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(54) **GEARED TURBOFAN GAS TURBINE ENGINE ARCHITECTURE**

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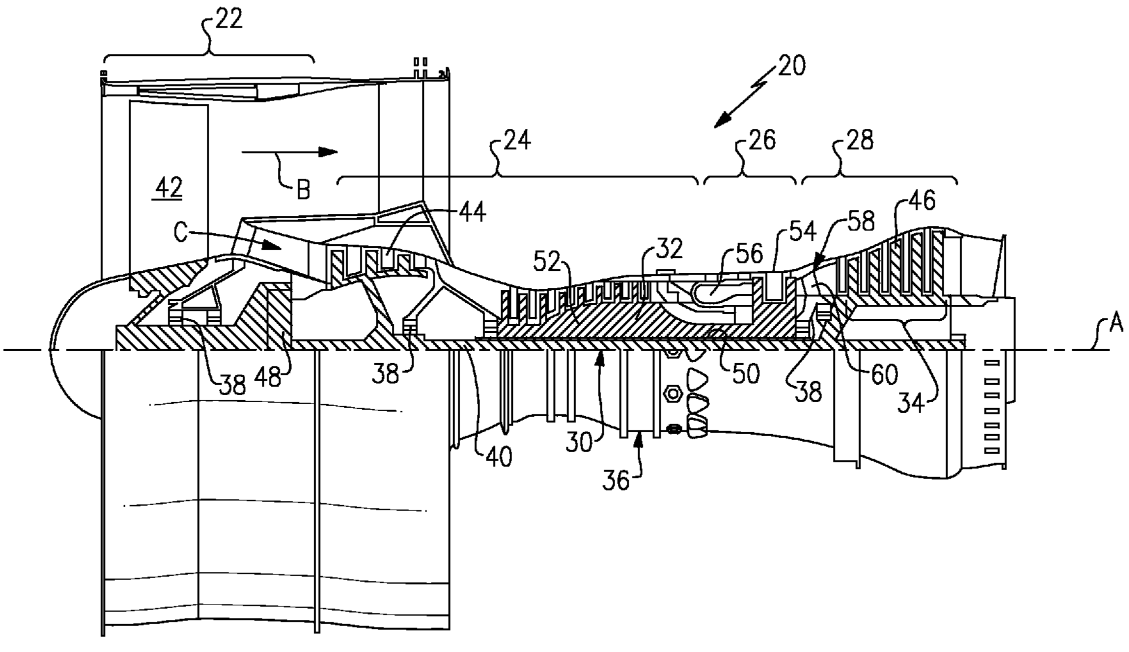
CPC . ***F02C 3/107*** (2013.01); ***F02C 7/06*** (2013.01); *F05D 2220/32* (2013.01); *F05D 2260/40311* (2013.01)

(57) **ABSTRACT**

A gas turbine engine typically includes a fan section, a compressor section, a combustor section and a turbine section. A speed reduction device such as an epicyclical gear assembly may be utilized to drive the fan section such that the fan section may rotate at a speed different than the turbine section so as to increase the overall propulsive efficiency of the engine. In such engine architectures, a shaft driven by one of the turbine sections provides an input to the epicyclical gear assembly that drives the fan section at a speed different than the turbine section such that both the turbine section and the fan section can rotate at closer to optimal speeds providing increased performance attributes and performance by desirable combinations of the disclosed features of the various components of the described and disclosed gas turbine engine.



