(12) UK Patent Application (19) GB (11) 2588955 (13) A

(43) Date of A Publication 19.05.2021

(21) Application No: 1916667.7

(22) Date of Filing: 15.11.2019

(71) Applicant(s):
Rolls-Royce plc
Kings Place, 90 York Way, London, N1 9FX, United Kingdom

(72) Inventor(s):
Bharat Lad

(74) Agent and/or Address for Service:
Rolls-Royce plc
IP Department (SinA-48), PO Box 31, DERBY, Derbyshire, DE24 8BJ, United Kingdom

(51) INT CL:
F01D 5/14 (2006.01) ***F01D 5/16*** (2006.01)
F01D 5/18 (2006.01) ***F01D 25/06*** (2006.01)

(56) Documents Cited:
GB 2481822 A **GB 0736835 A**
EP 1947294 A2 **EP 1785589 A1**
EP 1118747 A2 **WO 2015/065659 A1**

(58) Field of Search:
INT CL **F01D**
Other: **WPI, EPODOC**

(54) Title of the Invention: **A turbomachine blade**
Abstract Title: **A turbomachine blade with a passage from a pressure surface to a suction surface**

(57) A turbomachine blade comprises an aerofoil portion 64 extending between a tip 66 and a root 62, and having a pressure surface (78, fig 6a) and a suction surface 80. At least one passageway (76, fig 7) extends through the aerofoil portion from an inlet 72 in the pressure surface to an outlet 74 in the suction surface. The outlet is located in a section of the aerofoil portion extending between the tip and 50% of the length of the aerofoil portion from the tip. The outlet may be between the tip and 25% of the length of the aerofoil portion from the tip. The inlet and outlet may be located at different points along the chord length of the aerofoil portion. The passageway may extend in a direction from a leading edge 68 to a trailing edge 70. A plurality of passageways may extend through the aerofoil portion, and may have a common inlet and a plurality of outlets. The blade may be a fan blade. A gas turbine engine has a rotor comprising a plurality of the blades.

