



(51) International Patent Classification:

Not classified

(21) International Application Number:

PCT/US2017/032319

(22) International Filing Date:

12 May 2017 (12.05.2017)

(25) Filing Language:

English

(26) Publication Language:

English

(30) Priority Data:

15/164,726 25 May 2016 (25.05.2016) US

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(81) Designated States (unless otherwise indicated, for every
kind of national protection available): AE, AG, AL, AM,
AO, AT, AU, AZ, BA, BB, BG, BH, BN, BR, BW, BY, BZ,
CA, CH, CL, CN, CO, CR, CU, CZ, DE, DJ, DK, DM, DO,
DZ, EC, EE, EG, ES, FI, GB, GD, GE, GH, GM, GT, HN,
HR, HU, ID, IL, IN, IR, IS, JP, KE, KG, KH, KN, KP, KR,
KW, KZ, LA, LC, LK, LR, LS, LU, LY, MA, MD, ME, MG,
MK, MN, MW, MX, MY, MZ, NA, NG, NI, NO, NZ, OM,
PA, PE, PG, PH, PL, PT, QA, RO, RS, RU, RW, SA, SC,
SD, SE, SG, SK, SL, SM, ST, SV, SY, TH, TJ, TM, TN, TR,
TT, TZ, UA, UG, US, UZ, VC, VN, ZA, ZM, ZW.

(54) Title: METHOD AND SYSTEM FOR A TWO FRAME GAS TURBINE ENGINE

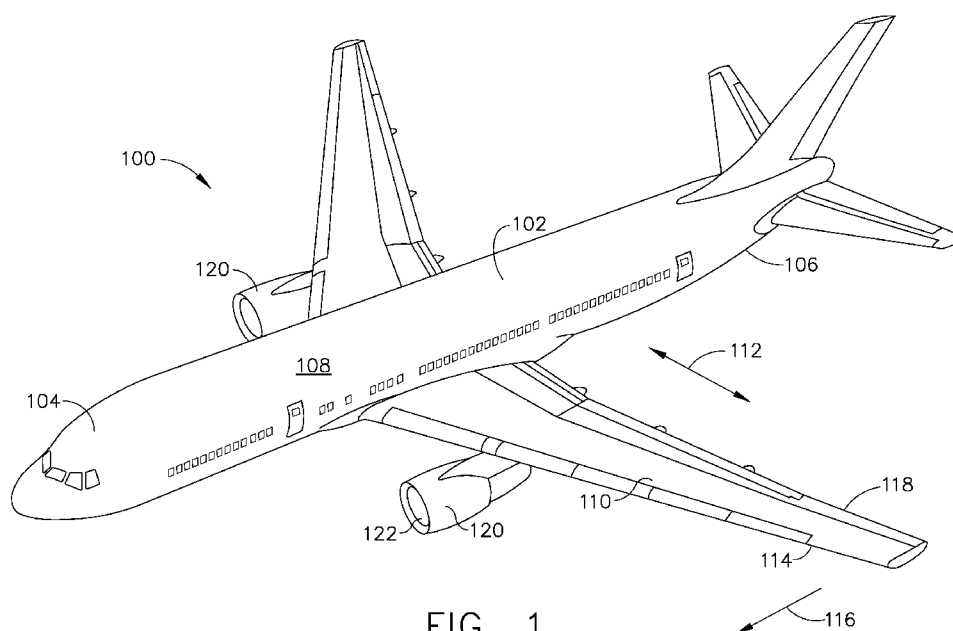


FIG. 1

(57) Abstract: The gas turbine engine includes a core engine including a HP compressor, a combustor, and a HP turbine in a serial flow arrangement. A LP turbine is positioned axially aft of the core engine and includes a plurality of stages of rotor blades. A last stage of rotor blades includes a low swirl rotor blade stage. A LP compressor is positioned axially forward of the core engine and coupled to the LP turbine through a gearbox. The LP compressor is positioned radially outward from the gearbox. A fan assembly is directly coupled to the LP compressor such that the fan assembly and the LP compressor rotate at the same speed. An engine frame assembly includes a forward fan frame member positioned axially between the LP compressor and the HP compressor. The engine frame assembly also includes a turbine center frame member positioned between the HP turbine and the LP turbine.

(84) Designated States (*unless otherwise indicated, for every kind of regional protection available*): ARIPO (BW, GH, GM, KE, LR, LS, MW, MZ, NA, RW, SD, SL, ST, SZ, TZ, UG, ZM, ZW), Eurasian (AM, AZ, BY, KG, KZ, RU, TJ, TM), European (AL, AT, BE, BG, CH, CY, CZ, DE, DK, EE, ES, FI, FR, GB, GR, HR, HU, IE, IS, IT, LT, LU, LV, MC, MK, MT, NL, NO, PL, PT, RO, RS, SE, SI, SK, SM, TR), OAPI (BF, BJ, CF, CG, CI, CM, GA, GN, GQ, GW, KM, ML, MR, NE, SN, TD, TG).

Declarations under Rule 4.17:

- *as to applicant's entitlement to apply for and be granted a patent (Rule 4.17(ii))*
- *as to the applicant's entitlement to claim the priority of the earlier application (Rule 4.17(iii))*

Published:

- *without international search report and to be republished upon receipt of that report (Rule 48.2(g))*

METHOD AND SYSTEM FOR A TWO FRAME GAS TURBINE ENGINE

BACKGROUND

[0001] The field of the disclosure relates generally to gas turbine engines and, more particularly, to a method and system for a reduced frame gas turbine engine assembly.