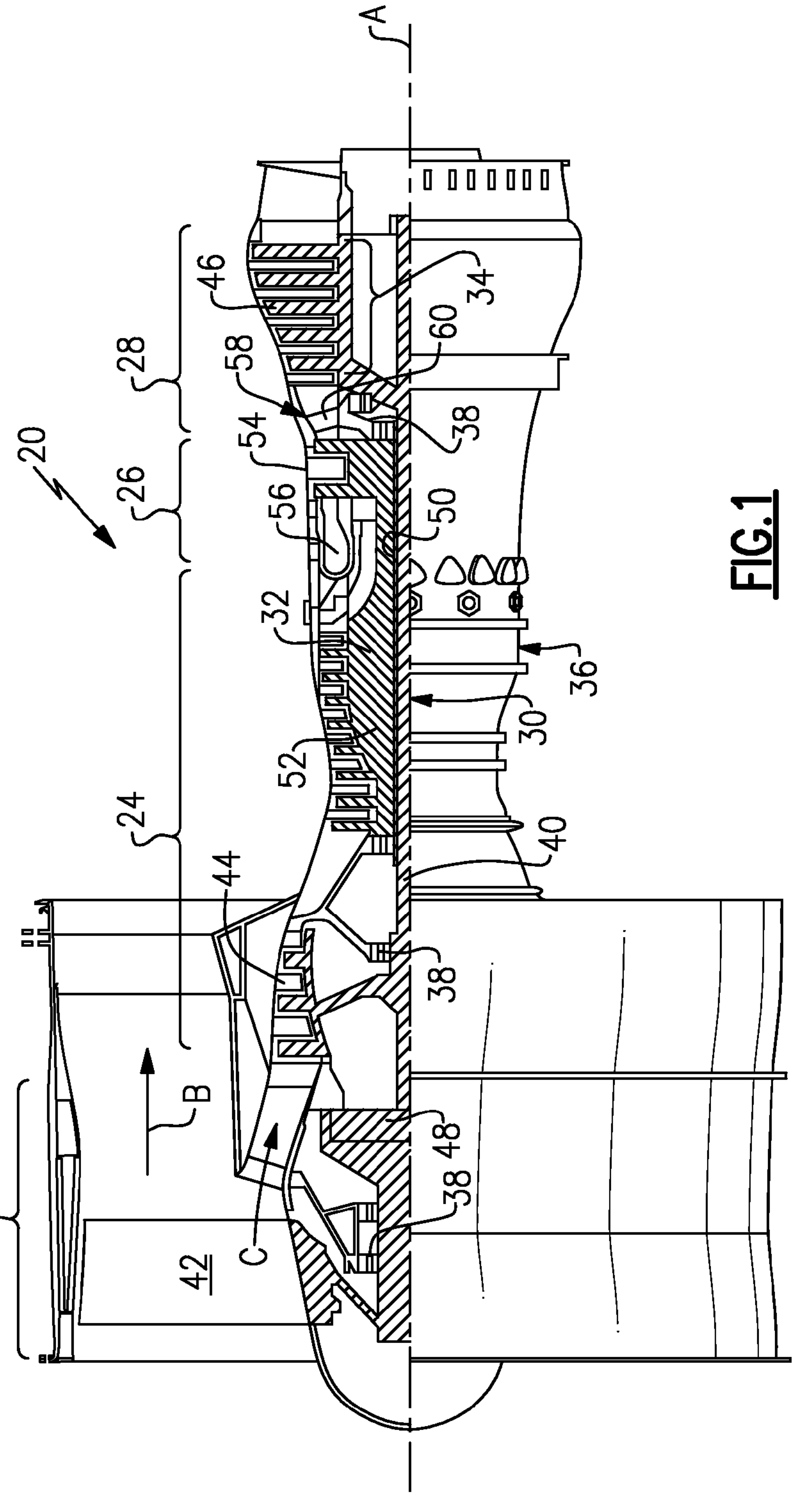


FIG.1

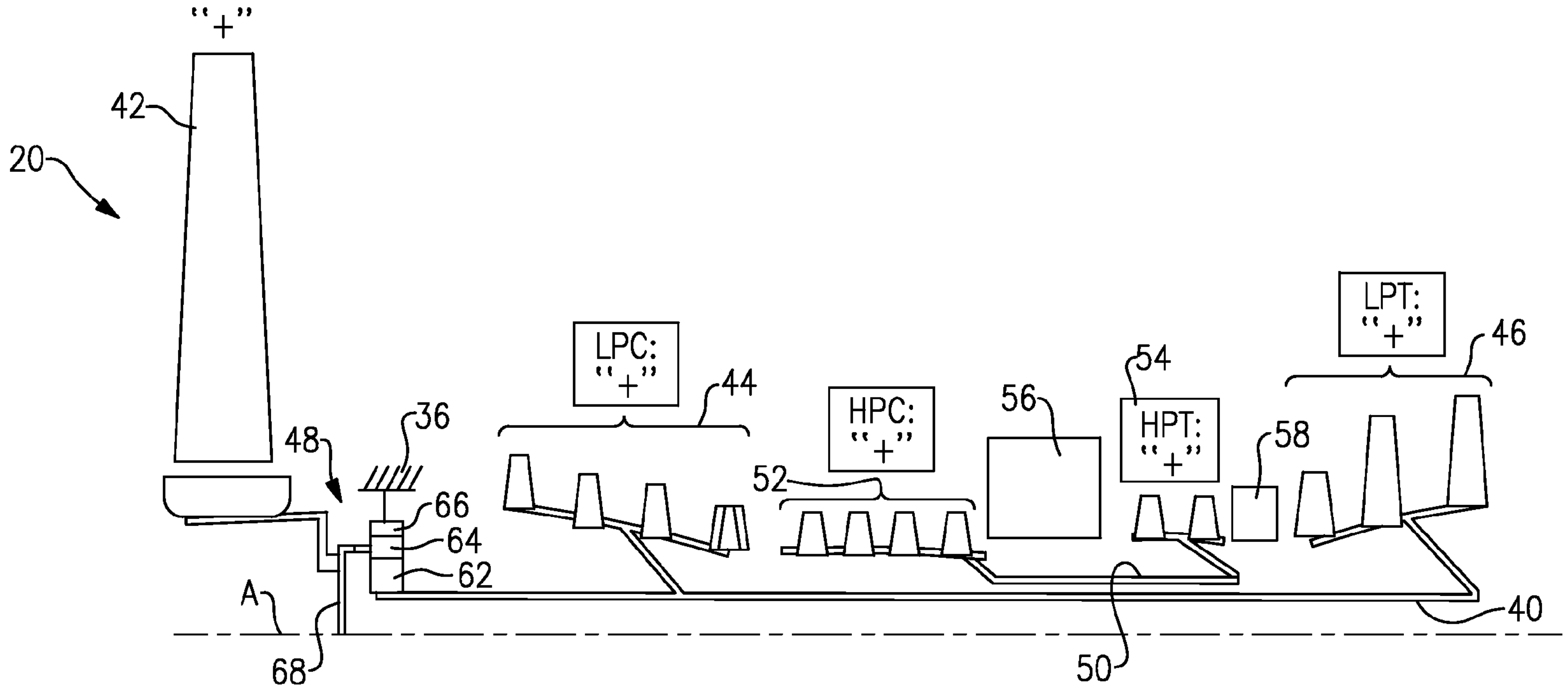


FIG.2

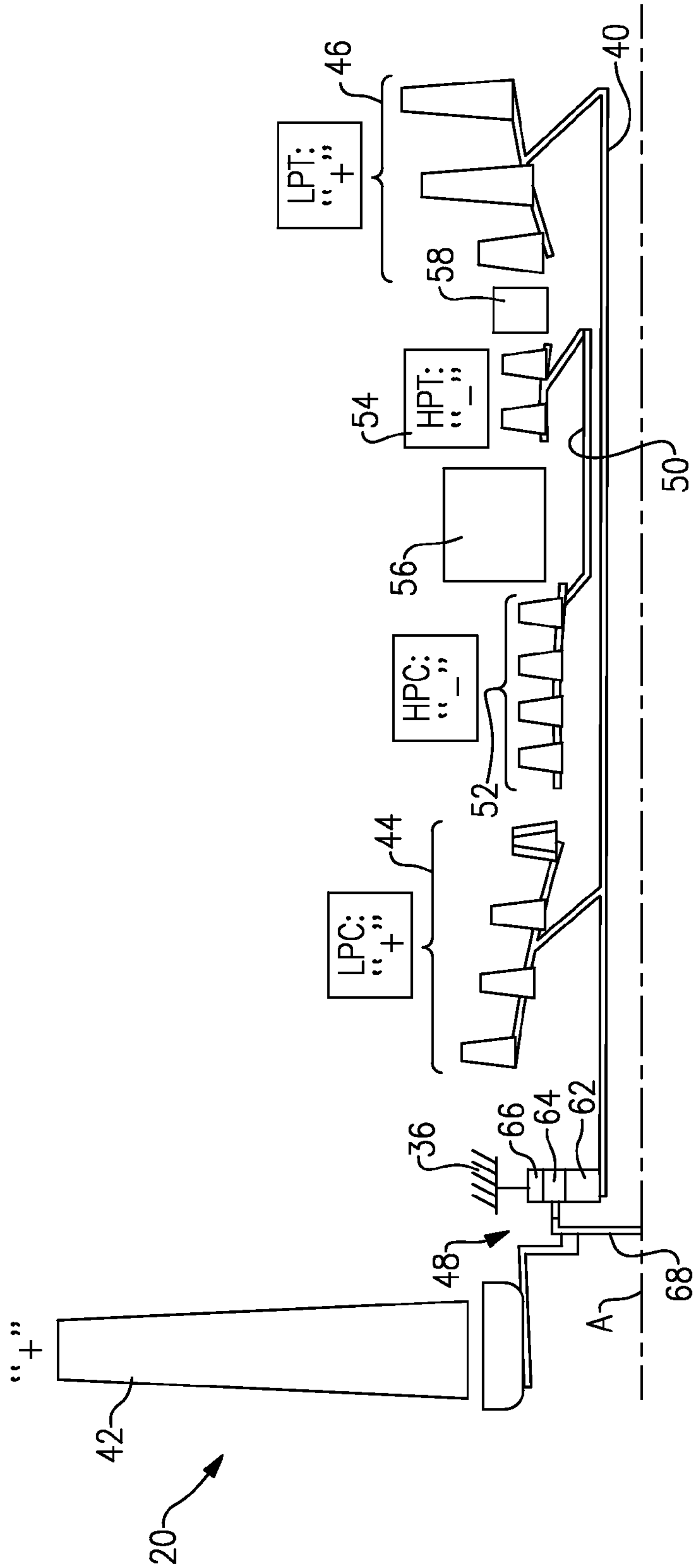


FIG.3

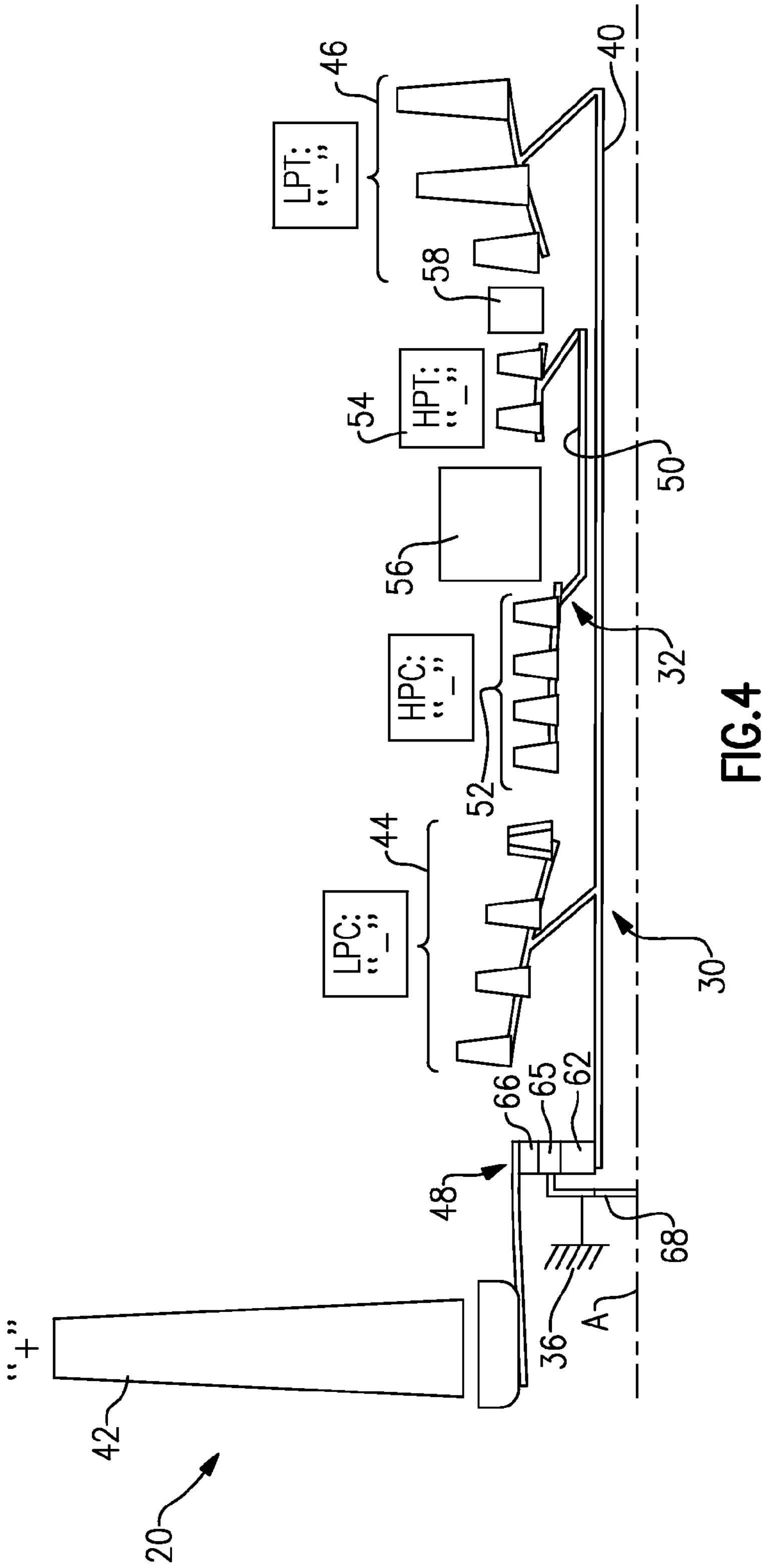


FIG.4

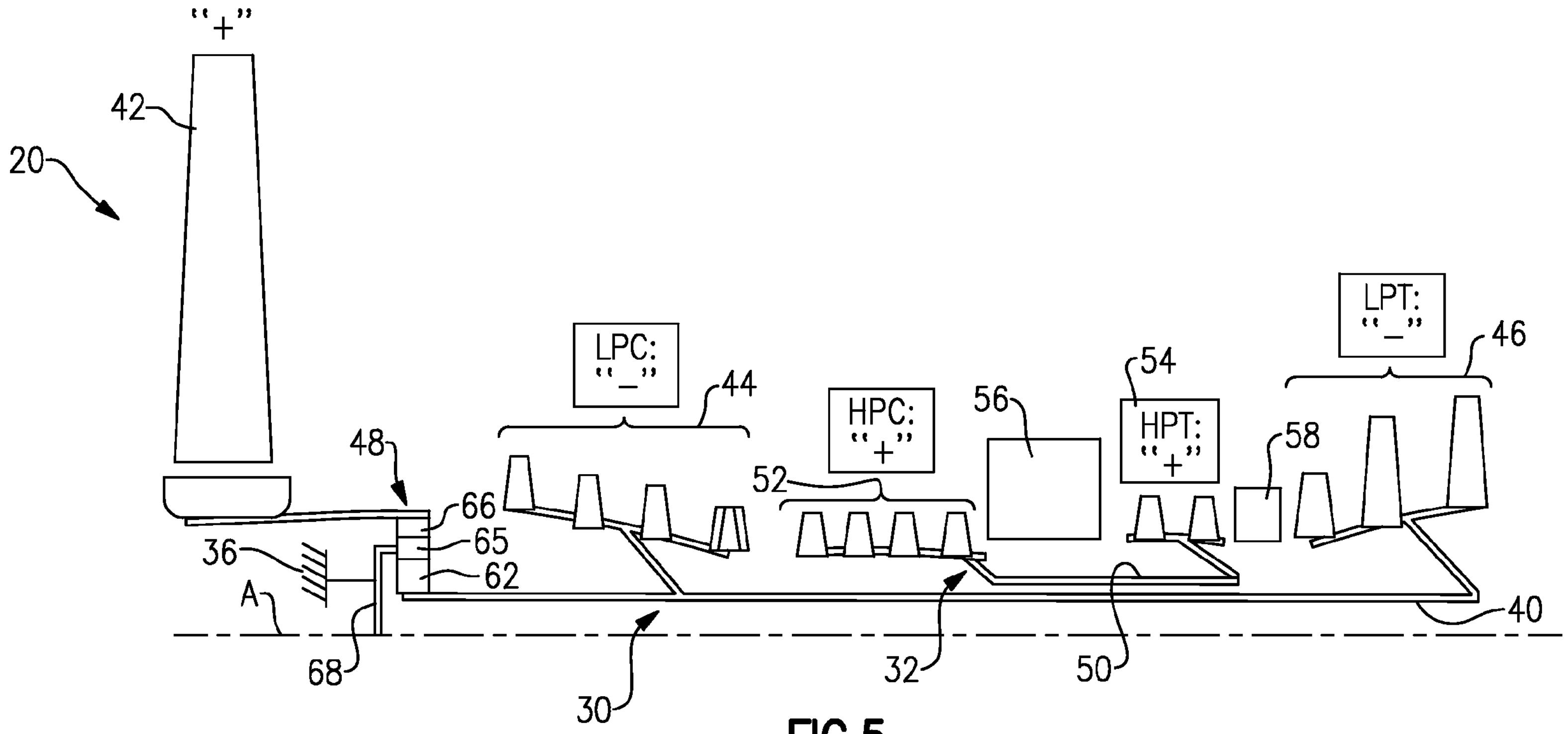


FIG.5

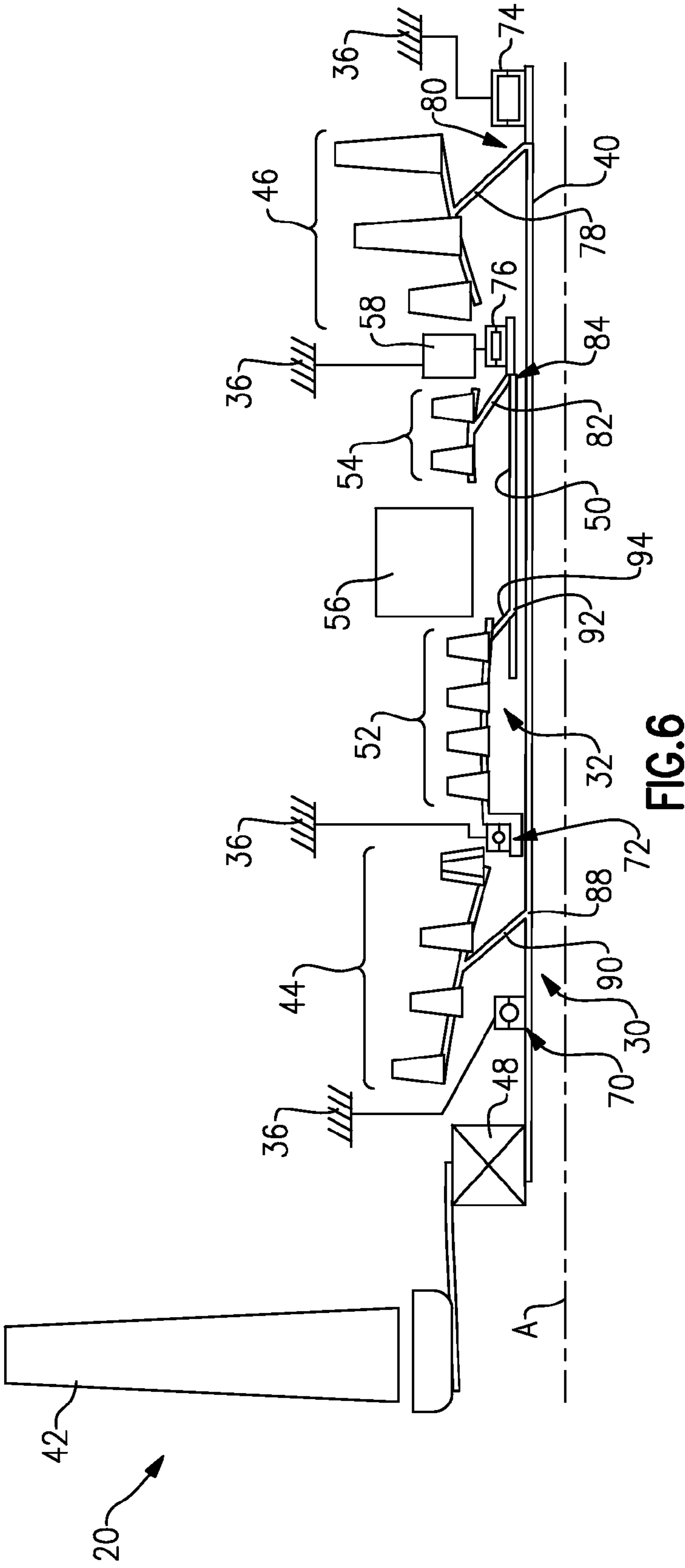

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FIG.6

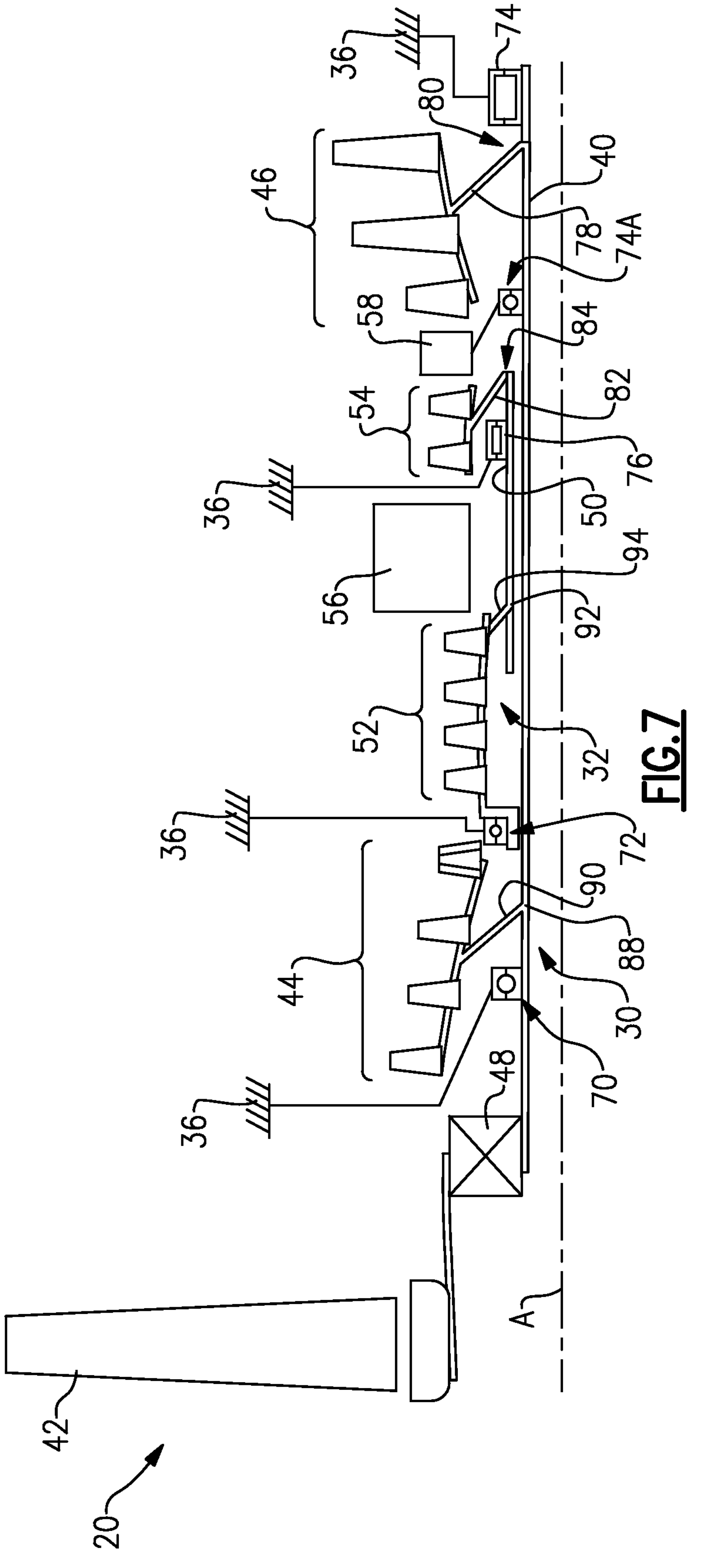