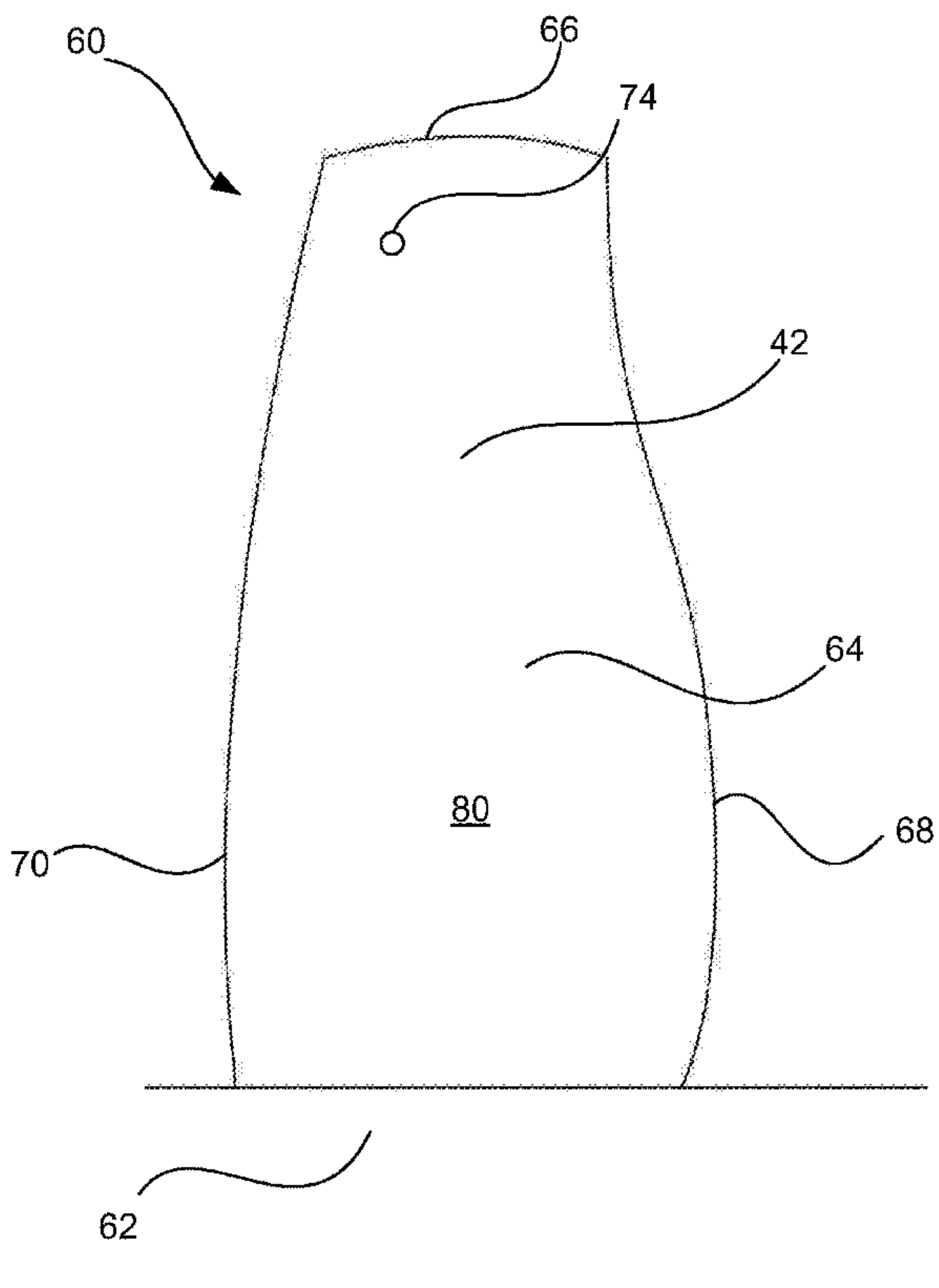

Figure 6b

GB 2588955 A

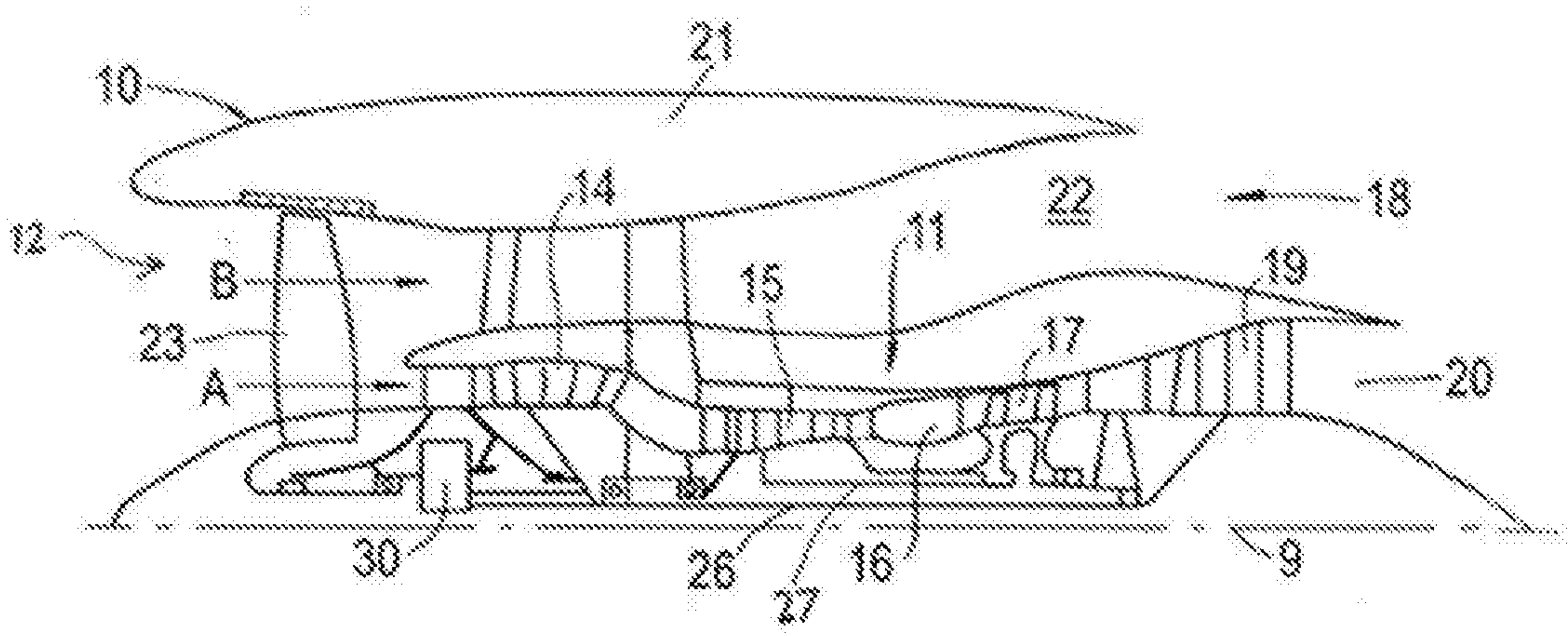

Figure 1

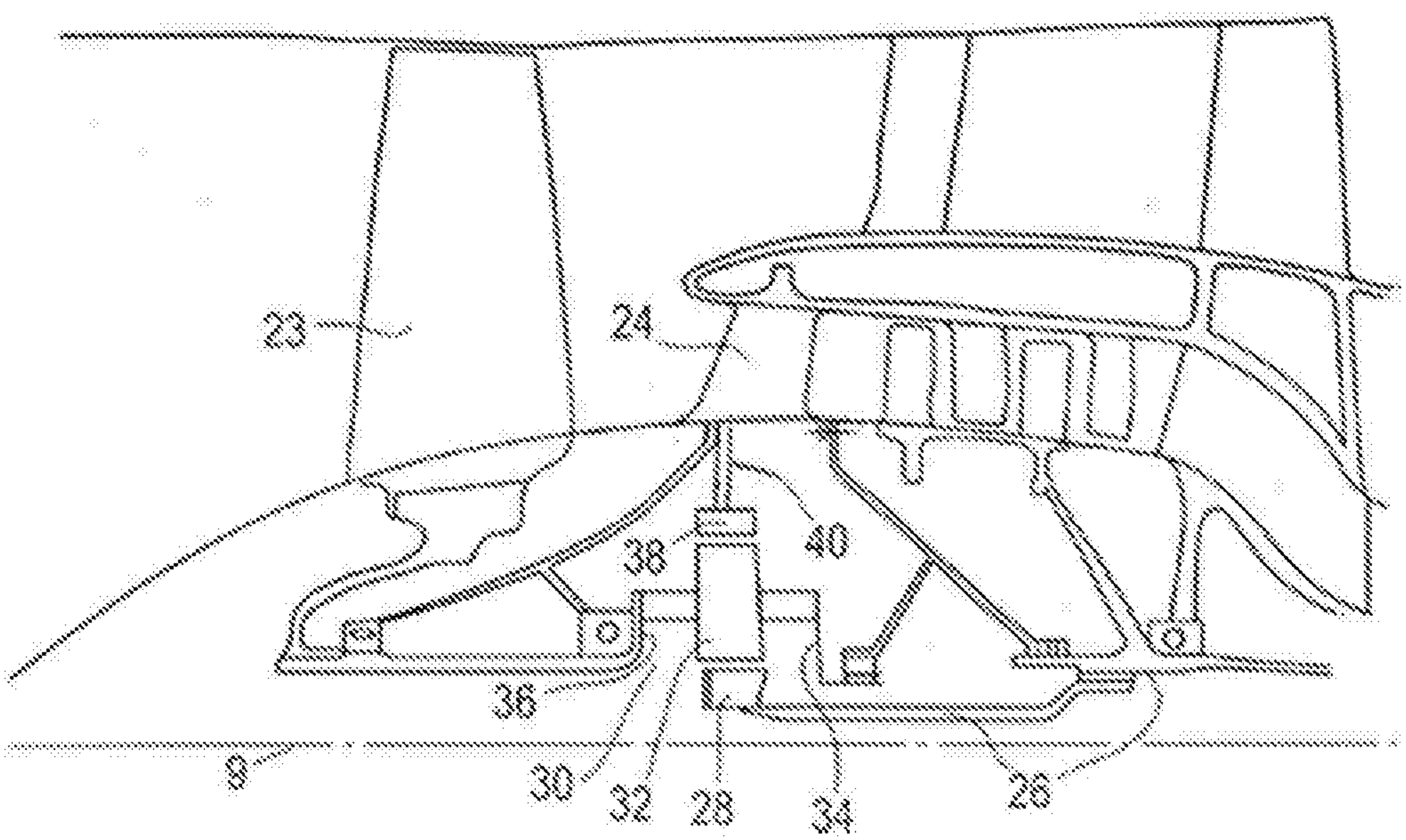

Figure 2

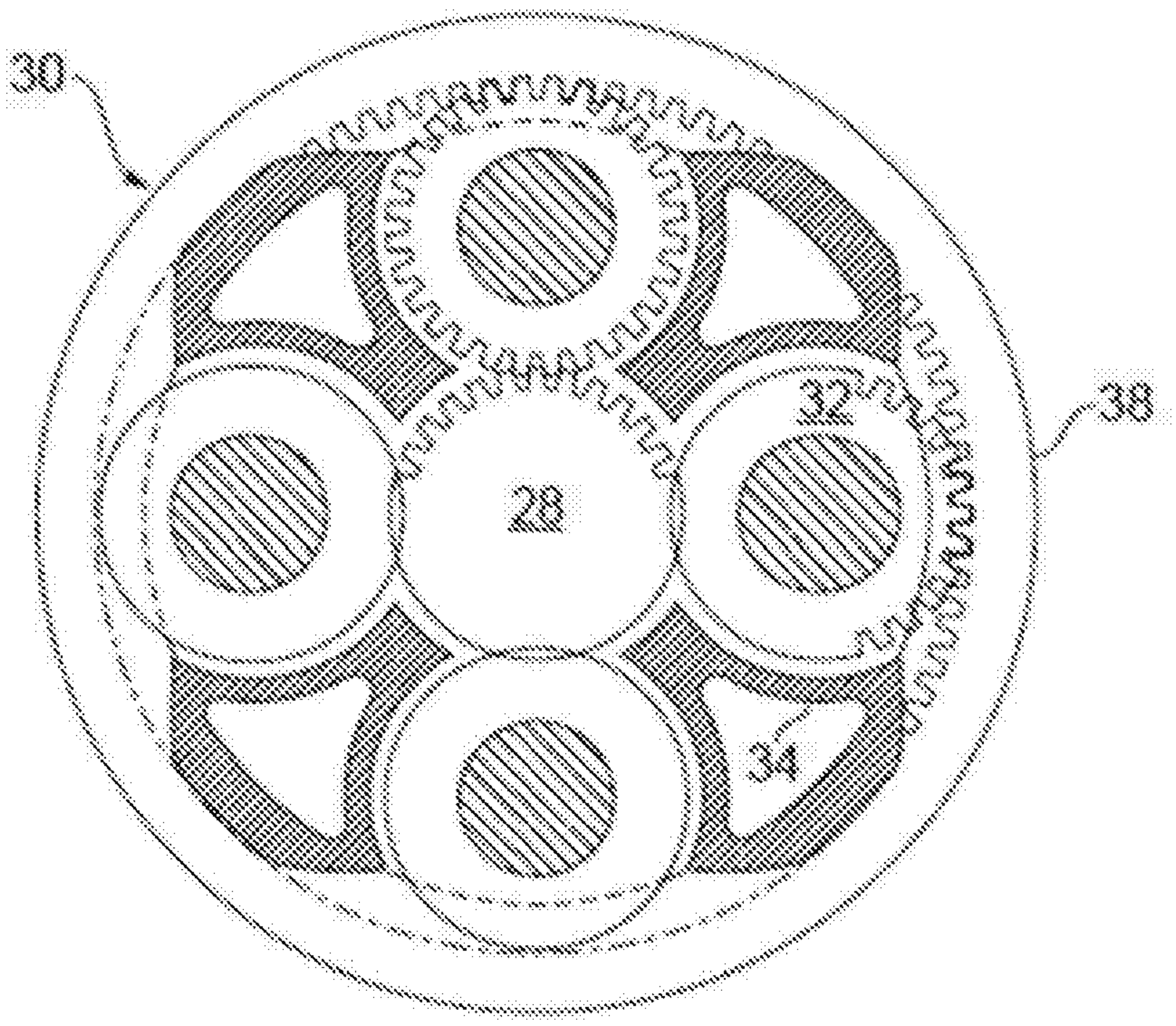

Figure 3

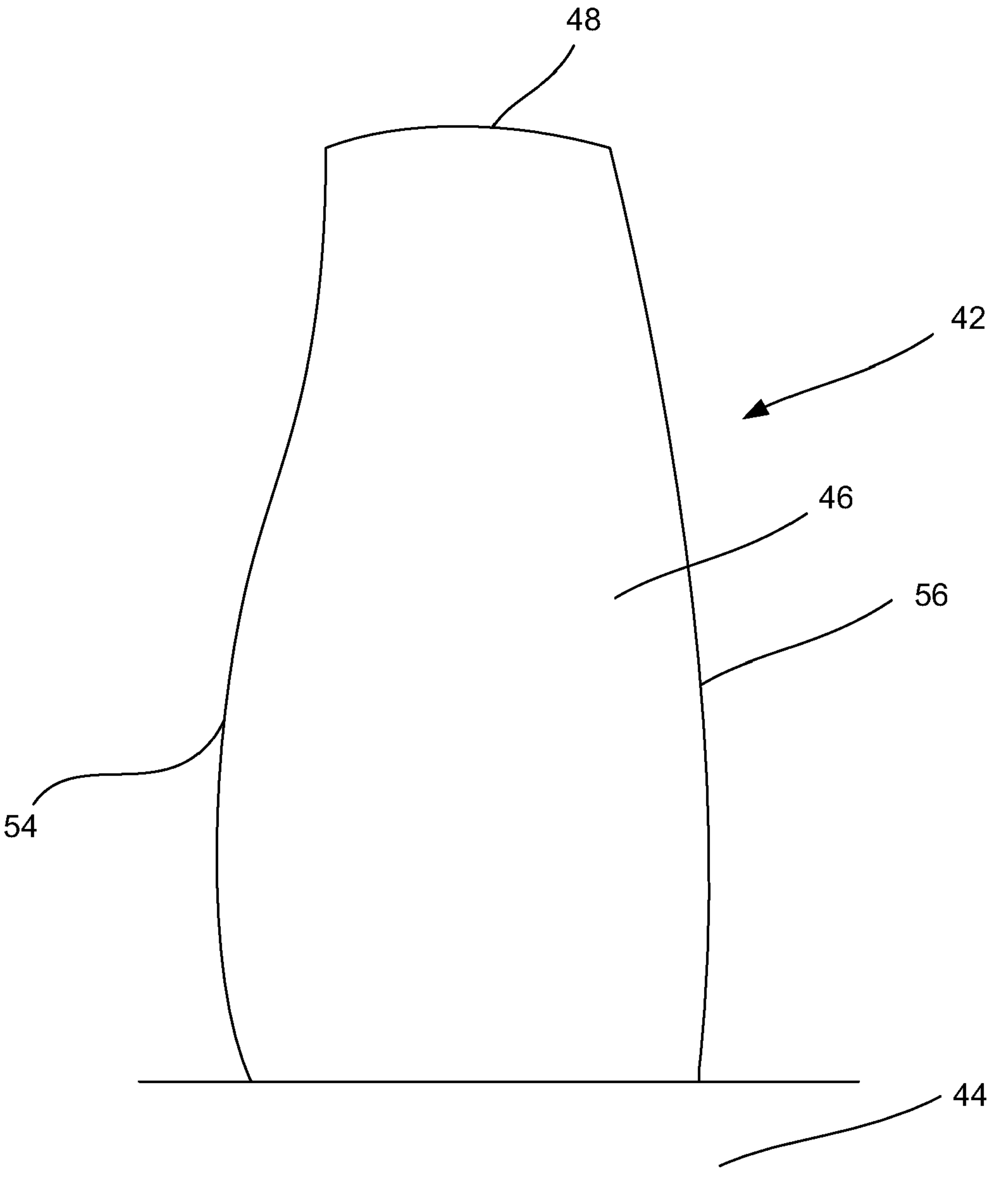

Figure 4

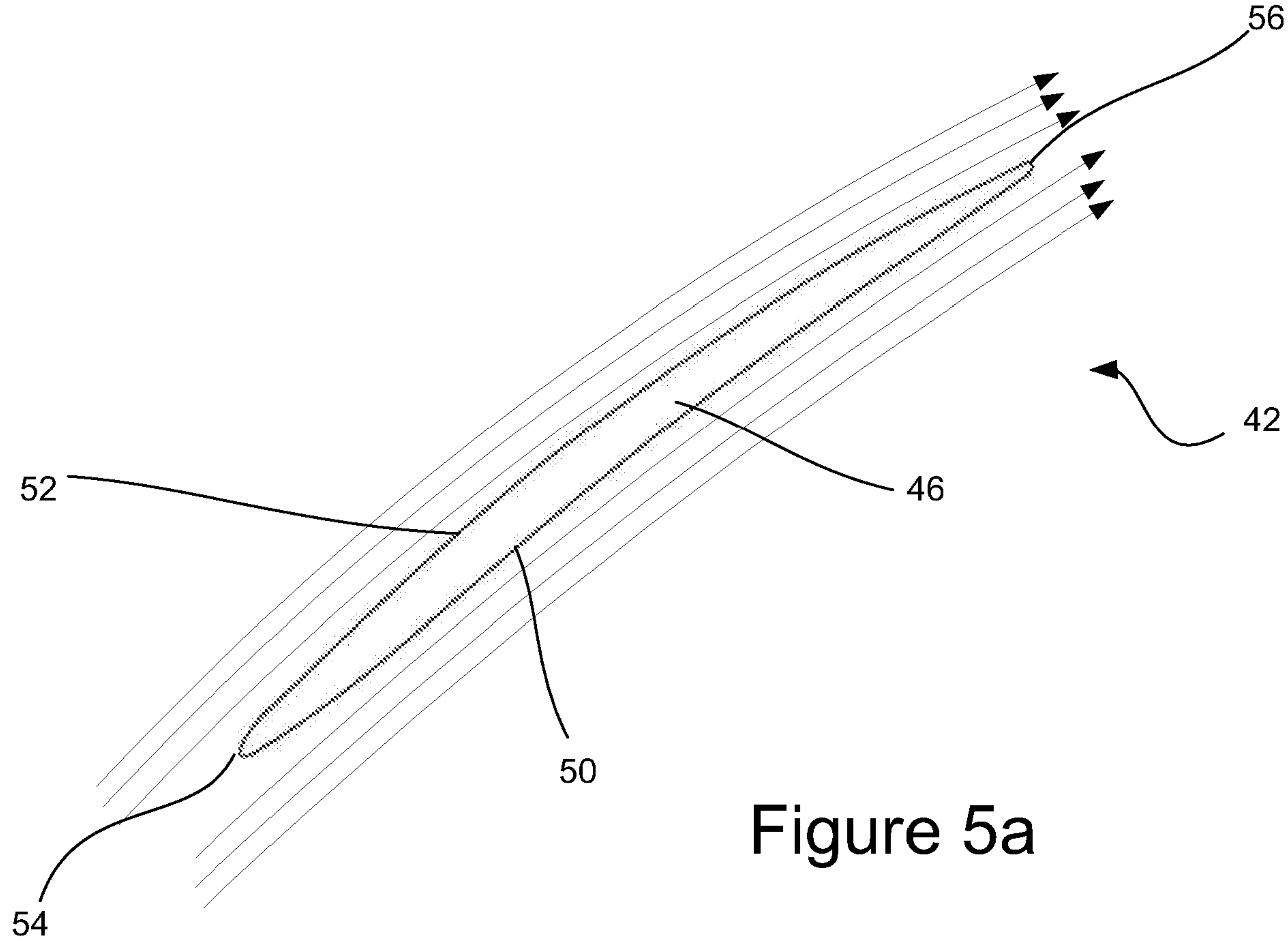

Figure 5a

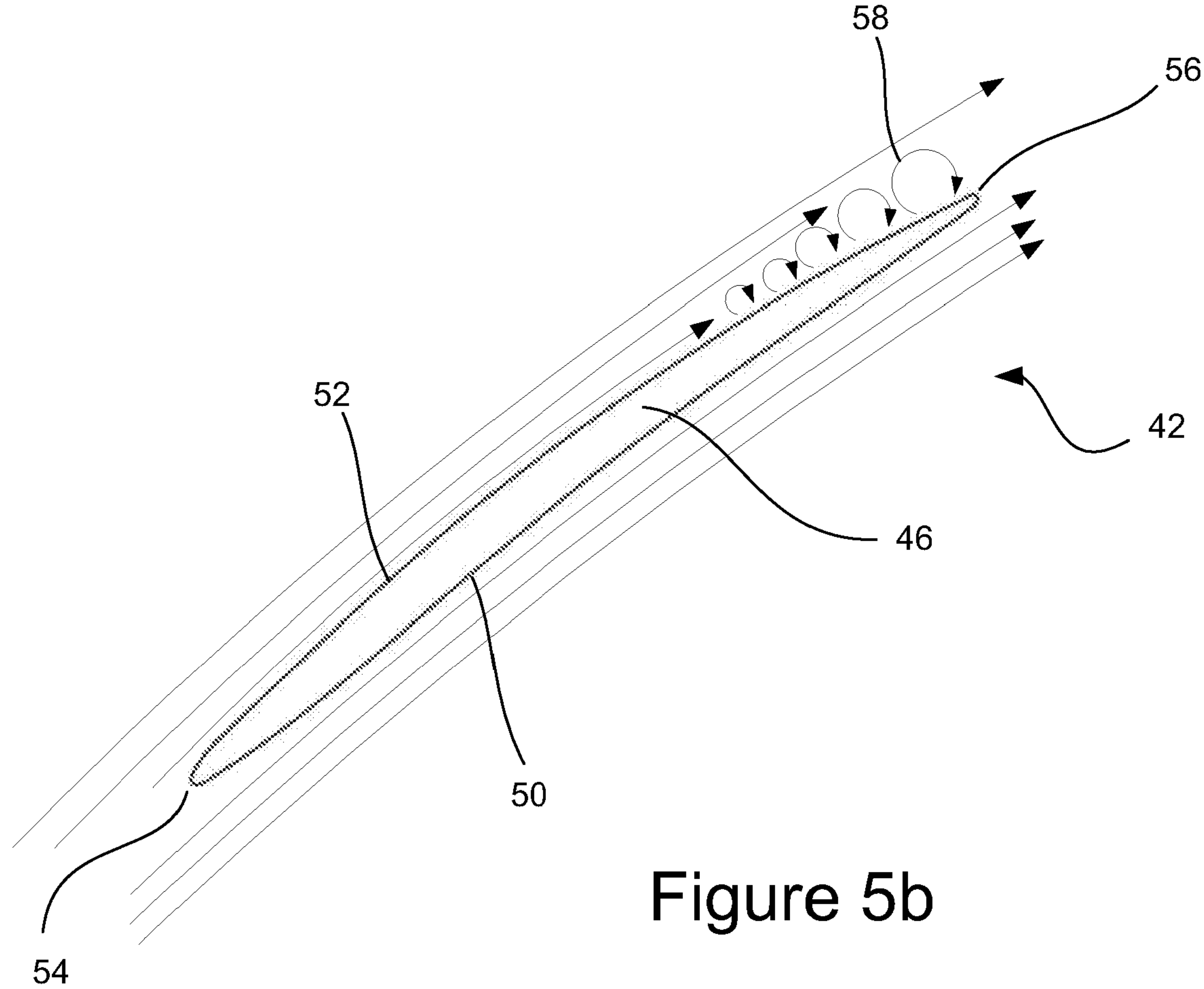

Figure 5b

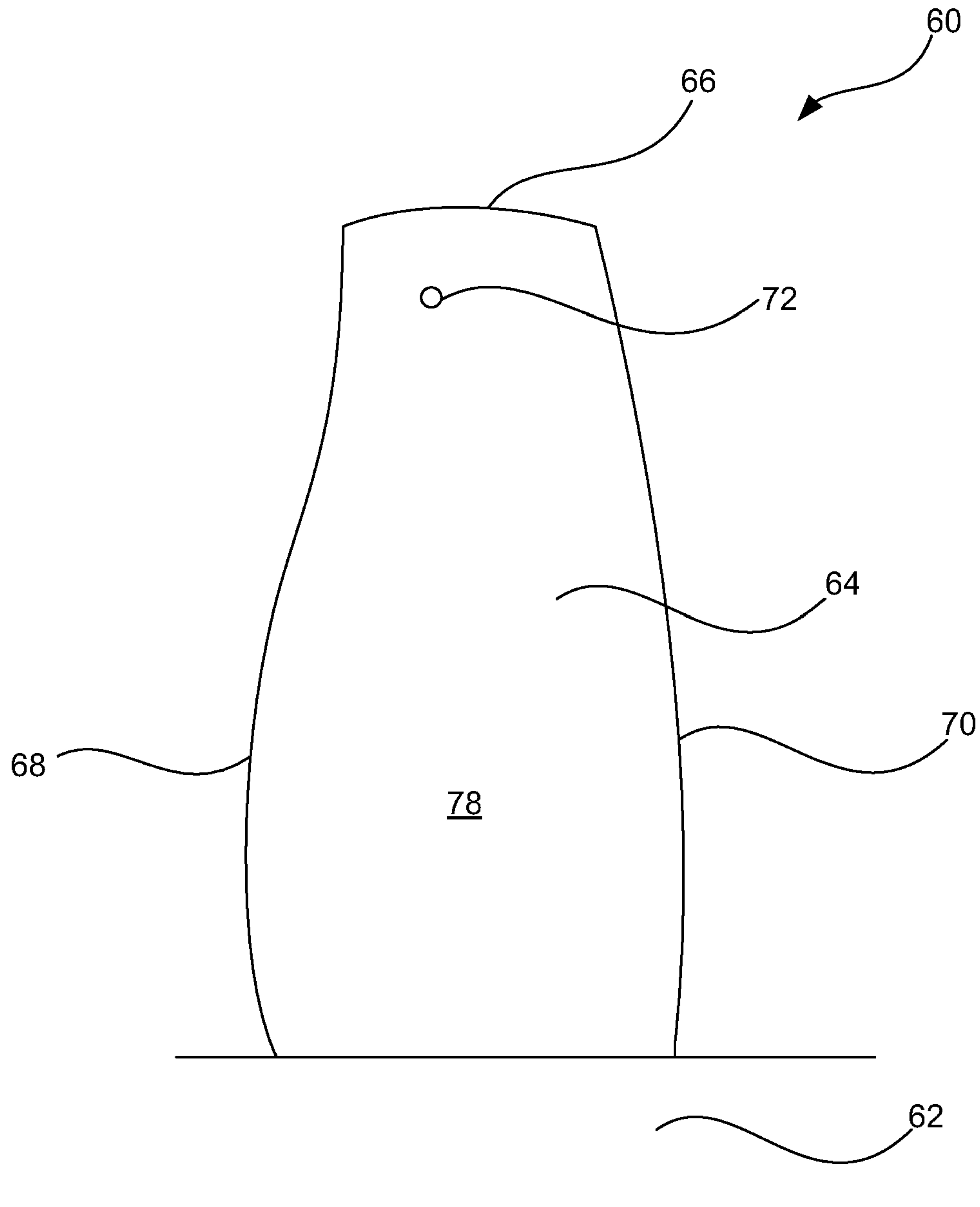

Figure 6a

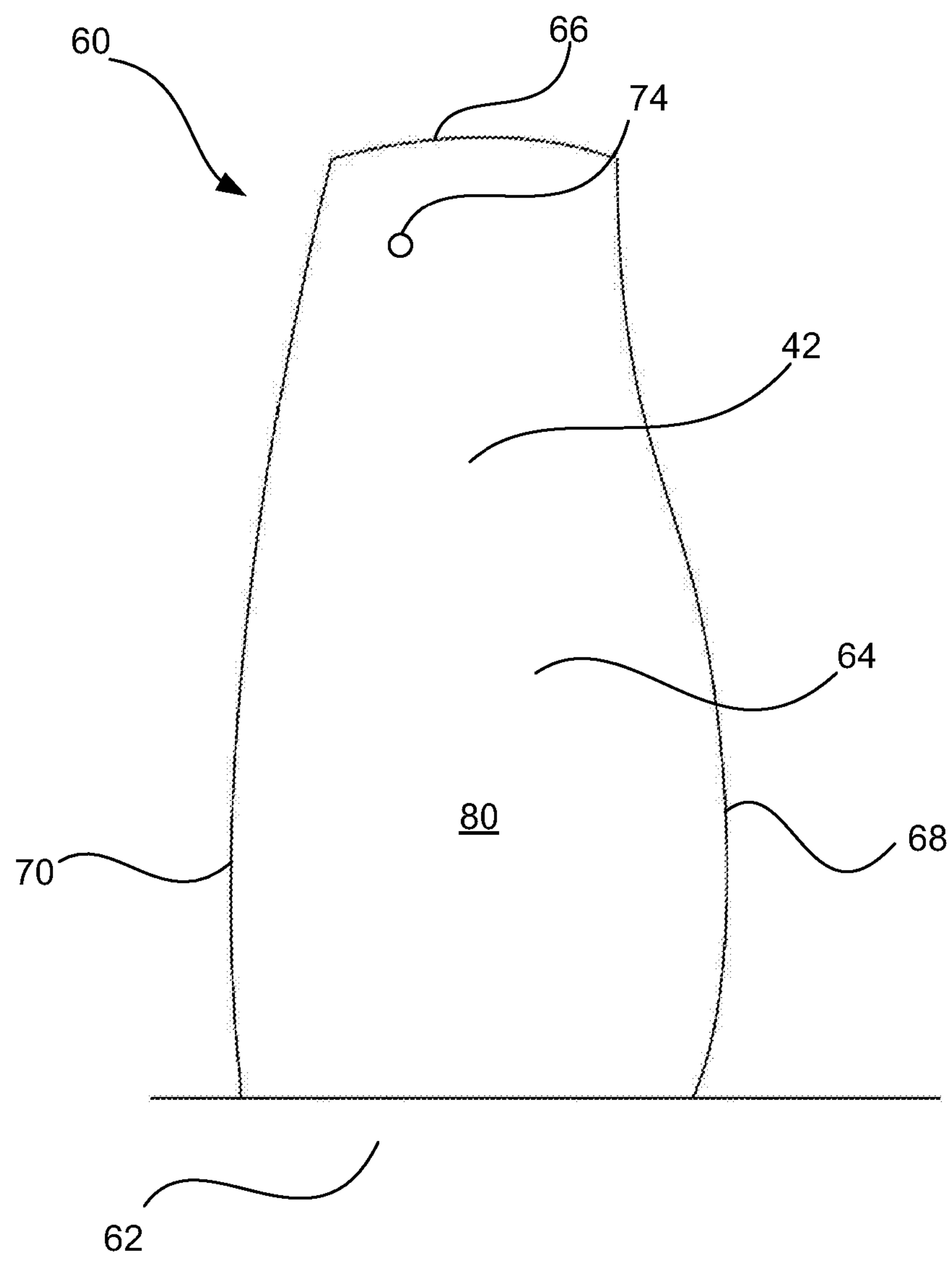

Figure 6b

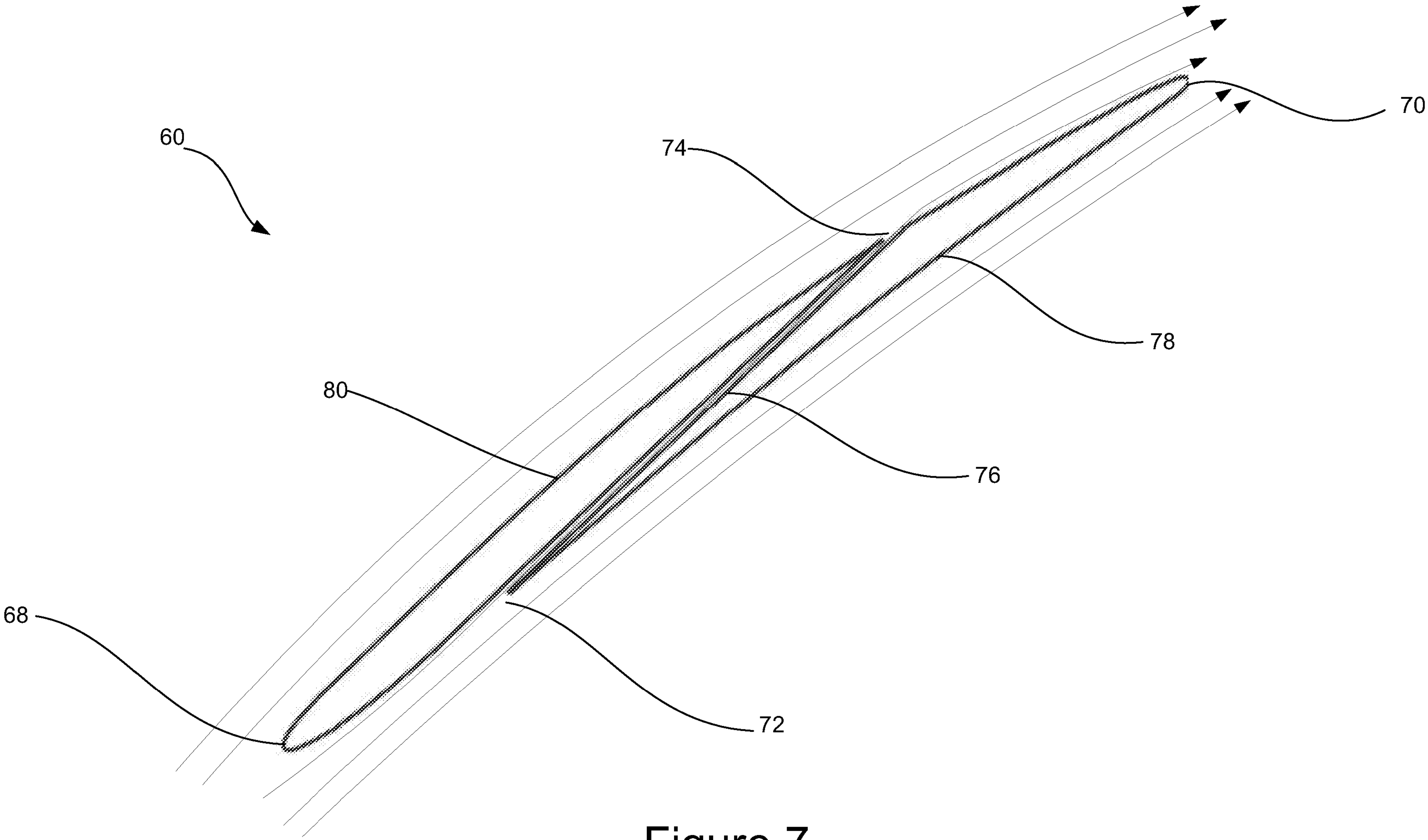

Figure 7

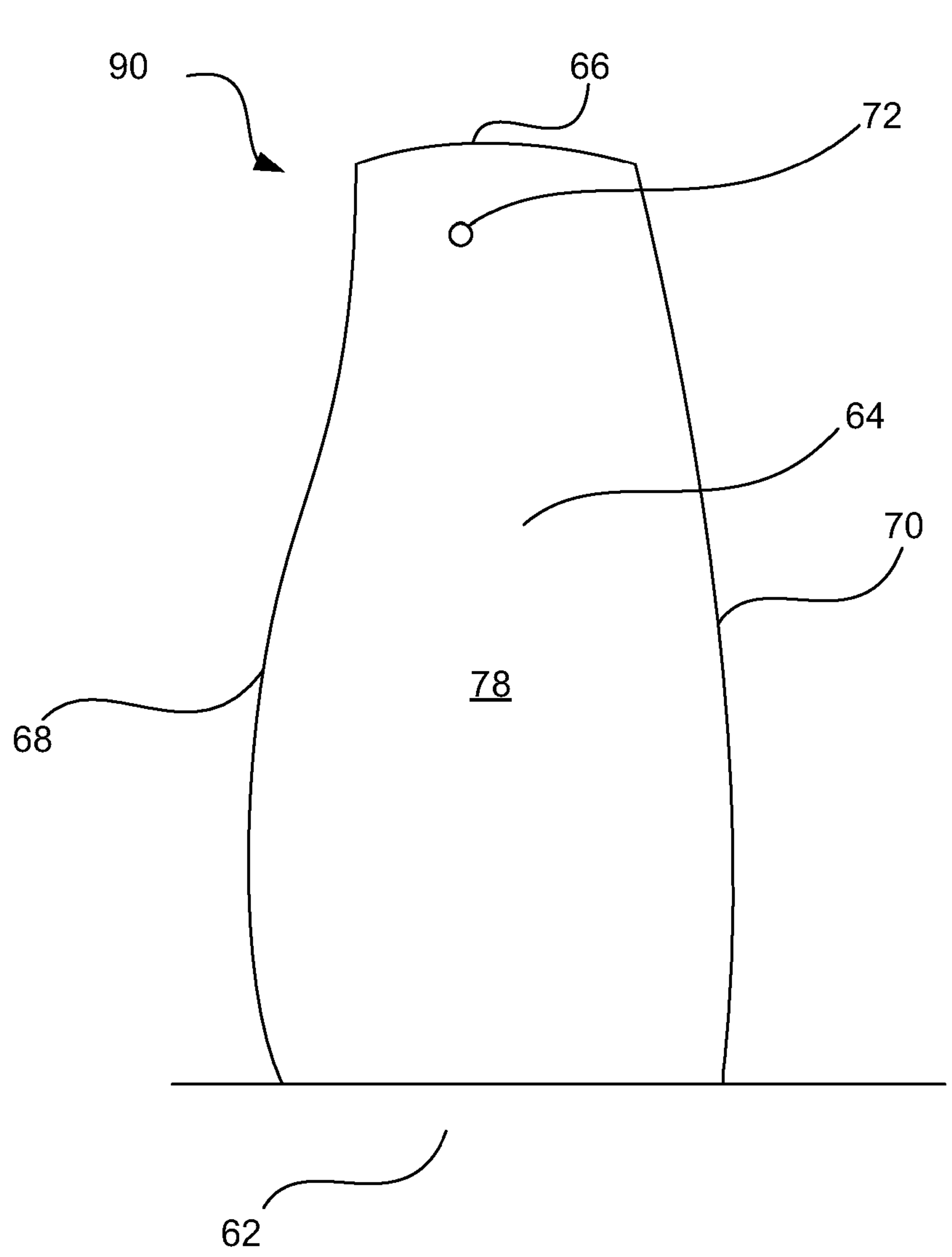

Figure 8a

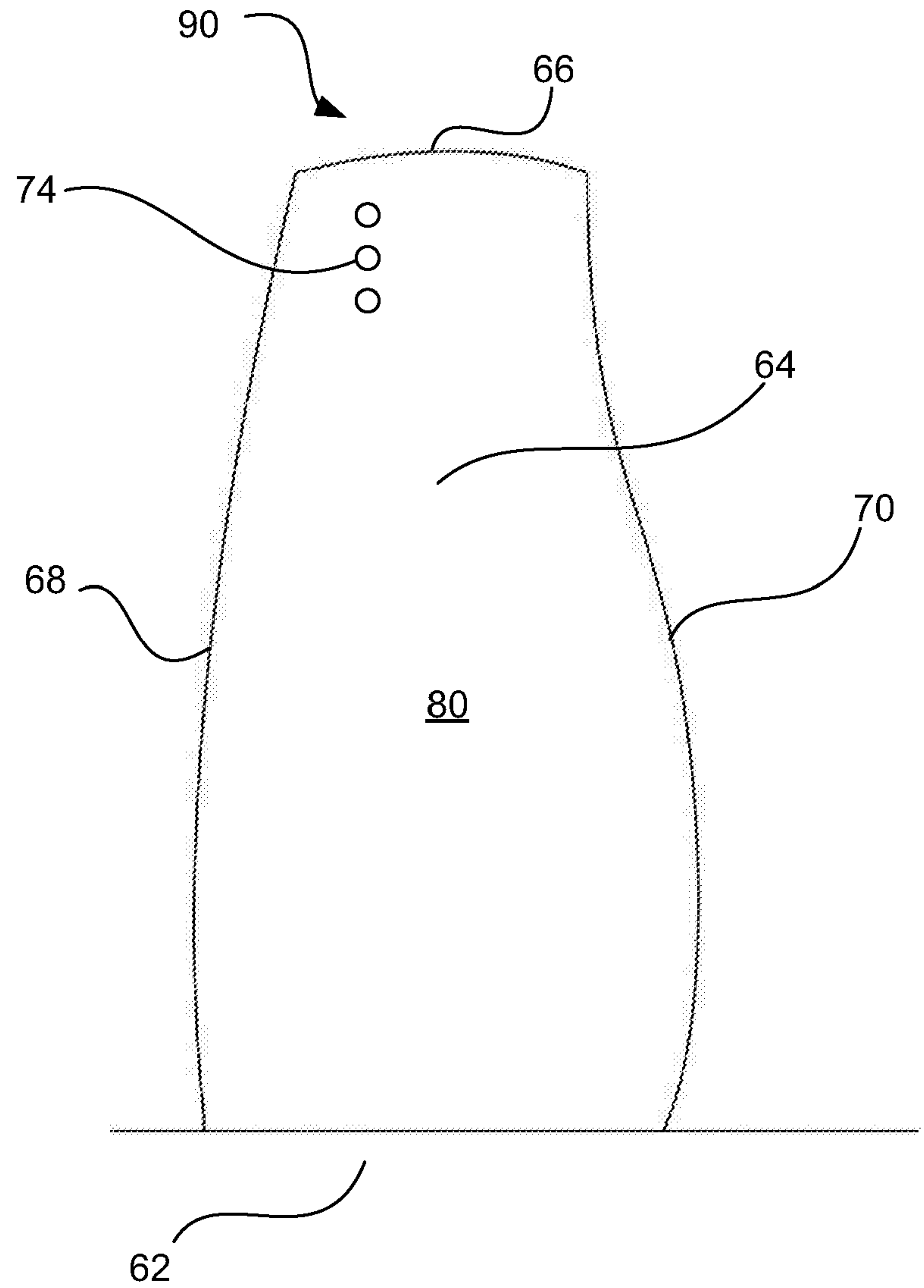

Figure 8b

A TURBOMACHINE BLADE

Field of the Disclosure

The present disclosure relates to a turbomachine blade, a rotor and a gas turbine engine.

Background of the Disclosure

