static pressure directly has the advantage that measurement lag is reduced relative to remote static pressure measurement. This may be advantageous when the fuel flow demanded is changing rapidly, as transient overshooting of delivered fuel flow beyond the thrust maximum **63**a, towards the rich extinction limit **62** may be minimised or prevented.

(85) In embodiments in which a plurality of static pressure offtake measurements are made, it will be appreciated that averaging of related static pressure offtakes **38** (for example, those at a common location) may alternatively be made numerically, rather than by fluidically coupling the offtakes **38** together.

(86) Pressure fluctuations may be present within the combustion chamber **30**. The pressure fluctuations may be generated by combustion and/or modulation of the fuel metering valve **32**. In some, but not necessarily all embodiments, frequency-based filtering (for example, smoothing) may be applied to the output of pressure transducer **37**.

(87) In embodiments in which filtering is applied to the output signals provided by the static pressure transducer, the filtering may be provided by a low-pass filter, a high-pass filter or a combination of both. The filtering may be provided to remove oscillations from the output signals caused by pressure oscillations present within the combustion chamber **30**. This may improve the stability of the control method.

(88) The filtering provided may be dependent upon the acoustic properties of the combustion chamber **30** and the properties of the working fluid within the combustion chamber. For example, the filtering may be provided to attenuate at least one frequency generated by combustion of the working fluid and fuel within the combustion chamber **30**.

(89) The at least one frequency may be generated based at least in part on an axial length of the combustion chamber, a characteristic cross-sectional dimension of the combustion chamber, such as a diameter of the combustion chamber, and/or a ratio of the length to the characteristic diameter.

(90) The at least one frequency may be based upon a temperature, such as an average temperature of the working fluid within the combustion chamber.

(91) The at least one frequency may be based upon a residence time of the working fluid within the combustion chamber.

(92) The at least one frequency may be based upon a blade-passing frequency of the high-pressure turbine **17**, intermediate pressure turbine **18** and/or a low-pressure turbine **19**.

(93) The at least one frequency may be based upon a frequency of fluctuation in the pressure of fuel provided to fuel injection apparatus **31**, as may be generated by a fuel pump providing fuel to the fuel injection apparatus **31**.

(94) From the above disclosure, it will be appreciated that the thrust augmentation produced by a reheat system may therefore be controlled by varying the fuel supplied by the fuel metering valve **32** such that a measured static pressure between the first axial location **35** and the second axial location **36** is substantially equal to a demanded static pressure drop between the first axial location **35** and the second axial location **36**, wherein the demanded pressure drop correlates to the desired thrust augmentation.

(95) It may also be appreciated though that the above method of control relates to changes in static pressure, and changes in thrust.

(96) As an aircraft equipped with a reheat system is operated over a range of flight conditions (altitude, forward speed), it may also be appreciated that the static pressure at any given location within the combustion chamber **30** of the reheat system **50** may be strongly influenced by the aircraft flight condition.

(97) For example, the static pressure at any given location within the combustion chamber **30** may be substantially higher when the aircraft is flown at high speed and low altitude, relative to when the aircraft is flown at a low speed and high altitude. This means that the pressure drop along the combustion chamber **30** of the reheat system **50** may also be strongly influenced by aircraft flight condition. Similarly, the maximum thrust developed by the engine **10** will be higher at low altitude to that achievable at higher altitude, as is known in the art.

(98) It is therefore desirable for a static pressure based reheat system thrust control method to take account of aircraft flight condition. The desired thrust augmentation around the aircraft flight envelope may be determined by the customer of the aircraft and may be a function of aircraft altitude and aircraft speed (for example, Mach number).

(99) The desired thrust augmentation at a flight condition correlates to the demanded pressure drop between the first axial location **35** and the second axial location **36**.

(100) Therefore, by obtaining a demanded power setting (the desired thrust augmentation) from either an aircraft control system or input from a pilot of the aircraft, and by accounting for the aircraft altitude and aircraft speed, the demanded pressure drop between the first axial location **35** and the second axial location **36** may be obtained.

(101) It may be appreciated that a variety of alternative approaches may be used to convert the input conditions (demanded thrust augmentation, aircraft altitude and aircraft speed) into the output condition (demanded pressure drop between the first axial location **35** and the second axial location **36**).