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FIG. 7

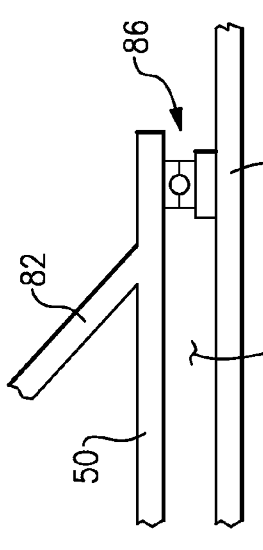
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FIG.8B

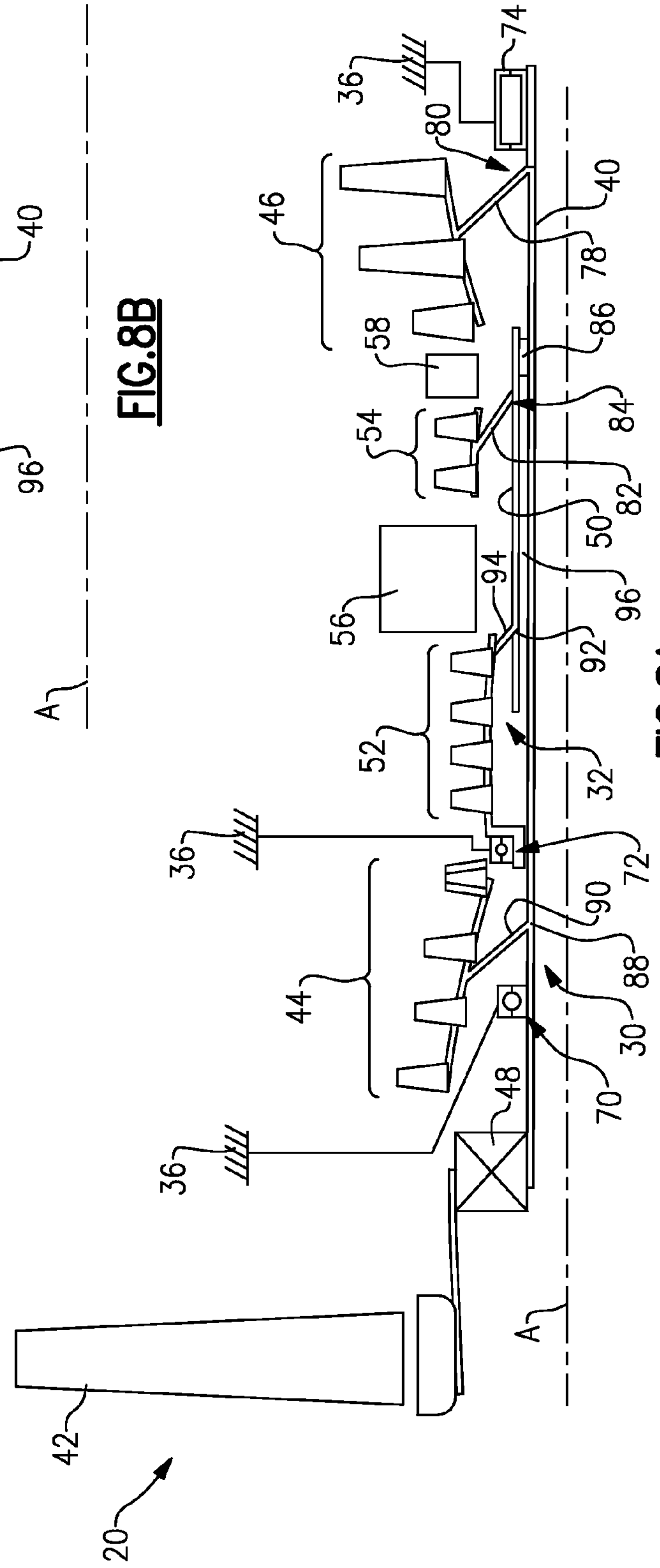
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FIG.8A

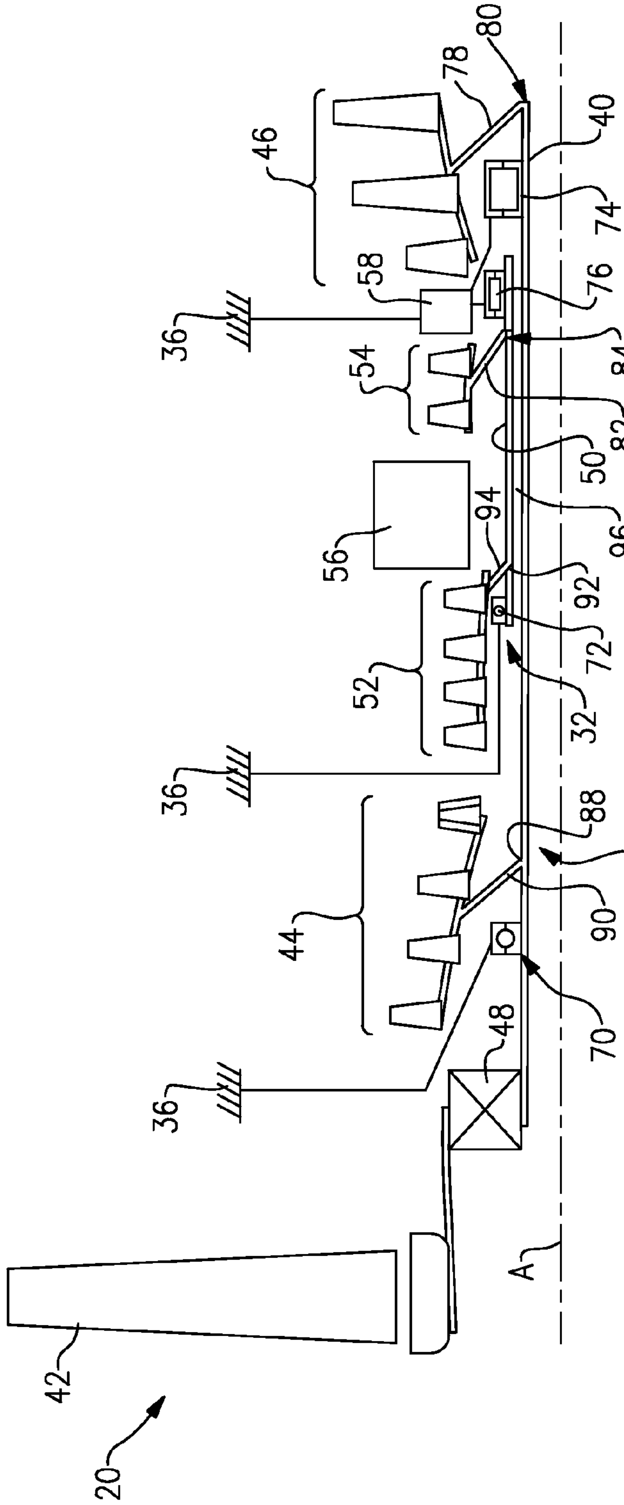
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FIG.9