A turbomachine, such as a gas turbine engine, may comprise a fan having a plurality of fan blades arranged around a fan disc. A fan blade typically comprises a root portion, for attaching the fan blade to the fan disc, and an aerofoil portion extending from the root and terminating at a tip.

Flutter is a flow phenomenon which affects the stability of the flow along turbomachine blades, in particular blades with high aspect ratios, e.g. fan blades. Flutter is a self-excited oscillation that occurs close to the natural frequency of the blade aerofoil and can result from unsteady aerodynamic loading. Flutter can cause undesirable blade oscillation and can lead to failure of the blade. Stall flutter is a particular type of dynamic instability which takes place when the flow along an aerofoil separates from the aerofoil and becomes unstable during the flutter oscillation. This can reduce the life span of the blade, in addition to being detrimental to the safety and efficiency of the gas turbine engine.

There is a need to develop an improved turbomachine blade to alleviate at least some of the aforementioned problems.

Summary of the Disclosure

According to a first aspect there is provided a turbomachine blade comprising: a root, an aerofoil portion and a tip, wherein: the aerofoil portion extends along a length between the tip and the root and comprises a pressure surface, a suction surface and at least one passageway extending through the aerofoil portion from an inlet at the pressure surface to an outlet at the suction surface, the passageway being configured to deliver airflow from the pressure surface to the suction surface; and the outlet is located within a section of the aerofoil portion extending between the tip and 50% of the length of the aerofoil portion from the tip.