(102) By way of example, one approach is illustrated in FIG. **5** which provides an example of a % augmented thrust:pressure ratio relationship **70**, in which:

(103) % Thrust is defined as a % increase in thrust generated by the reheat system relative to a datum. The datum may for example, correspond to a maximum thrust that may be provided by the gas turbine engine **10** at a current aircraft flight condition; and

(104) Pressure ratio is defined as the static pressure at the first axial location **35**, divided by the

static pressure at the second axial location **36**.

(105) Relationship **70** may be used for thrust control, as is subsequently disclosed.

(106) Relationship **70** may be derived by numerical analysis or test, or a combination of both.

(107) The ratio of static pressure at the first axial location **35** to static pressure at the second axial location **36** may be in the range of 1.1 to 1.55.

(108) FIG. **5** additionally comprises an arrow representing a direction of increasing fuel:air ratio **71**. The corresponding range of fuel:air ratios of relationship **70** ranges from a low fuel air:ratio (at or close to the lean stability limit of the combustion chamber), to the fuel air ratio corresponding to the maximum of the thrust:fuel air ratio relationship illustrated in FIG. **3**.

(109) In the example relationship **70** illustrated in FIG. **5**, the relationship does not comprise data for fuel air ratios richer than the maximum **63***a* illustrated in FIG. **3**, as there is no operational benefit in operating the reheat system at fuel air ratios richer than the maximum.

(110) The use of the above relationship for thrust control is now disclosed, in relation to the method of FIG. **6**.

(111) At step **601**, a demanded thrust (power setting) is obtained. The demanded power setting may be derived from an output from a throttle, actuatable by a pilot of an aircraft or by a controller of an aircraft. In some but not necessarily all embodiments, a common throttle may be used to control both the demanded thrust of the gas turbine engine **10**, and the demanded augmented thrust of the reheat system. In these embodiments, at lower power settings, only the gas turbine engine **10** develops thrust, with the reheat system inactive. At higher power settings, the reheat system is additionally activated (for example, by providing fuel to the at least one fuel injection apparatus & energising ignitor **33**).

(112) At step **602**, a value of static pressure at a first axial location **35** within the combustion chamber **30** is obtained, by measurement of the static pressure at the least one static pressure offtake **38** at the first axial location **35**.

(113) At **603**, a value of static pressure at a second axial location **36** within the combustion chamber **30** is obtained, by measurement of the static pressure at the least one static pressure offtake **38** at the second axial location **36**.

(114) The value of static pressure at the first axial location **35** and the second axial location **36** may be an average measured static pressure at the respective first and/or second axial location.

(115) The value of the static pressure at the first axial location **35** may be obtained by fluidic averaging, achieved by fluidically connecting static pressure offtakes at the first axial location **35**. The value of the static pressure at the first axial location **35** may alternatively be obtained by numerical averaging of a plurality of static pressure measurements made independently at a plurality of static pressure offtakes **38** at the first axial location **35**. At least one of frequency-based filtering and/or smoothing may be applied to the output signal produced by a pressure transducer, to obtain the value of the static pressure at the first axial location **35**.

(116) The value of the static pressure at the second axial location **36** may be obtained by fluidic averaging, achieved by fluidically connecting static pressure offtakes at the second axial location **36**. The value of the static pressure at the second axial location **36** may be obtained by obtained by numerical averaging of a plurality of static pressure measurements made independently at a plurality of static pressure offtakes **38** at the second axial location **36**. At least one of frequency-based filtering and/or smoothing may be applied to the output signal produced by a pressure transducer, to obtain the value of the static pressure at the second axial location **36**.

(117) At step **604**, a ratio of the value of static pressure at the first axial location **35** to the value of static pressure at the second axial location **36** is obtained. This corresponds to a non-dimensionalised thrust developed by combustion within the combustion chamber. This ratio of static pressure values is then compared to a demanded pressure ratio value. The demanded pressure ratio value is obtained from a known pressure ratio to thrust relationship **70**, by inputting the demanded thrust, obtained at step **601**, into the relationship **70**.

(118) Relationship **70** may be in the form of a look-up table or an equation.