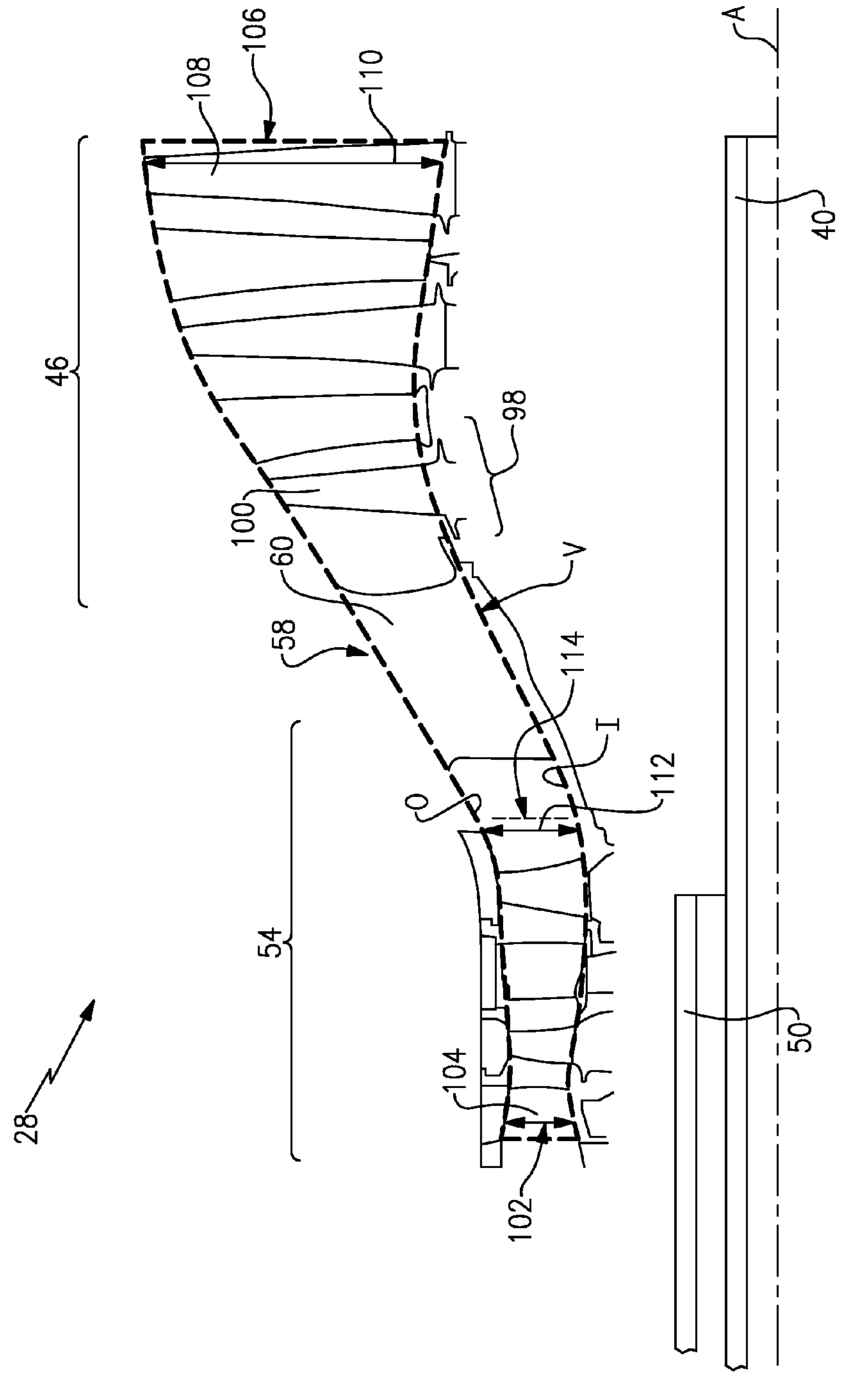

FIG.10

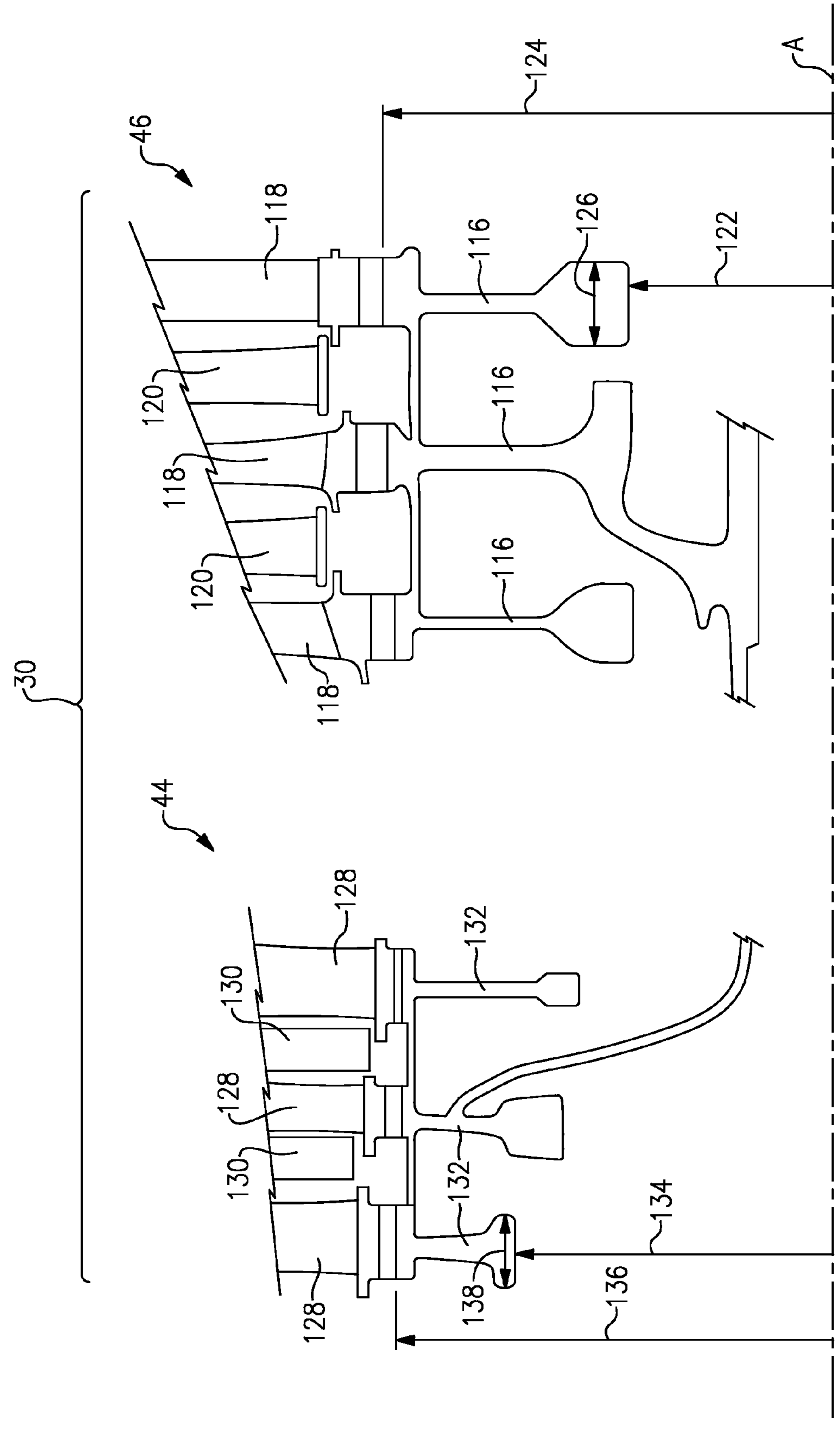
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FIG.11