It may be that the outlet is located within a section of the aerofoil portion extending between the tip and 25% of the length of the aerofoil portion from the tip.

It may be that at least a portion of the passageway or the inlet and the passageway is/are also located within the section of the aerofoil portion extending between the tip and 50% of the length of the aerofoil portion from the tip. It may be that at least a portion of the passageway or the inlet and the passageway is/are also located within the section of the aerofoil portion extending between the tip and 25% of the length of the aerofoil portion from the tip.

It may be that the inlet and outlet are located at different positions along the chord length of the aerofoil portion. The passageway may extend in a direction from a leading edge to a trailing edge of the aerofoil portion.

It may be that the aerofoil portion comprises a plurality of passageways extending through the aerofoil portion.

Each of the plurality of passageways may have a respective inlet at the pressure surface and a respective outlet at the suction surface.

It may be that the plurality of passageways have a common inlet, from which the plurality of passageways diverge to a plurality of outlets. The number of inlets on the pressure surface may be fewer than the number of outlets on the pressure surface.

It may be that the turbomachine blade is a fan blade. It may be that the turbomachine blade is a compressor blade.

According to another aspect there is provided a rotor comprising a plurality of the turbomachine blades as described above.

According to another aspect there is provided a gas turbine engine comprising a rotor as described above.