(119) Closed loop control is then used to vary the actual, measurement-derived static pressure ratio to be substantially equal to the demanded pressure ratio. From FIGS. **3** and **5** and the above disclosure, it is evident that the actual, measurement-derived static pressure ratio may be varied by varying the fuel flow rate delivered to the fuel injection apparatus **31** by the fuel metering valve **32** —i.e., closed loop pressure ratio based control is achieved by changing a demand in a position or orientation of a component, such as a metering orifice, within the fuel metering valve **32**, such that the actual static pressure ratio, obtained from pressure measurements at the first and second axial locations, is substantially equal to the demanded static pressure ratio, obtained from the pressure ratio to thrust relationship **70**.

(120) Optionally, several derivatives to step **604** are also envisaged.

(121) For example, although the pressure ratio to thrust relationship **70** illustrated in FIG. **5** is non-dimensional, and may thus compensate for variations in flight condition (altitude, speed) of an aircraft utilising the above control method, it may be appreciated that at certain flight conditions, it may be desirable to apply a modifier, such as a multiplier to the non-dimensional pressure ratio to thrust relationship **70** to determine demanded pressure ratio at a specific flight condition. This is because the actual capability of the reheat-system and/or gas turbine engine may be limited at specific flight conditions, due to another limit, such as a rotational speed of a shaft of the gas turbine engine **10**, being reached. The multiplier may be determined from a look-up table or multi-dimensional equation, having parameters related to the aircraft flight condition as input, and the multiplier as output.

(122) Parameters relating to the aircraft flight condition may comprise aircraft altitude, aircraft velocity, ambient atmospheric static pressure and/or ambient static temperature.

(123) The demanded pressure ratio may also be further modified based upon a configuration of the airbreathing jet engine. For example, the demanded pressure ratio may also be further modified based upon a demanded cross-sectional area of variable area propelling nozzle **40** (if present) or a demanded jet discharge angle from a thrust-vectoring propelling nozzle **40** (if present).

(124) It may be appreciated that changes in the discharge angle of a jet relating to principal axis **11** due to actuation of the thrust-vectoring nozzle **40** may change the static pressure distribution within the jet-pipe at a constant thrust.

(125) As the fuel flow controller is configured to control fuel flow rate based on a pressure ratio within the combustion chamber **30**, varying the discharge angle of the jet, may therefore undesirably change the thrust developed by the reheat system, for no change in demanded power setting, as the fuel flow controller compensates for the change in static pressure distribution caused by actuation of the thrust-vectoring nozzle.

(126) Step **604** may therefore additionally comprise the determination and application of an additional modifier to the demanded pressure ratio, the additional multiplier compensating for the demanded discharge angle of the jet caused by thrust vectoring. The additional multiplier may be determined from a look-up table, array or multi-dimensional equation, having the demanded thrust vectoring nozzle angle as an input and the additional multiplier as output.

(127) Similarly, it may also be appreciated that changing the cross-sectional area of a variable area propelling nozzle **40** at a demanded thrust may also change the magnitude of static pressures within different portions of the combustion chamber at a constant thrust. In some embodiments, the demanded pressure ratio may therefore be further amended based upon the configuration (demanded cross-sectional area) of the variable area propelling nozzle.

(128) The above thrust control method of steps **601** to **604** may additionally be used to control fuel distribution within the reheat system by controlling at least one fuel distribution valve. As previously disclosed, in some embodiments, the airbreathing jet engine **50** may comprise at least one fuel distribution valve, the at least one fuel distribution valve distributing fuel to different portions of the at least one fuel injection apparatus **31**. The different portions of the at least one fuel

injection apparatus **31** may injection fuel at different locations within the combustion chamber **30**.

(129) A notable difference between the combustion chamber **30** of a reheat system **50** (jet pipe) and the combustion equipment **16** of a gas turbine engine **10**, is that although the combustion equipment **16** of the gas turbine engine **10** is configured to combust all or substantially all of the fuel injected into it before expansion of the resulting combustion products in downstream turbines **17**, **18**, **19**, the combustion efficiency of the combustion chamber **30** of the reheat system **50** may be substantially less than 100%. This means that uncombusted fuel, in the form of vapour may be present in the propelling jet as it leaves propelling nozzle **40**.

(130) It is desirable to increase the combustion efficiency of the reheat system, as this may have an operational benefit of increasing the operational range of an aircraft equipped with the reheat system **50**.

(131) The combustion efficiency of the reheat system **50** may be altered by altering the distribution of fuel injected by different portions of the at least one fuel injection apparatus **31**, by actuating the at least one fuel distribution valve.