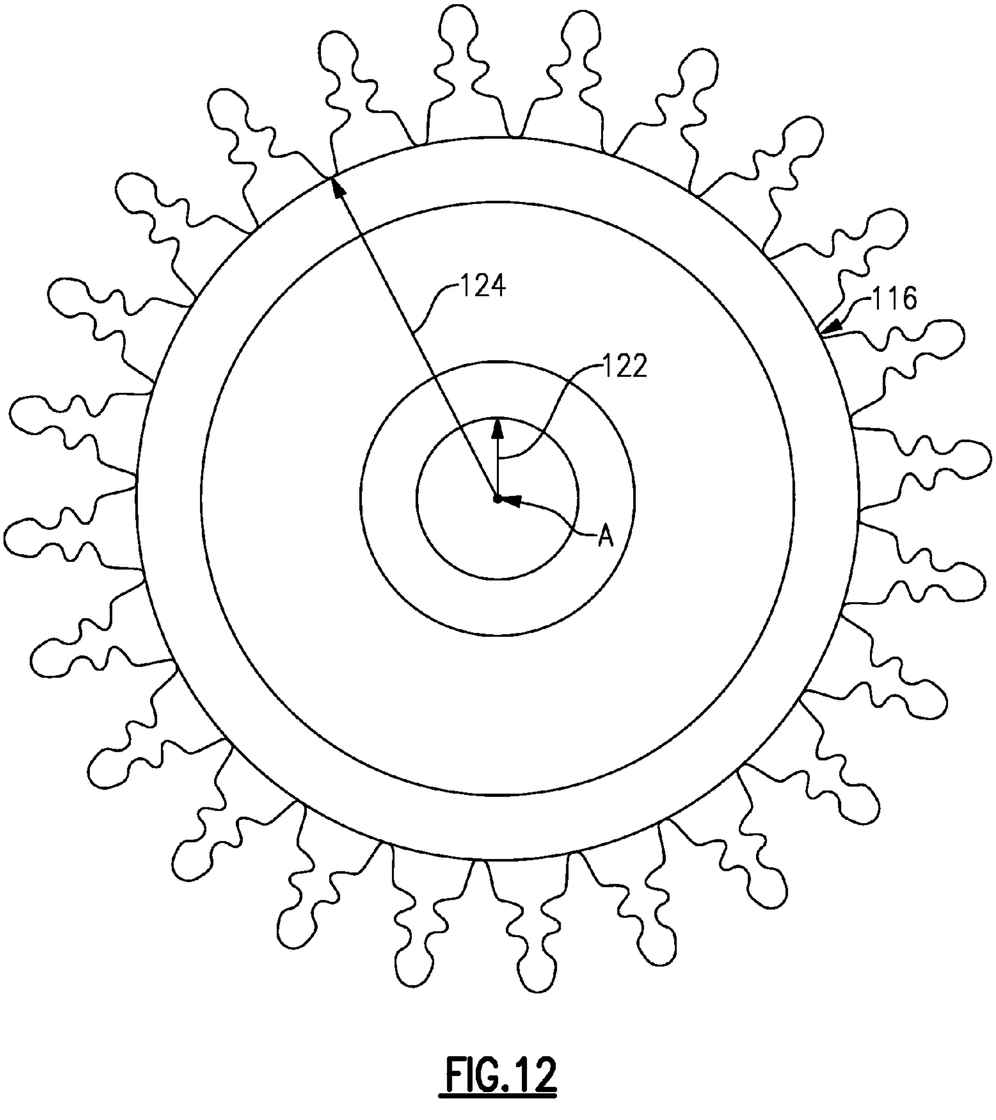

FIG.12

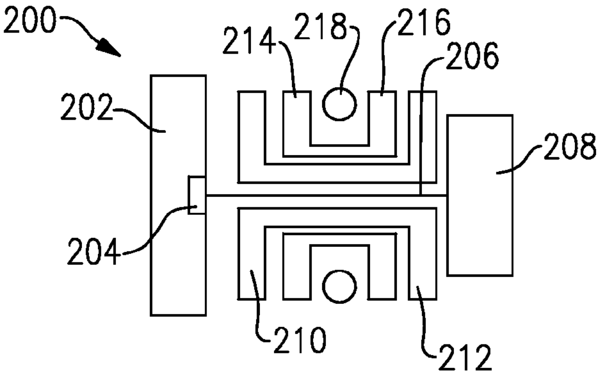

FIG.13

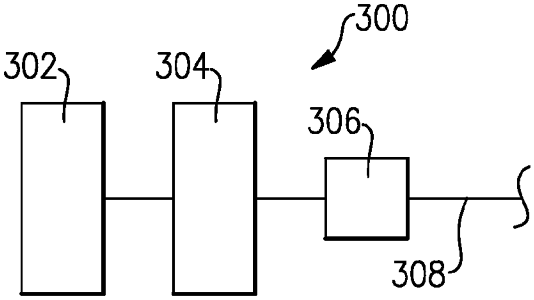

FIG.14

GEARED TURBOFAN GAS TURBINE ENGINE ARCHITECTURE

CROSS REFERENCE TO RELATED APPLICATION

[0001] This application is a continuation-in-part of International Application No. PCT/US13/23559 filed Jan. 29, 2013, which claims priority to of U.S. application Ser. No. 13/645,606 filed Oct. 5, 2012, now U.S. Pat. No. 8,935,913 granted Jan. 20, 2015, which was a continuation in part of U.S. application Ser. No. 13/363,154 filed on Jan. 31, 2012 and claims priority to U.S. Provisional Application No. 61/653,745 filed on May 31, 2012.

BACKGROUND

[0002] A gas turbine engine typically includes a fan section, a compressor section, a combustor section and a turbine section. Air entering the compressor section is compressed and delivered into the combustion section where it is mixed with fuel and ignited to generate a high-speed exhaust gas flow. The high-speed exhaust gas flow expands through the turbine section to drive the compressor and the fan section. The compressor section typically includes low and high pressure compressors, and the turbine section includes low and high pressure turbines.

[0003] The high pressure turbine drives the high pressure compressor through an outer shaft to form a high spool, and the low pressure turbine drives the low pressure compressor through an inner shaft to form a low spool. The inner shaft may also drive the fan section. A direct drive gas turbine engine includes a fan section driven by the inner shaft such that the low pressure compressor, low pressure turbine and fan section rotate at a common speed in a common direction.

[0004] A speed reduction device such as an epicyclical gear assembly may be utilized to drive the fan section such that the fan section may rotate at a speed different than the turbine section so as to increase the overall propulsive efficiency of the engine. In such engine architectures, a shaft driven by one of the turbine sections provides an input to the epicyclical gear assembly that drives the fan section at a speed different than the turbine section such that both the turbine section and the fan section can rotate at closer to optimal speeds.

[0005] Although geared architectures have improved propulsive efficiency, turbine engine manufacturers continue to seek further improvements to engine performance including improvements to thermal, transfer and propulsive efficiencies.

SUMMARY

[0006] A gas turbine engine according to an exemplary embodiment of this disclosure, among other possible things includes a fan including a plurality of fan blades rotatable about an engine axis, a compressor section, a combustor in fluid communication with the compressor section, a turbine section in fluid communication with the combustor, the turbine section including a fan drive turbine and a second turbine, wherein the second turbine is disposed forward of the fan drive turbine and the fan drive turbine includes a plurality of fan drive turbine stages with a ratio between the number of fan blades and the number of fan drive turbine stages is greater than about 2.5, and a speed change system driven by the fan drive turbine for rotating the fan about the engine axis, wherein the fan drive turbine has a first exit area and is configured to rotate at a first speed, the second turbine section has a second exit area and is configured to rotate at a second speed, which is faster than the first speed, wherein the turbine section includes a volume defined within an inner periphery and an outer periphery between a leading edge of a most upstream vane to a trailing edge of a most downstream rotating airfoil and is configured to provide a thrust density greater than 1.5 lbf/in3 and less than or equal to 5.5 lbf/in3 at Sea Level Takeoff Thrust.

[0007] In a further embodiment of the foregoing engine, the speed change system comprises a gearbox, and the fan and the fan drive turbine both rotate in a first direction about the engine axis and the second turbine section rotates in a second direction opposite the first direction.

[0008] In a further embodiment of the foregoing engine, the speed change system comprises a gearbox, and the fan, the fan drive turbine, and the second turbine section all rotate in a first direction about the engine axis.

[0009] In a further embodiment of the foregoing engine, the speed change system comprises a gearbox, and wherein the fan and the second turbine both rotate in a first direction about the engine axis and the fan drive turbine rotates in a second direction opposite the first direction.

[0010] In a further embodiment of the foregoing engine, the speed change system comprises a gearbox, and wherein the fan is rotatable in a first direction and the fan drive turbine, and the second turbine section rotate in a second direction opposite the first direction about the engine axis.

[0011] In a further embodiment of the foregoing engine, the speed change system comprises a gear reduction having a gear ratio greater than 2.3.

[0012] In a further embodiment of the foregoing engine, the fan delivers a portion of air into a bypass duct, and a bypass ratio being defined as the portion of air delivered into the bypass duct divided by the amount of air delivered into the compressor section, with the bypass ratio being greater than 6.0.

[0013] In a further embodiment of the foregoing engine, the bypass ratio is greater than 10.0.

[0014] In a further embodiment of the foregoing engine, a fan pressure ratio across the fan is less than 1.5.

[0015] In a further embodiment of the foregoing engine, the fan has 26 or fewer blades.

[0016] In a further embodiment of the foregoing engine, the fan drive turbine section has up to 6 stages.

[0017] In a further embodiment of the foregoing engine, the ratio between the number of fan blades and the number of fan drive turbine stages is less than 8.5.

[0018] In a further embodiment of the foregoing engine, a pressure ratio across the fan drive turbine is greater than about 5:1.

[0019] In a further embodiment of the foregoing engine, the fan drive turbine includes a first aft rotor attached to a first shaft, the second turbine includes a second aft rotor attached to a second shaft, and a first bearing assembly and a second bearing assembly are disposed aft of the combustor, wherein the first bearing assembly is disposed axially aft of a first connection between the first aft rotor and the first shaft, and the second bearing assembly is disposed axially aft of a second connection between the second aft rotor and the second shaft.

[0020] In a further embodiment of the foregoing engine, the fan drive turbine includes a first aft rotor attached to a first shaft, the second turbine includes a second aft rotor attached to a second shaft, and a first bearing assembly and a second bearing assembly are disposed aft of the combustor, wherein the first bearing assembly is disposed axially aft of a first connection between the first aft rotor and the first shaft, and the second bearing assembly is disposed axially forward of a second connection between the second aft rotor and the second shaft.

[0021] In a further embodiment of the foregoing engine, the fan drive turbine includes a first aft rotor attached to a first shaft, the second turbine includes a second aft rotor attached to a second shaft, and a first bearing assembly and a second bearing assembly are disposed aft of the combustor, wherein the first bearing assembly is disposed axially aft of a first connection between the first aft rotor and the first shaft, and the second bearing assembly is disposed within the annular space defined between the first shaft and the second shaft.

[0022] In a further embodiment of the foregoing engine, the fan drive turbine includes a first aft rotor attached to a first shaft, the second turbine includes a second aft rotor attached to a second shaft, and a first bearing assembly and a second bearing assembly are disposed aft of the combustor, wherein the first bearing assembly is disposed axially forward of a first connection between the first aft rotor and the first shaft, and the second bearing assembly is disposed axially aft of a second connection between the second aft rotor and the second shaft.

[0023] In a further embodiment of the foregoing engine, the fan drive turbine is one of three turbine rotors, while the other two of the turbine rotors each drives a compressor rotor.

[0024] In a further embodiment of the foregoing engine, the fan drive turbine drives a compressor rotor.

[0025] In a further embodiment of the foregoing engine, the speed change system is positioned intermediate a compressor rotor driven by the fan drive turbine section and the fan.

[0026] In a further embodiment of the foregoing engine, speed change system is positioned intermediate the fan drive turbine and the compressor rotor driven by the fan drive turbine.

[0027] Although the different examples have the specific components shown in the illustrations, embodiments of this disclosure are not limited to those particular combinations. It is possible to use some of the components or features from one of the examples in combination with features or components from another one of the examples.

[0028] These and other features disclosed herein can be best understood from the following specification and drawings, the following of which is a brief description.

BRIEF DESCRIPTION OF THE DRAWINGS

[0029] FIG. **1** is a schematic view of an example gas turbine engine.

[0030] FIG. **2** is a schematic view indicating relative rotation between sections of an example gas turbine engine.

[0031] FIG. **3** is another schematic view indicating relative rotation between sections of an example gas turbine engine.

[0032] FIG. **4** is another schematic view indicating relative rotation between sections of an example gas turbine engine.

[0033] FIG. **5** is another a schematic view indicating relative rotation between sections of an example gas turbine engine.

[0034] FIG. **6** is a schematic view of a bearing configuration supporting rotation of example high and low spools of the example gas turbine engine.

[0035] FIG. **7** is another schematic view of a bearing configuration supporting rotation of example high and low spools of the example gas turbine engine.

[0036] FIG. **8**A is another schematic view of a bearing configuration supporting rotation of example high and low spools of the example gas turbine engine.

[0037] FIG. **8**B is an enlarged view of the example bearing configuration shown in FIG. **8**A.

[0038] FIG. **9** is another schematic view of a bearing configuration supporting rotation of example high and low spools of the example gas turbine engine.

[0039] FIG. **10** is a schematic view of an example compact turbine section.

[0040] FIG. **11** is a schematic cross-section of example stages for the disclosed example gas turbine engine.

[0041] FIG. **12** is a schematic view an example turbine rotor perpendicular to the axis or rotation.

[0042] FIG. **13** is another embodiment of an example gas turbine engine for use with the present invention.

[0043] FIG. **14** is yet another embodiment of an example gas turbine engine for use with the present invention.