As noted elsewhere herein, the present disclosure may relate to a gas turbine engine. Such a gas turbine engine may comprise 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 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, a 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 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), 350cm, 360cm (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 270cm) 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 13 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 units in this paragraph being $Jkg^{-1}K^{-1}/(ms^{-1})^2$). 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 form

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 engine core. 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 stagnation 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 $Nkg^{-1}s$, 105 $Nkg^{-1}s$, 100 $Nkg^{-1}s$, 95 $Nkg^{-1}s$, 90 $Nkg^{-1}s$, 85 $Nkg^{-1}s$ or 80 $Nkg^{-1}s$. 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 $Nkg^{-1}s$ to 100 $Nkg^{-1}s$, or 85 $Nkg^{-1}s$ to 95 $Nkg^{-1}s$. 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: 160kN, 170kN, 180kN, 190kN, 200kN, 250kN, 300kN, 350kN, 400kN, 450kN, 500kN, or 550kN. 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 330kN to 420 kN, for example 350kN to 400kN. The thrust referred to above may be the maximum net thrust at standard atmospheric conditions at sea level plus 15 degrees C (ambient pressure 101.3kPa, 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 30kN to 35kN) at a forward Mach number of 0.8 and standard atmospheric conditions (according to the International Standard Atmosphere) at an altitude of 38000ft (11582m). 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 50kN to 65kN) 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 will now be described by way of example only, with reference to the Figures, in which:

Figure 1 is a sectional side view of a gas turbine engine;

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

Figure 3 is a partially cut-away view of a gearbox for a gas turbine engine;

Figure 4 is a side view of a known turbomachine blade;

Figures 5a and 5b are cross-sectional views near the tip of the turbomachine blade of Figure 4 showing air flow over the blade;

Figures 6a and 6b are side views of an example turbomachine blade;

Figure 7 is a cross-sectional view near the tip of the example turbomachine blade of Figure 6 showing air flow over the blade; and

Figures 8a and 8b are side views of a second example turbomachine blade.

Detailed Description

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

Figure 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 core exhaust 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 **Figure 2**. The low pressure turbine 19 (see Figure 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 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 **Figure 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 Figure 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 Figures 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 Figures 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 Figure 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 Figure 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 Figure 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.

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 Figure 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 exhaust 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 Figure 1), and a circumferential direction (perpendicular to the page in the Figure 1 view). The axial, radial and circumferential directions are mutually perpendicular.

Figure 4 shows a known turbomachine blade 42. The blade 42 is of conventional design and comprises a root 44, an aerofoil portion 46 and a tip 48. In an example, the blade 42 may be a fan blade. The root 44 is configured to secure the blade 42 into a hub. Typically, in a fan, a plurality of blades 42 are circumferentially arranged around and secured to a hub. In other arrangements, the blade 42 may be integrally formed with the hub (as in a bladed disk or ring) and so the root may be defined as the base of the aerofoil portion 46 adjacent the hub. The aerofoil portion 46 extends along a length between the root 44 and the tip 48. The aerofoil portion 46 also extends between a leading edge 54 and a trailing edge 56.

Figures 5a and 5b show cross-sectional views of the known turbomachine blade 42, the cross-section taken through the aerofoil portion 46, near to the tip 48. In use, air flows across the aerofoil portion 46 from the leading edge 54 to the trailing edge 56. Air flows over a pressure surface 50 and a suction surface 52 of the aerofoil portion 46. The air flowing across the suction surface 52 has a comparatively lower static pressure and higher velocity than the air flowing across the pressure surface 50.

Figure 5a shows air flowing across the aerofoil portion 46 under normal operating conditions. The flow is stable and attached to the suction surface 52. In contrast, **Figure 5b** shows air flowing across the aerofoil portion 46 under stall flutter conditions.

As discussed previously, flutter is an aerodynamic phenomenon which describes a self-excited oscillation of a blade due to the interaction of structural-dynamic and aerodynamic forces. In stall flutter conditions, flow across the aerofoil portion 46 becomes periodically separated from the aerofoil during the oscillations. In particular, as shown in Figure 5b, the flow rate of air adjacent to the suction surface

52 reduces, such that the flow adjacent to the suction surface 52 becomes separated from the suction surface 52. This results in unstable flow and the formation of eddies 58 across the suction surface 52. Such flow separation and unstable flow is undesirable and can result in the reduction of performance and life of the blade 42.

Figures 6a and 6b show an example turbomachine blade 60 according to the present disclosure. The blade 60 may be any turbomachine blade having a high-aspect ratio, including a fan blade or a compressor blade. The blade 60 comprises a root 62, an aerofoil portion 64 and a tip 66. The root 62 is configured to secure the blade 60 into a hub. Typically, a plurality of blades 60 may be circumferentially arranged around and secured to a hub. The aerofoil portion 64 extends along a length between the root 62 and the tip 66. The aerofoil portion 64 also extends between a leading edge 68 and a trailing edge 70. The aerofoil portion 64 has a pressure surface 78 and a suction surface 80. **Figure 6a** shows a plan view in the direction of the pressure surface 78, whilst **Figure 6b** shows a plan view in the direction of the suction surface 80.

The aerofoil portion 64 comprises a passageway 76 (see Figure 7) extending through the aerofoil portion 64. The passageway 76 extends from an inlet 72 at the pressure surface 78 to an outlet 74 at the suction surface 80. The passageway 76, inlet 72 and outlet 74 are located in a section of the aerofoil portion 64 near the tip 66 (i.e. nearer to the tip 66 than the root 62). In this example, the passageway 76, inlet 72 and outlet 74 are located within a section of the aerofoil portion 64 extending between the tip 66 and 25% of the length of the aerofoil portion 64 from the tip 66. In other examples, the passageway 76, inlet 72 and outlet 74 can be located within a section of the aerofoil portion 64 extending between the tip 66 and 50% of the length of the aerofoil portion 64 from the tip 66. In further examples, the inlet 72 may be located outside a section of the aerofoil portion 64 extending between the tip 66 and 25% of the length of the aerofoil portion 64 from the tip 66, or even outside a section of the aerofoil portion 64 extending between the tip 66 and 50% of the length of the aerofoil portion 64 from the tip 66, whilst at least a portion of the passageway 76 and the outlet 74 are located within these sections.

The passageway 76 has a circular cross-section. In other examples, the cross-section of the passageway 76 may not be circular and may, for example, be rectangular, square-shaped, elliptical, triangular or wedge-shaped. The cross-section of the passageway 76 may change along its length such that it has one shape at the inlet 72 and another shape at the outlet 74.

Figure 7 shows a cross-section of the blade 60 taken through the passageway 76. As shown, the inlet 72 and outlet 74 are located at different positions along the chord length of the aerofoil portion 64. Specifically, the outlet 74 is located closer to the trailing edge 70 (or, conversely, further from the leading edge 68) than the inlet 72. In particular, in this example, the outlet 74 is located at a position which is closer to the trailing edge 70 than the leading edge 68 and the inlet 72 is located at a position which is closer to the leading edge 68 than the trailing edge 70. The passageway 76 is therefore oblique to the pressure surface 78 and suction surface 80. The passageway 76 is thus angled in a direction which has a chordwise component and thus extends from the leading edge 68 towards the trailing edge 70.

Figure 7 illustrates how air flow interacts with the blade 60. Air flows in a direction from the leading edge 68 to the trailing edge 70 and along the pressure surface 78 and the suction surface 80. The passageway 76 is configured to allow air to flow through the blade 60 from the pressure surface 78 to the suction surface 80 of the aerofoil portion 64. Specifically, a portion of the air flowing along the pressure surface 78 enters the passageway 76 at the inlet 72 and flows along the passageway 76 before exiting via the outlet 74 at the suction surface 80. As shown, the air exiting the outlet 74 flows along the suction surface 80.

It is understood that by allowing a portion of the air to flow through the aerofoil portion 64, from the pressure surface 78 to the suction surface 80, the air flowing adjacent to the suction surface 80 remains attached to the suction surface 80. It is understood that this minimises the effects of stall flutter by ensuring that the air flow remains attached to the aerofoil portion 64 at the suction surface 80 during flutter, thereby minimising unstable flow over the aerofoil portion 64.

The angle of entry of air to the inlet 72 and the angle of exit at the outlet 74 may be varied by modifying the angle of the passageway 76 with respect to the aerofoil portion 64, or by modifying the size and shape of the inlet 72 and/or outlet 74.