(132) In a derivative of the above method, a configuration of the at least one fuel distribution valve is changed in response to the ratio of the value of static pressure obtained at the first axial location **35** to the value of static pressure at the second axial location **36**.

(133) The configuration of the at least one fuel distribution valve may for example, be configured by the fuel flow controller to be changed such that the ratio of the value of static pressure at the first axial location **35** to the value of static pressure at the second axial location **36** is a maximum for a fixed position or orientation of a fuel-metering component of the at least one fuel metering valve **32**. This is because the maximum corresponds to a peak combustion efficiency for variations in the fuel distribution valve setting at the demanded thrust setting, and consequently, a minimum in fuel consumption at this thrust setting.

(134) Alternatively, in embodiments in which the reheat system **50** comprise a plurality of independent fuel injection apparatus, in which each fuel injection apparatus provides fuel to a different portion of the combustion chamber **30**, fuel may be distributed to different portions of the combustion chamber **30** by independently controlling the fuel provided by each independent fuel injection apparatus.

(135) In these embodiments, the proportion of fuel provided to each independent fuel injection apparatus may be varied such that the ratio of the value of static pressure at the first axial location **35** to the value of static pressure at the second axial location **36** is a maximum for a fixed total fuel flow delivered, the total fuel flow delivered being the sum of the fuel delivered by each independent fuel injection apparatus.

(136) In some but not necessarily all embodiments, the above method of control may be used as a sole method of thrust control for the airbreathing jet engine.

(137) In other embodiments, the above method of control may be used as one of several methods of control (control loops), running simultaneously within a fuel flow controller of the airbreathing jet engine.

(138) For example, the above method of control may run in parallel with a conventional fuel metering valve position demand based control system. In these embodiments, selection of which method of control to use may be based upon a comparison of the fuel metering valve position demanded. For example, the selected control loop may be determined based on a "minimum wins" criterion, in which the control loop, of a plurality of control loops running simultaneously, which outputs the lowest demanded fuel metering valve position is selected. In these embodiments, the above control loop functions as a fuel flow rate limiter, thereby preventing overfuelling at demanded % thrust augmentations.

(139) Other uses of the static pressure offtake measurements are also envisaged.

(140) For example, a drop in static pressure at the second axial location relative to the first axial location, for no substantial change in aircraft flight condition (altitude, speed) following provision

of fuel by the at least one fuel Injection apparatus **31**, and energising an ignitor **33**, may indicate that ignition of fuel has occurred in the combustion chamber.

(141) Similarly, a rise in static pressure at the second axial location relative to the first axial location, for no substantial change in aircraft flight condition may indicate that a flameout event has occurred in the combustion chamber.

(142) Thus, by comparing the static pressure measured at the second axial location to the static pressure measured at the first axial location, it is possible to determine the current state of the combustion chamber (lit, unlit, ignition, flameout).

(143) Although the above method of thrust control has been disclosed in relation to a reheat-system **50**, it will be appreciated that the method is also applicable to other forms of airbreathing jet engine in which a measured pressure drop or pressure ratio between two axial locations may be correlated to a thrust develop by the air breathing engine. For example, the method may also be applied for thrust control of a ramjet, or other forms of athodyd.

(144) It will be understood that the invention is not limited to the examples above described and various modifications and improvements can be made without departing from the concepts described herein. Except where mutually exclusive, any of the features may be employed separately or in combination with any other features and the disclosure extends to and includes all combinations and sub-combinations of one or more features described herein.

Claims

1. A method of thrust control for an airbreathing jet engine that has a combustion chamber, the method comprising: obtaining a demanded thrust setting; obtaining a value of static pressure at a first axial location of the combustion chamber of the airbreathing jet engine; obtaining a value of static pressure at a second axial location of the combustion chamber of the airbreathing jet engine, wherein the second axial location is downstream of the first axial location; obtaining a ratio of the value of static pressure at the first axial location of the combustion chamber to the value of static pressure at the second axial location of the combustion chamber; and controlling a fuel flow rate of the airbreathing engine based at least in part on the demanded thrust setting and the ratio of the value of the static pressure at the first axial location to the value of the static pressure at the second axial location.

2. The method of claim 1, wherein the first axial location is upstream of a fuel injection apparatus within the combustion chamber, and the second axial location is downstream of the fuel injection apparatus of the combustion chamber.