DETAILED DESCRIPTION

[0044] FIG. **1** schematically illustrates an example gas turbine engine **20** that includes a fan section **22**, a compressor section **24**, a combustor section **26** and a turbine section **28**. Alternative engines might include an augmenter section (not shown) among other systems or features. The fan section **22** drives air along a bypass flow path B while the compressor section **24** draws air in along a core flow path C where air is compressed and communicated to a combustor section **26**. In the combustor section **26**, air is mixed with fuel and ignited to generate a high pressure exhaust gas stream that expands through the turbine section **28** where energy is extracted and utilized to drive the fan section **22** and the compressor section **24**.

[0045] Although the disclosed non-limiting embodiment depicts a turbofan gas turbine engine, it should be understood that the concepts described herein are not limited to use with turbofans as the teachings may be applied to other types of turbine engines; for example a turbine engine including a three-spool architecture in which three spools concentrically rotate about a common axis such that a low spool enables a low pressure turbine to drive a fan via a gearbox, an intermediate spool enables an intermediate pressure turbine to drive a first compressor of the compressor section, and a high spool enables a high pressure turbine to drive a high pressure compressor of the compressor section.

[0046] The example engine **20** generally includes a low speed spool **30** and a high speed spool **32** mounted for rotation about an engine central longitudinal axis A relative to an engine static structure **36** via several bearing systems **38**. It should be understood that various bearing systems **38** at various locations may alternatively or additionally be provided.

[0047] The low speed spool **30** generally includes an inner shaft **40** that connects a fan **42** and a low pressure (or first) compressor section **44** to a low pressure (or first) turbine section **46**. The inner shaft **40** drives the fan **42** through a speed change device, such as a geared architecture **48**, to drive the fan **42** at a lower speed than the low speed spool **30**. The high-speed spool **32** includes an outer shaft **50** that interconnects a high pressure (or second) compressor section **52** and a high pressure (or second) turbine section **54**. The inner shaft

40 and the outer shaft **50** are concentric and rotate via the bearing systems **38** about the engine central longitudinal axis A.

[0048] A combustor **56** is arranged between the high pressure compressor **52** and the high pressure turbine **54**. In one example, the high pressure turbine **54** includes at least two stages to provide a double stage high pressure turbine **54**. In another example, the high pressure turbine **54** includes only a single stage. As used herein, a "high pressure" compressor or turbine experiences a higher pressure than a corresponding "low pressure" compressor or turbine.

[0049] The example low pressure turbine **46** has a pressure ratio that is greater than about 5. The pressure ratio of the example low pressure turbine **46** is measured prior to an inlet of the low pressure turbine **46** as related to the pressure measured at the outlet of the low pressure turbine **46** prior to an exhaust nozzle.

[0050] A mid-turbine frame **58** of the engine static structure **36** is arranged generally between the high pressure turbine **54** and the low pressure turbine **46**. The mid-turbine frame **58** further supports bearing systems **38** in the turbine section **28** as well as setting airflow entering the low pressure turbine **46**.

[0051] The core airflow C is compressed by the low pressure compressor **44** then by the high pressure compressor **52** mixed with fuel and ignited in the combustor **56** to produce high speed exhaust gases that are then expanded through the high pressure turbine **54** and low pressure turbine **46**. The mid-turbine frame **58** includes vanes **60**, which are in the core airflow path and function as an inlet guide vane for the low pressure turbine **46**. Utilizing the vane **60** of the mid-turbine frame **58** as the inlet guide vane for low pressure turbine **46** decreases the length of the low pressure turbine **46** without increasing the axial length of the mid-turbine frame **58**. Reducing or eliminating the number of vanes in the low pressure turbine **46** shortens the axial length of the turbine section **28**. Thus, the compactness of the gas turbine engine **20** is increased and a higher power density may be achieved.

[0052] The disclosed gas turbine engine **20** in one example is a high-bypass geared aircraft engine. In a further example, the gas turbine engine **20** includes a bypass ratio greater than about six (6), with an example embodiment being greater than about ten (10). The example geared architecture **48** is an epicyclical gear train, such as a planetary gear system, star gear system or other known gear system, with a gear reduction ratio of greater than about 2.3.

[0053] In one disclosed embodiment, the gas turbine engine **20** includes a bypass ratio greater than about ten (10:1) and the fan diameter is significantly larger than an outer diameter of the low pressure compressor **44**. It should be understood, however, that the above parameters are only exemplary of one embodiment of a gas turbine engine including a geared architecture and that the present disclosure is applicable to other gas turbine engines.

[0054] A significant amount of thrust is provided by the bypass flow B due to the high bypass ratio. The fan section **22** of the engine **20** is designed for a particular flight condition—typically cruise at about 0.8 Mach and about 35,000 feet. The flight condition of 0.8 Mach and 35,000 ft., with the engine at its best cruise fuel consumption relative to the thrust it produces—also known as "bucket cruise Thrust Specific Fuel Consumption ('TSFC')"—is the industry standard parameter of pound-mass (lbm) of fuel per hour being burned divided by pound-force (lbf) of thrust the engine produces at that minimum bucket cruise point.

[0055] "Low fan pressure ratio" is the pressure ratio across the fan blade alone, without a Fan Exit Guide Vane ("FEGV") system. The low fan pressure ratio as disclosed herein according to one non-limiting embodiment is less than about 1.50. In another non-limiting embodiment the low fan pressure ratio is less than about 1.45.

[0056] "Low corrected fan tip speed" is the actual fan tip speed in ft/sec divided by an industry standard temperature correction of $[(\text{Tram } ^\circ \text{R})/518.7)^{0.5}]$. The "Low corrected fan tip speed", as disclosed herein according to one non-limiting embodiment, is less than about 1150 ft/second.

[0057] The example gas turbine engine includes the fan **42** that comprises in one non-limiting embodiment less than about 26 fan blades. In another non-limiting embodiment, the fan section **22** includes less than about 18 fan blades. Moreover, in one disclosed embodiment the low pressure turbine **46** includes no more than about 6 turbine stages schematically indicated at **34**. In another non-limiting example embodiment the low pressure turbine **46** includes about 3 or more turbine stages. A ratio between the number of fan blades **42** and the number of low pressure turbine stages is between about 2.5 and about 8.5. The example low pressure turbine **46** provides the driving power to rotate the fan section **22** and therefore the relationship between the number of turbine stages **34** in the low pressure turbine **46** and the number of blades **42** in the fan section **22** disclose an example gas turbine engine **20** with increased power transfer efficiency.

[0058] Increased power transfer efficiency is provided due in part to the increased use of improved turbine blade materials and manufacturing methods such as directionally solidified castings, and single crystal materials that enable increased turbine speed and a reduced number of stages. Moreover, the example low pressure turbine **46** includes improved turbine disks configurations that further enable desired durability at the higher turbine speeds.

[0059] Referring to FIGS. **2** and **3**, an example disclosed speed change device is an epicyclical gearbox of a planet type, where the input is to the center "sun" gear **62**. Planet gears **64** (only one shown) around the sun gear **62** rotate and are spaced apart by a carrier **68** that rotates in a direction common to the sun gear **62**. A ring gear **66**, which is non-rotatably fixed to the engine static casing **36** (shown in FIG. **1**), contains the entire gear assembly. The fan **42** is attached to and driven by the carrier **68** such that the direction of rotation of the fan **42** is the same as the direction of rotation of the carrier **68** that, in turn, is the same as the direction of rotation of the input sun gear **62**.

[0060] In the following figures nomenclature is utilized to define the relative rotations between the various sections of the gas turbine engine **20**. The fan section is shown with a "+" sign indicating rotation in a first direction. Rotations relative to the fan section **22** of other features of the gas turbine engine are further indicated by the use of either a "+" sign or a "−" sign. The "−" sign indicates a rotation that is counter to that of any component indicated with a "+" sign.

[0061] Moreover, the term fan drive turbine is utilized to indicate the turbine that provides the driving power for rotating the blades **42** of the fan section **22**. Further, the term "second turbine" is utilized to indicate the turbine before the fan drive turbine that is not utilized to drive the fan **42**. In this disclosed example, the fan drive turbine is the low pressure turbine **46**, and the second turbine is the high pressure turbine **54**. However, it should be understood that other turbine section configurations that include more than the shown high and low pressure turbines **54**, **46** are within the contemplation of this disclosure. For example, a three spool engine configuration may include an intermediate turbine (not shown) utilized to drive the fan section **22** and is within the contemplation of this disclosure.

[0062] In one disclosed example embodiment (FIG. **2**) the fan drive turbine is the low pressure turbine **46** and therefore the fan section **22** and low pressure turbine **46** rotate in a common direction as indicated by the common "+" sign indicating rotation of both the fan **42** and the low pressure turbine **46**. Moreover in this example, the high pressure turbine **54** or second turbine rotates in a direction common with the fan drive turbine **46**. In another example shown in FIG. **3**, the high pressure turbine **54** or second turbine rotates in a direction opposite the fan drive turbine (low pressure turbine **46**) and the fan **42**.

[0063] Counter rotating the low pressure compressor **44** and the low pressure turbine **46** relative to the high pressure compressor **52** and the high pressure turbine **54** provides certain efficient aerodynamic conditions in the turbine section **28** as the generated high speed exhaust gas flow moves from the high pressure turbine **54** to the low pressure turbine **46**. The relative rotations in the compressor and turbine sections provide approximately the desired airflow angles between the sections, which improves overall efficiency in the turbine section **28**, and provides a reduction in overall weight of the turbine section **28** by reducing or eliminating airfoils or an entire row of vanes.

[0064] Referring to FIGS. **4** and **5**, another example disclosed speed change device is an epicyclical gearbox referred to as a star type gearbox, where the input is to the center "sun" gear **62**. Star gears **65** (only one shown) around the sun gear **62** rotate in a fixed position around the sun gear and are spaced apart by a carrier **68** that is fixed to a static casing **36** (best shown in FIG. **1**). A ring gear **66** that is free to rotate contains the entire gear assembly. The fan **42** is attached to and driven by the ring gear **66** such that the direction of rotation of the fan **42** is opposite the direction of rotation of the input sun gear **62**. Accordingly, the low pressure compressor **44** and the low pressure turbine **46** rotate in a direction opposite rotation of the fan **42**.

[0065] In one disclosed example embodiment shown in FIG. **4**, the fan drive turbine is the low pressure turbine **46** and therefore the fan **42** rotates in a direction opposite that of the low pressure turbine **46** and the low pressure compressor **44**. Moreover in this example the high spool **32** including the high pressure turbine **54** and the high pressure compressor **52** rotate in a direction counter to the fan **42** and common with the low spool **30** including the low pressure compressor **44** and the fan drive turbine **46**.

[0066] In another example gas turbine engine shown in FIG. **5**, the high pressure or second turbine **54** rotates in a direction common with the fan **42** and counter to the low spool **30** including the low pressure compressor **44** and the fan drive turbine **46**.

[0067] Referring to FIG. **6**, the bearing assemblies near the forward end of the shafts in the engine at locations **70** and **72**, which bearings support rotation of the inner shaft **40** and the outer shaft **50**, counter net thrust forces in a direction parallel to the axis A that are generated by the rearward load of low pressure turbine **46** and the high pressure turbine **54**, minus the high pressure compressor **52** and the low pressure compressor **44**, which also contribute to the thrust forces acting on the corresponding low spool **30** and the high spool **32**.