Figures 8a and 8b show a second example turbomachine blade 90 according to the present disclosure. The second example blade 90 comprises similar features to that of the first example as previously described, with like reference numerals indicating like features. This example differs in that it comprises a plurality of passageways rather than a single passageway.

In this example, the plurality of passageways 76 have a common inlet 72 at the pressure surface 78, from which the plurality of passageways diverge to a plurality of outlets 74 at the suction surface 80. The plurality of outlets 74 are spaced from one another along a spanwise direction of the blade 90. In this example, three outlets 74 are shown, but in other examples there may be any number of outlets 74. In other examples, there may be a plurality of inlets 72 at the pressure surface 78. There may be a plurality of passageways 76 diverging from each of the plurality of inlets 72 to a plurality of outlets 74 at the suction surface 80. The number of inlets 72 at the pressure surface 78 may be fewer than the number of outlets 74 at the suction surface 80. In further examples, the plurality of passageways 76 may each have a respective inlet 72 at the pressure surface 78 and a respective outlet 74 at the suction surface 80. The number of inlets 72 at the pressure surface 78 may therefore be the same as the number of as the outlets 74 at the suction surface 80.

Similar to the example previously described, the plurality of passageways 76, inlet 72 and plurality of outlets 74 are located in a section of the aerofoil portion 64 near the tip 66. In this example, the plurality of passageways 76, inlet 72 and plurality of outlets 74 are located within a section of the aerofoil portion 64 extending between the tip 66 and 25% of the length of the aerofoil portion 64 from the tip 66. In other examples, the plurality of passageways 76, inlet 72 and plurality of outlets 74 can be located within a section of the aerofoil portion 64 extending between the tip 66 and 50% of the length of the aerofoil portion 64 from the tip 66. In further examples, the inlet 72 may be located outside a section of the aerofoil portion 64 extending between the tip 66 and 25% of the length of the aerofoil portion 64 from the tip 66, or even outside a section of the aerofoil portion 64 extending between the tip 66 and 50% of the length of the aerofoil portion 64 from the tip 66, whilst the plurality of passageways 76 and plurality of outlets 74 are located within these sections.

The interaction of air flow with the aerofoil portion 64 of the blade 90 is similar to that described for the previous example, and would appear the same in cross-section to that shown in Figure 7. Specifically, the outlets 74 are located closer to the trailing edge 70 (or, conversely, further from the leading edge 68) than the inlet 72. In particular, in this example, the outlets 74 are located at a position which is closer to the trailing edge 70 than the leading edge 68 and the inlet 72 is located at a position which is closer to the leading edge 68 than the trailing edge 70. The passageways 76 are therefore oblique to the pressure surface 78 and suction surface 80. The passageways 76 are thus angled in a direction which has a chordwise component and thus extend from the leading edge 68 towards the trailing edge 70. Each of the plurality of outlets 74 may be located at different positions along the chord length of the aerofoil portion 64.

Air flows in a direction from the leading edge 68 to the trailing edge 70 and along the pressure surface 78 and the suction surface 80. The plurality of passageways 76 are configured to allow air to flow through the blade 60 from the pressure surface 78 to the suction surface 80 of the aerofoil portion 64. Specifically, a portion of the air flowing along the pressure surface 78 enters the passageways 76 at the inlet 72 and flows along the passageways 76 before exiting via the outlets 74 at the suction surface 80. Air exiting the outlets 74 flows along the suction surface 80. As described previously, it is understood that by allowing a portion of air to flow through the aerofoil portion 64, from the pressure surface 78 to the suction surface 80, the air flowing adjacent to the suction surface 80 remains attached to the suction surface 80.

Varying the number of inlets 72 and outlets 74 may enable the rate of air flow through the blade 90 to be controlled. Further, the spread of air flow on the suction surface 80 may be controlled by varying the number and position of outlets 74. The shape of the or each outlet 74 or the cross-sectional area of the or each passageway

76 (i.e. the diameter) may also be varied in order to provide the desired rate and/or distribution of air flow.

It will be understood that the invention is not limited to the embodiments 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 turbomachine blade comprising: a root (62), an aerofoil portion (64) and a tip (66), wherein:

the aerofoil portion (64) extends along a length between the tip (66) and the root (62) and comprises a pressure surface (78), a suction surface (80) and at least one passageway (76) extending through the aerofoil portion from an inlet (72) at the pressure surface (78) to an outlet (74) at the suction surface (80), the passageway (76) being configured to deliver airflow from the pressure surface (78) to the suction surface (80); and

wherein the outlet (74) is located within a section of the aerofoil portion (64) extending between the tip and 50% of the length of the aerofoil portion (64) from the tip (66).

2. The turbomachine blade according to claim 1, wherein the outlet (74) is located within a section of the aerofoil portion (64) extending between the tip (66) and 25% of the length of the aerofoil portion (64) from the tip (66).

3. The turbomachine blade according to any preceding claim, wherein at least a portion of the passageway (76) or the inlet (72) and the passageway (76) is/are located within the section of the aerofoil portion (64) extending between the tip (66) and 50% of the length of the aerofoil portion (64) from the tip (66).

4. The turbomachine blade according to claim 3, wherein at least a portion of the passageway (76) or the inlet (72) and the passageway (76) is/are located within the section of the aerofoil portion (64) extending between the tip (66) and 25% of the length of the aerofoil portion (64) from the tip (66).

5. The turbomachine blade according to any preceding claim, wherein the inlet (72) and outlet (74) are located at different positions along the chord length of the aerofoil portion (64).

6. The turbomachine blade according to claim 5, wherein the passageway (76) extends in a direction from a leading edge (68) to a trailing edge (70) of the aerofoil portion (64).

7. The turbomachine blade according to any preceding claim, wherein the aerofoil portion (64) comprises a plurality of passageways (76) extending through the aerofoil portion (64).

8. The turbomachine blade according to claim 7, wherein each of the plurality of passageways (76) has a respective inlet (72) at the pressure surface (78) and a respective outlet (74) at the suction surface (80).

9. The turbomachine blade according to claim 7, wherein the plurality of passageways (76) have a common inlet (72), from which the plurality of passageways (76) diverge to a plurality of outlets (74).

10. The turbomachine blade according to any preceding claim, wherein the turbomachine blade is a fan blade.

11. A rotor comprising a plurality of turbomachine blades according to any preceding claim.

12. A gas turbine engine comprising a rotor according to claim 11.

Intellectual Property Office

Application No:	GB1916667.7	**Examiner:**	Rachel Smith
Claims searched:	1-12	**Date of search:**	23 March 2020

Patents Act 1977: Search Report under Section 17

Documents considered to be relevant:

Category	Relevant to claims	Identity of document and passage or figure of particular relevance
X	1-8, 10-12	GB 2481822 A (ROLLS ROYCE PLC) See figures, page 2 line 34, page 3 line 31 - page 4 line 4 and page 4 lines 17-18.
X	1-7, 10-12	WO 2015/065659 A1 (UNITED TECHNOLOGIES CORP) See figures 3 & 8B and paragraphs 52-55 & 63.
X	1-6, 11, 12	EP 1785589 A1 (SIEMENS AG) See figures and paragraphs 1, 7 & 13.
X	1-9, 11, 12	EP 1947294 A2 (UNITED TECHNOLOGIES CORP) See figures and paragraphs 10, 11 & 16.
X	1-7, 9, 11, 12	EP 1118747 A2 (ROLLS ROYCE PLC) See figures and paragraphs 31, 32 & 40.
X	1, 3, 7, 8, 11	GB 736835 A (MASCHF AUGSBURG NUERNBERG AG) See figure 5 and page 3 lines 33-56.

Categories:

X	Document indicating lack of novelty or inventive step	A	Document indicating technological background and/or state of the art.
Y	Document indicating lack of inventive step if combined with one or more other documents of same category.	P	Document published on or after the declared priority date but before the filing date of this invention.
&	Member of the same patent family	E	Patent document published on or after, but with priority date earlier than, the filing date of this application.

Field of Search:

Search of GB, EP, WO & US patent documents classified in the following areas of the UKC^{X} :

Worldwide search of patent documents classified in the following areas of the IPC

F01D

The following online and other databases have been used in the preparation of this search report

WPI, EPODOC

Intellectual Property Office

International Classification:

Subclass	Subgroup	Valid From
F01D	0005/14	01/01/2006
F01D	0005/16	01/01/2006
F01D	0005/18	01/01/2006
F01D	0025/06	01/01/2006