3. The method of claim 1, wherein the value of static pressure at the first axial location is obtained by measuring a static pressure at at least one static pressure offtake in a wall of the combustion chamber at the first axial location; and the value of static pressure at the second axial location is obtained by measuring a static pressure at at least one static pressure offtake in a wall of the combustion chamber at the second axial location.

4. The method of claim 3, wherein the value of static pressure at the first axial location is an average of the static pressure measured at the at least one static pressure offtake at the first axial location; and the value of static pressure at the second axial location is an average of the static pressure measured at the at least one static pressure offtake at the second axial location.

5. The method of claim 4, wherein the average of static pressure measured at the first axial location is a fluidic average and the average of static pressure measured at the second axial location is a fluidic average.

6. The method of claim 4, wherein the average of static pressure measured at the first axial location is a numerical average; and the average of static pressure measured at the second axial location is a numerical average.

7. The method of claim 1, wherein controlling fuel flow rate based at least in part on the demanded

thrust setting and a ratio of the value of the static pressure at the first axial location to the value of the static pressure at the second axial location comprises: referencing a look-up table or equation having the demanded thrust setting as an input to determine a demanded ratio of the value of the static pressure at the first axial location to the value of the static pressure at the second axial location from an output of the look-up table; and varying a fuel flow rate to the airbreathing jet engine such that the ratio of the value of the static pressure at the first axial location of the combustion chamber to the value of static pressure at the second axial location of the combustion chamber is substantially equal to the demanded ratio of the value of the static pressure at the first axial location to the value of the static pressure at the second axial location.

8. The method of claim 7, wherein the demanded ratio of the value of the static pressure at the first axial location to the value of the static pressure at the second axial location is additionally based upon a flight condition of an aircraft comprising the airbreathing jet engine.

9. The method of claim 8, wherein the flight condition of the airbreathing engine comprises at least one of a velocity of the airbreathing jet engine, a static temperature of an atmosphere through which the airbreathing jet engine is travelling, a static pressure of the atmosphere through which the airbreathing jet engine is travelling.

10. The method of claim 7, wherein the demanded ratio of the value of the static pressure at the first axial location to the value of the static pressure at the second axial location is additionally based upon a demanded discharge angle of a jet through a thrust-vectoring propelling nozzle.

11. The method of claim 7, wherein the demanded ratio of the value of the static pressure at the first axial location to the value of the static pressure at the second axial location is additionally based upon a demanded cross-sectional area of a variable area propelling nozzle.

12. The method of claim 11, wherein the demanded ratio of the value of the static pressure at the first axial location to the value of the static pressure at the second axial location is additionally based upon a demanded discharge angle of a jet through a thrust-vectoring propelling nozzle.

13. The method of claim 1, wherein controlling the fuel flow rate of the airbreathing jet engine additionally comprises varying a configuration of at least one fuel distribution valve to vary the distribution of fuel to different portions of the fuel injection apparatus such that the fuel flow rate provided to the airbreathing jet engine is minimised for the demanded thrust setting.

14. An airbreathing jet engine comprising: a combustion chamber; at least one fuel injection apparatus configured to inject fuel into the combustion chamber; at least one static pressure offtake at a first axial location within the combustion chamber; at least one static pressure offtake at a second axial location within the combustion chamber; at least one pressure transducer, the at least one pressure transducer configured to obtain a value of static pressure at the first axial location, and to obtain a value of static pressure at the second axial location; a fuel metering valve; and a fuel flow controller; wherein the fuel flow controller controls a flow rate of fuel provided by the fuel metering valve to the combustion chamber via the at least one fuel injection apparatus based at least in part on a demanded thrust setting for the airbreathing jet engine and a ratio of a value of the static pressure at the first axial location to a value of the static pressure at the second axial location.

15. The airbreathing jet engine of claim 14, wherein the fuel flow controller controls the flow rate of fuel additionally based upon a flight condition of an aircraft comprising the airbreathing jet engine.

16. The airbreathing jet engine of claim 14, wherein the fuel flow controller controls the flow rate of fuel additionally based upon a configuration of the airbreathing jet engine.

17. The airbreathing jet engine of claim 16, wherein the fuel flow controller controls the flow rate of fuel additionally based upon a flight condition of an aircraft comprising the airbreathing jet engine.

18. The airbreathing jet engine of claim 14, wherein the airbreathing jet engine is an afterburner.

19. The airbreathing jet engine of claim 14, wherein the airbreathing jet engine is a ramjet.