[0068] In this example embodiment, a first forward bearing assembly **70** is supported on a portion of the static structure schematically shown at **36** and supports a forward end of the inner shaft **40**. The example first forward bearing assembly **70** is a thrust bearing and controls movement of the inner shaft **40** and thereby the low spool **30** in an axial direction. A second forward bearing assembly **72** is supported by the static structure **36** to support rotation of the high spool **32** and substantially prevent movement along in an axial direction of the outer shaft **50**. The first forward bearing assembly **70** is mounted to support the inner shaft **40** at a point forward of a connection **88** of a low pressure compressor rotor **90**. The second forward bearing assembly **72** is mounted forward of a connection referred to as a hub **92** between a high pressure compressor rotor **94** and the outer shaft **50**. A first aft bearing assembly **74** supports the aft portion of the inner shaft **40**. The first aft bearing assembly **74** is a roller bearing and supports rotation, but does not provide resistance to movement of the shaft **40** in the axial direction. Instead, the aft bearing **74** allows the shaft **40** to expand thermally between its location and the bearing **72**. The example first aft bearing assembly **74** is disposed aft of a connection hub **80** between a low pressure turbine rotor **78** and the inner shaft **40**. A second aft bearing assembly **76** supports the aft portion of the outer shaft **50**. The example second aft bearing assembly **76** is a roller bearing and is supported by a corresponding static structure **36** through the mid turbine frame **58** which transfers the radial load of the shaft across the turbine flow path to ground **36**. The second aft bearing assembly **76** supports the outer shaft **50** and thereby the high spool **32** at a point aft of a connection hub **84** between a high pressure turbine rotor **82** and the outer shaft **50**.

[0069] In this disclosed example, the first and second forward bearing assemblies **70**, **72** and the first and second aft bearing assemblies **74**, **76** are supported to the outside of either the corresponding compressor or turbine connection hubs **80**, **88** to provide a straddle support configuration of the corresponding inner shaft **40** and outer shaft **50**. The straddle support of the inner shaft **40** and the outer shaft **50** provide a support and stiffness desired for operation of the gas turbine engine **20**.

[0070] Referring to FIG. **7**, another example shaft support configuration includes the first and second forward bearing assemblies **70**, **72** disposed to support the forward portion of the corresponding inner shaft **40** and outer shaft **50**. The first aft bearing **74** is disposed aft of the connection **80** between the rotor **78** and the inner shaft **40**. The first aft bearing **74** is a roller bearing and supports the inner shaft **40** in a straddle configuration. The straddle configuration can require additional length of the inner shaft **40** and therefore an alternate configuration referred to as an overhung configuration can be utilized. In this example the outer shaft **50** is supported by the second aft bearing assembly **76** that is disposed forward of the connection **84** between the high pressure turbine rotor **82** and the outer shaft **50**. Accordingly, the connection hub **84** of the high pressure turbine rotor **82** to the outer shaft **50** is overhung aft of the bearing assembly **76**. This positioning of the second aft bearing **76** in an overhung orientation potentially provides for a reduced length of the outer shaft **50**.

[0071] Moreover the positioning of the aft bearing **76** may also eliminate the need for other support structures such as the mid turbine frame **58** as both the high pressure turbine **54** is supported at the bearing assembly **76** and the low pressure turbine **46** is supported by the bearing assembly **74**. Optionally the mid turbine frame strut **58** can provide an optional roller bearing **74**A which can be added to reduce vibratory modes of the inner shaft **40**.

[0072] Referring to FIG. **8**A and **8**B, another example shaft support configuration includes the first and second forward bearing assemblies **70**, **72** disposed to support corresponding forward portions of each of the inner shaft **40** and the outer shaft **50**. The first aft bearing **74** provides support of the outer shaft **40** at a location aft of the connection **80** in a straddle mount configuration. In this example, the aft portion of the outer shaft **50** is supported by a roller bearing assembly **86** supported within a space **96** defined between an outer surface of the inner shaft **40** and an inner surface of the outer shaft **50**.

[0073] The roller bearing assembly **86** supports the aft portion of the outer shaft **50** on the inner shaft **40**. The use of the roller bearing assembly **86** to support the outer shaft **50** eliminates the requirements for support structures that lead back to the static structure **36** through the mid turbine frame **58**. Moreover, the example bearing assembly **86** can provide both a reduced shaft length, and support of the outer shaft **50** at a position substantially in axial alignment with the connection hub **84** for the high pressure turbine rotor **82** and the outer shaft **50**. As appreciated, the bearing assembly **86** is positioned aft of the hub **82** and is supported through the rearmost section of shaft **50**. Referring to FIG. **9**, another example shaft support configuration includes the first and second forward bearing assemblies **70**, **72** disposed to support corresponding forward portions of each of the inner shaft **40** and the outer shaft **50**. The first aft bearing assembly **74** is supported at a point along the inner shaft **40** forward of the connection **80** between the low pressure turbine rotor **78** and the inner shaft **40**.

[0074] Positioning of the first aft bearing **74** forward of the connection **80** can be utilized to reduce the overall length of the engine **20**. Moreover, positioning of the first aft bearing assembly **74** forward of the connection **80** provides for support through the mid turbine frame **58** to the static structure **36**. Furthermore, in this example the second aft bearing assembly **76** is deployed in a straddle mount configuration aft of the connection **84** between the outer shaft **50** and the rotor **82**. Accordingly, in this example, both the first and second aft bearing assemblies **74**, **76** share a common support structure to the static outer structure **36**. As appreciated, such a common support feature provides for a less complex engine construction along with reducing the overall length of the engine. Moreover, the reduction or required support structures will reduce overall weight to provide a further improvement in aircraft fuel burn efficiency.

[0075] Referring to FIG. **10**, a portion of the example turbine section **28** is shown and includes the low pressure turbine **46** and the high pressure turbine **54** with the mid turbine frame **58** disposed between an outlet of the high pressure turbine and the low pressure turbine. The mid turbine frame **58** and vane **60** are positioned to be upstream of the first stage **98** of the low pressure turbine **46**. While a single vane **60** is illustrated, it should be understood these would be plural vanes **60** spaced circumferentially. The vane **60** redirects the flow downstream of the high pressure turbine **54** as it approaches the first stage **98** of the low pressure turbine **46**. As can be appreciated, it is desirable to improve efficiency to have flow between the high pressure turbine **54** and the low pressure turbine **46** redirected by the vane **60** such that the flow of expanding gases is aligned as desired when entering the low pressure turbine **46**. Therefore vane **60** may be an actual airfoil with camber and turning, that aligns the airflow as desired into the low pressure turbine **46**.

[0076] By incorporating a true air-turning vane **60** into the mid turbine frame **58**, rather than a streamlined strut and a stator vane row after the strut, the overall length and volume of the combined turbine sections **46**, **54** is reduced because the vane **60** serves several functions including streamlining the mid turbine frame **58**, protecting any static structure and any oil tubes servicing a bearing assembly from exposure to heat, and turning the flow entering the low pressure turbine **46** such that it enters the rotating airfoil **100** at a desired flow angle. Further, by incorporating these features together, the overall assembly and arrangement of the turbine section **28** is reduced in volume.

[0077] The above features achieve a more or less compact turbine section volume relative to the prior art including both high and low pressure turbines **54**, **46**. Moreover, in one example, the materials for forming the low pressure turbine **46** can be improved to provide for a reduced volume. Such materials may include, for example, materials with increased thermal and mechanical capabilities to accommodate potentially increased stresses induced by operating the low pressure turbine **46** at the increased speed. Furthermore, the elevated speeds and increased operating temperatures at the entrance to the low pressure turbine **46** enables the low pressure turbine **46** to transfer a greater amount of energy, more efficiently to drive both a larger diameter fan **42** through the geared architecture **48** and an increase in compressor work performed by the low pressure compressor **44**.

[0078] Alternatively, lower priced materials can be utilized in combination with cooling features that compensate for increased temperatures within the low pressure turbine **46**. In three exemplary embodiments a first rotating blade **100** of the low pressure turbine **46** can be a directionally solidified casting blade, a single crystal casting blade or a hollow, internally cooled blade. The improved material and thermal properties of the example turbine blade material provide for operation at increased temperatures and speeds, that in turn provide increased efficiencies at each stage that thereby provide for use of a reduced number of low pressure turbine stages. The reduced number of low pressure turbine stages in turn provide for an overall turbine volume that is reduced, and that accommodates desired increases in low pressure turbine speed.

[0079] The reduced stages and reduced volume provide improve engine efficiency and aircraft fuel burn because overall weight is less. In addition, as there are fewer blade rows, there are: fewer leakage paths at the tips of the blades; fewer leakage paths at the inner air seals of vanes; and reduced losses through the rotor stages.

[0080] The example disclosed compact turbine section includes a power density, which may be defined as thrust in pounds force (lbf) produced divided by the volume of the entire turbine section **28**. The volume of the turbine section **28** may be defined by an inlet **102** of a first turbine vane **104** in the high pressure turbine **54** to the exit **106** of the last rotating airfoil **108** in the low pressure turbine **46**, and may be expressed in cubic inches. The static thrust at the engine's flat rated Sea Level Takeoff condition divided by a turbine section volume is defined as power density and a greater power density may be desirable for reduced engine weight. The sea level take-off flat-rated static thrust may be defined in pounds-force (lbf), while the volume may be the volume from the annular inlet **102** of the first turbine vane **104** in the high pressure turbine **54** to the annular exit **106** of the downstream end of the last airfoil **108** in the low pressure turbine **46**. The maximum thrust may be Sea Level Takeoff Thrust "SLTO thrust" which is commonly defined as the flat-rated static thrust produced by the turbofan at sea-level.

[0081] The volume V of the turbine section may be best understood from FIG. **10**. As shown, the mid turbine frame **58** is disposed between the high pressure turbine **54**, and the low pressure turbine **46**. The volume V is illustrated by a dashed line, and extends from an inner periphery I to an outer periphery O. The inner periphery is defined by the flow path of rotors, but also by an inner platform flow paths of vanes. The outer periphery is defined by the stator vanes and outer air seal structures along the flowpath. The volume extends from a most upstream end of the vane **104**, typically its leading edge, and to the most downstream edge of the last rotating airfoil **108** in the low pressure turbine section **46**. Typically this will be the trailing edge of the airfoil **108**.

[0082] The power density in the disclosed gas turbine engine is much higher than in the prior art. Eight exemplary engines are shown below which incorporate turbine sections and overall engine drive systems and architectures as set forth in this application, and can be found in Table I as follows:

TABLE 1

Engine	Thrust SLTO (lbf)	Turbine section volume from the Inlet	Thrust/turbine section volume (lbf/in^3)
1	17,000	3,859	4.40
2	23,300	5,330	4.37
3	29,500	6,745	4.37
4	33,000	6,745	4.84
5	96,500	31,086	3.10
6	96,500	62,172	1.55
7	96,500	46,629	2.07
8	37,098	6,745	5.50

[0083] Thus, in example embodiments, the power density would be greater than or equal to about 1.5 lbf/in^3. More narrowly, the power density would be greater than or equal to about 2.0 lbf/in^3. Even more narrowly, the power density would be greater than or equal to about 3.0 lbf/in^3. More narrowly, the power density is greater than or equal to about 4.0 lbf/in^3. Also, in embodiments, the power density is less than or equal to about 5.5 lbf/in^3.

[0084] Engines made with the disclosed architecture, and including turbine sections as set forth in this application, and with modifications within the scope of this disclosure, thus provide very high efficient operation, and increased fuel efficiency and lightweight relative to their thrust capability.

[0085] An exit area **112** is defined at the exit location for the high pressure turbine **54** and an exit area **110** is defined at the outlet **106** of the low pressure turbine **46**. The gear reduction **48** (shown in FIG. **1**) provides for a range of different rotational speeds of the fan drive turbine, which in this example embodiment is the low pressure turbine **46**, and the fan **42** (FIG. **1**). Accordingly, the low pressure turbine **46**, and thereby the low spool **30** including the low pressure compressor **44** may rotate at a very high speed. Low pressure turbine **46** and high pressure turbine **54** operation may be evaluated looking at a performance quantity which is the exit area for the respective turbine section multiplied by its respective speed squared. This performance quantity ("PQ") is defined as:

$$PQ_{ltp}=(A_{lpt}\times V_{lpt}{}^{2}) \qquad \text{Equation 1:}$$

$$PQ_{bpt}=(A_{hpt}\times V_{hpt}{}^{2}) \qquad \text{Equation 2:}$$

[0086] where A_{lpt} is the area **110** of the low pressure turbine **46** at the exit **106**, V_{lpt} is the speed of the low pressure turbine section; A_{hpt} is the area of the high pressure turbine **54** at the exit **114**, and where V_{hpt} is the speed of the high pressure turbine **54**.

[0087] Thus, a ratio of the performance quantity for the low pressure turbine **46** compared to the performance quantify for the high pressure turbine **54** is:

$$(A_{lpt}\times V_{lpt}{}^{2})/(A_{hpt}\times V_{hpt}{}^{2})=PQ_{lpt/}PQ_{hpt} \qquad \text{Equation 3:}$$

[0088] In one turbine embodiment made according to the above design, the areas of the low and high pressure turbines **46**, **54** are 557.9 in^2 and 90.67 in^2, respectively. Further, the speeds of the low and high pressure turbine **46**, **54** are 10179 rpm and 24346 rpm, respectively. Thus, using Equations 1 and 2 above, the performance quantities for the example low and high pressure turbines **46**, **54** are:

$$PQ_{ltp}=(A_{lpt}\times V_{lpt}{}^{2})=(557.9\ \text{in}^2)(10179\ \text{rpm})^2=57805157673.9\ \text{in}^2\ \text{rpm}^2 \qquad \text{Equation 1:}$$

$$PQ_{hpt}=(A_{hpt}\times V_{hpt}{}^{2})=(90.67\ \text{in}^2)(24346\ \text{rpm})^2=53742622009.72\ \text{in}^2\ \text{rpm}^2 \qquad \text{Equation 2:}$$

and using Equation 3 above, the ratio for the low pressure turbine section to the high pressure turbine section is:

$$\text{Ratio}=PQ_{lpt/}PQ_{hpt}=57805157673.9\ \text{in}^2\ \text{rpm}^2/53742622009.72\ \text{in}^2\ \text{rpm}^2=1.075$$

[0089] In another embodiment, the ratio is greater than about 0.5 and in another embodiment the ratio is greater than about 0.8. With $PQ_{ltp/}PQ_{hpt}$ ratios in the 0.5 to 1.5 range, a very efficient overall gas turbine engine is achieved. More narrowly, $PQ_{ltp/}PQ_{hpt}$ ratios of above or equal to about 0.8 provides increased overall gas turbine efficiency. Even more narrowly, $PQ_{ltp/}PQ_{hpt}$ ratios above or equal to 1.0 are even more efficient thermodynamically and from an enable a reduction in weight that improves aircraft fuel burn efficiency. As a result of these $PQ_{ltp/}PQ_{hpt}$ ratios, in particular, the turbine section **28** can be made much smaller than in the prior art, both in diameter and axial length. In addition, the efficiency of the overall engine is greatly increased.

[0090] Referring to FIG. **11**, portions of the low pressure compressor **44** and the low pressure turbine **46** of the low spool **30** are schematically shown and include rotors **116** of the low pressure turbine **46** and rotors **132** of the low pressure compressor **44**. Each of the rotors **116** includes a bore radius **122**, a live disk radius **124** and a bore width **126** in a direction parallel to the axis A. The rotor **116** supports turbine blades **118** that rotate relative to the turbine vanes **120**. The low pressure compressor **44** includes rotors **132** including a bore radius **134**, a live disk radius **136** and a bore width **138**. The rotor **132** supports compressor blades **128** that rotate relative to vanes **130**.

[0091] The bore radius **122** is that radius between an inner most surface of the bore and the axis. The live disk radius **124** is the radial distance from the axis of rotation A and a portion of the rotor supporting airfoil blades. The bore width **126** of the rotor in this example is the greatest width of the rotor and is disposed at a radial distance spaced apart form the axis A determined to provide desired physical performance properties.

[0092] The rotors for each of the low compressor **44** and the low pressure turbine **46** rotate at an increased speed compared to prior art low spool configurations. The geometric shape including the bore radius, live disk radius and the bore width are determined to provide the desired rotor performance in view of the mechanical and thermal stresses selected to be imposed during operation. Referring to FIG. **12**, with continued reference to FIG. **11**, a turbine rotor **116** is shown to further illustrate the relationship between the bore radius **126** and the live disk radius **124**. Moreover, the relationships disclosed are provided within a known range of materials commonly utilized for construction of each of the rotors.

[0093] Accordingly, the increased performance attributes and performance are provided by desirable combinations of the disclosed features of the various components of the described and disclosed gas turbine engine embodiments.

[0094] FIG. **13** shows an embodiment **200**, wherein there is a fan drive turbine **208** driving a shaft **206** to in turn drive a fan rotor **202**. A gear reduction **204** may be positioned between the fan drive turbine **208** and the fan rotor **202**. This gear reduction **204** may be structured and operate like the gear reduction disclosed above. A compressor rotor **210** is driven by an intermediate pressure turbine **212**, and a second stage compressor rotor **214** is driven by a turbine rotor **216**. A combustion section **218** is positioned intermediate the compressor rotor **214** and the turbine section **216**.

[0095] FIG. **14** shows yet another embodiment **300** wherein a fan rotor **302** and a first stage compressor **304** rotate at a common speed. The gear reduction **306** (which may be structured as disclosed above) is intermediate the compressor rotor **304** and a shaft **308** which is driven by a low pressure turbine section.

[0096] The embodiments **200**, **300** of FIG. **13** or **14** may be utilized with the features disclosed above.

[0097] Although an example embodiment has been disclosed, a worker of ordinary skill in this art would recognize that certain modifications would come within the scope of this disclosure. For that reason, the following claims should be studied to determine the scope and content of this disclosure.

What is claimed is:

1. A gas turbine engine comprising:

a fan including a plurality of fan blades rotatable about an engine axis;

a compressor section;

a combustor in fluid communication with the compressor section;

a turbine section in fluid communication with the combustor, the turbine section including a fan drive turbine and a second turbine, wherein the second turbine is disposed forward of the fan drive turbine and the fan drive turbine includes a plurality of fan drive turbine stages with a ratio between the number of fan blades and the number of fan drive turbine stages is greater than about 2.5; and

a speed change system driven by the fan drive turbine for rotating the fan about the engine axis;

wherein the fan drive turbine has a first exit area and is configured to rotate at a first speed, the second turbine section has a second exit area and is configured to rotate at a second speed, which is faster than the first speed,

wherein the turbine section includes a volume defined within an inner periphery and an outer periphery between a leading edge of a most upstream vane to a trailing edge of a most downstream rotating airfoil and is configured to provide a thrust density greater than 1.5 lbf/in3 and less than or equal to 5.5 lbf/in3 at Sea Level Takeoff Thrust.

2. The engine as recited in claim **1**, wherein the speed change system comprises a gearbox, and wherein the fan and the fan drive turbine both rotate in a first direction about the engine axis and the second turbine section rotates in a second direction opposite the first direction.

3. The engine as recited in claim **1**, wherein the speed change system comprises a gearbox, and wherein the fan, the fan drive turbine, and the second turbine section all rotate in a first direction about the engine axis.

4. The engine as recited in claim **1**, wherein the speed change system comprises a gearbox, and wherein the fan and the second turbine both rotate in a first direction about the engine axis and the fan drive turbine rotates in a second direction opposite the first direction.

5. The engine as recited in claim **1**, wherein the speed change system comprises a gearbox, and wherein the fan is rotatable in a first direction and the fan drive turbine, and the second turbine section rotate in a second direction opposite the first direction about the engine axis.

6. The engine as recited in claim **1**, wherein the speed change system comprises a gear reduction having a gear ratio greater than 2.3.

7. The engine as set forth in claim **1**, wherein said fan delivers a portion of air into a bypass duct, and a bypass ratio being defined as the portion of air delivered into the bypass duct divided by the amount of air delivered into the compressor section, with the bypass ratio being greater than 6.0.

8. The engine as set forth in claim **7**, wherein the bypass ratio is greater than 10.0.

9. The engine as set forth in claim **1**, wherein a fan pressure ratio across the fan is less than 1.5.

10. The engine as set forth in claim **1**, wherein said fan has 26 or fewer blades.

11. The engine as set forth in claim **10**, wherein said fan drive turbine section has up to 6 stages.

12. The engine as set forth in claim **1**, wherein the ratio between the number of fan blades and the number of fan drive turbine stages is less than 8.5.

13. The engine as set forth in claim **1**, wherein a pressure ratio across the fan drive turbine is greater than about 5:1.

14. The engine as recited in claim **1**, wherein the fan drive turbine includes a first aft rotor attached to a first shaft, the second turbine includes a second aft rotor attached to a second shaft, and a first bearing assembly and a second bearing assembly are disposed aft of the combustor, wherein the first bearing assembly is disposed axially aft of a first connection between the first aft rotor and the first shaft, and the second bearing assembly is disposed axially aft of a second connection between the second aft rotor and the second shaft.

15. The engine as recited in claim **1**, wherein the fan drive turbine includes a first aft rotor attached to a first shaft, the second turbine includes a second aft rotor attached to a second shaft, and a first bearing assembly and a second bearing assembly are disposed aft of the combustor, wherein the first bearing assembly is disposed axially aft of a first connection between the first aft rotor and the first shaft, and the second bearing assembly is disposed axially forward of a second connection between the second aft rotor and the second shaft.

16. The engine as recited in claim **1**, wherein the fan drive turbine includes a first aft rotor attached to a first shaft, the second turbine includes a second aft rotor attached to a second shaft, and a first bearing assembly and a second bearing assembly are disposed aft of the combustor, wherein the first bearing assembly is disposed axially aft of a first connection between the first aft rotor and the first shaft, and the second bearing assembly is disposed within the annular space defined between the first shaft and the second shaft.

17. The engine as recited in claim **1**, wherein the fan drive turbine includes a first aft rotor attached to a first shaft, the second turbine includes a second aft rotor attached to a second shaft, and a first bearing assembly and a second bearing assembly are disposed aft of the combustor, wherein the first bearing assembly is disposed axially forward of a first connection between the first aft rotor and the first shaft, and the second bearing assembly is disposed axially aft of a second connection between the second aft rotor and the second shaft.

18. The engine as recited in claim **1**, wherein said fan drive turbine is one of three turbine rotors, while the other two of said turbine rotors each drives a compressor rotor.

19. The engine as recited in claim **18**, wherein said fan drive turbine drives a compressor rotor.

20. The engine as recited in claim **19**, wherein said speed change system is positioned intermediate a compressor rotor driven by said fan drive turbine section and said fan.

21. The engine as recited in claim **20**, wherein said speed change system is positioned intermediate said fan drive turbine and said compressor rotor driven by said fan drive turbine.

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