[0002] Gas turbine engine assemblies using integral drive with high speed booster compressors directly coupled to a low pressure (LP) turbine require a frame between the booster compressor and fan assembly. An additional frame is typically required aft of a LP turbine. These frames tend to increase the length of the gas turbine engine assembly and thereby also tend to increase weight and cost of the gas turbine engine assembly. Moreover, the booster compressor rotating at a high speed, for example, approximately the LP turbine speed, highly loads the booster compressor causing it to operate at a non-optimal pressure ratio than might otherwise be attained.

BRIEF DESCRIPTION

[0003] In one aspect, a gas turbine engine assembly is provided. The gas turbine engine includes a core engine, a low pressure turbine, a low pressure compressor, a fan assembly, and an engine frame assembly. The core engine includes a high pressure compressor, a combustor, and a high pressure (HP) turbine in a serial flow arrangement. The low pressure turbine is positioned axially aft of the core engine and includes a plurality of stages of stator vanes and rotor blades. A last stage of rotor blades of the plurality of stages of stator vanes and rotor blades of the low pressure turbine includes a low swirl outlet rotor blade stage. The low pressure compressor is positioned axially forward of the core engine and rotatably coupled to the low pressure turbine through a gearbox. The low pressure compressor is aligned axially with the gearbox and positioned radially outward from the gearbox. The fan assembly is directly coupled to the low pressure compressor such that the fan assembly and the low pressure compressor rotate at the same speed. The engine frame assembly includes a forward fan frame member positioned axially between the low pressure compressor and the high pressure compressor, and positioned axially aft of the gearbox. The engine frame assembly also includes

a turbine center frame member positioned axially between the high pressure turbine and the low pressure turbine.

[0004] In another aspect, a method of assembling a two-frame gas turbine engine includes providing a core gas turbine engine including a high pressure compressor, a combustor, and a high pressure turbine coupled together in serial flow communication. The method also includes coupling the core gas turbine engine to a forward fan frame member positioned axially forward of the core engine. The method further includes coupling the core gas turbine engine to a turbine center frame member. The turbine center frame member is coupled to the core gas turbine engine axially aft of the high pressure turbine. The method also includes coupling a low pressure turbine to a first shaft axially aft of the turbine center frame member. The method also includes coupling an input of a gearbox to the first shaft axially forward of the forward fan frame member. The method further includes coupling a fan assembly and a low pressure compressor to an output of the gearbox axially forward of the forward fan frame member.

[0005] In yet another aspect, gas turbine engine assembly configured to drive a bladed rotatable member of a fan assembly is provided. The gas turbine engine includes a core engine, a low pressure turbine, a low pressure compressor, and an engine frame assembly. The core engine includes a high pressure compressor, a combustor, and a high pressure turbine in a serial flow arrangement. The low pressure turbine is positioned axially aft of the core engine and includes a plurality of stages of stator vanes and rotor blades. A last stage of rotor blades of the plurality of stages of stator vanes and rotor blades of the low pressure turbine includes a low swirl outlet rotor blade stage. The low pressure compressor is positioned axially forward of the core engine and rotatably coupled to the low pressure turbine through a gearbox. The low pressure compressor is aligned axially with the gearbox. The low pressure compressor is positioned radially outward from the gearbox. The engine frame assembly includes a forward fan frame member positioned axially between the low pressure compressor and the high pressure compressor. The gearbox is positioned axially forward of the forward frame. The gearbox is positioned radially inward.

DRAWINGS

[0006] These and other features, aspects, and advantages of the present disclosure will become better understood when the following detailed description is read with reference to the accompanying drawings in which like characters represent like parts throughout the drawings,

wherein:

[0007] FIG. 1 is a perspective view of an aircraft.

[0008] FIG. 2 is a schematic cross-sectional view of a gas turbine engine in accordance with an exemplary embodiment of the present disclosure that may be used with the aircraft shown in FIG. 1.

[0009] FIG. 3 is a side elevation view of the turbofan engine shown in FIGS. 1 and 2.

[0010] FIG. 4 is a side elevation view of an aft portion of the turbofan engine shown in FIGS. 1, 2, and 3.

[0011] FIG. 5 is a flow diagram of a method of constructing the turbofan engine shown in FIGS. 1, 2, and 3.

[0012] Unless otherwise indicated, the drawings provided herein are meant to illustrate features of embodiments of this disclosure. These features are believed to be applicable in a wide variety of systems comprising one or more embodiments of this disclosure. As such, the drawings are not meant to include all conventional features known by those of ordinary skill in the art to be required for the practice of the embodiments disclosed herein.

DETAILED DESCRIPTION

[0013] In the following specification and the claims, reference will be made to a number of terms, which shall be defined to have the following meanings.

[0014] The singular forms “a,” “an,” and “the” include plural references unless the context clearly dictates otherwise.

[0015] “Optional” or “optionally” means that the subsequently described event or circumstance may or may not occur, and that the description includes instances where the event occurs and instances where it does not.

[0016] Approximating language, as used herein throughout the specification and claims, may be applied to modify any quantitative representation that could permissibly vary without resulting in a change in the basic function to which it is related. Accordingly, a value modified by a term or terms, such as “about,” “approximately,” and “substantially,” are not to be limited to the precise value specified. In at least some instances, the approximating language may correspond to the precision of an instrument for measuring the value. Here and throughout the

specification and claims, range limitations may be combined and/or interchanged; such ranges are identified and include all the sub-ranges contained therein unless context or language indicates otherwise.

[0017] Embodiments of the gas turbine engine and method of assembly described herein provide a cost effective method for providing a gas turbine engine assembly that is shorter and lighter than known engines of similar capability. Gas turbine engine assemblies using integral drive with high speed booster compressors typically require a frame between the booster compressor and the fan assembly. An additional frame is typically required aft of a LP turbine. These frames tend to increase the length of the gas turbine engine assembly and thereby also tend to increase weight and cost of the gas turbine engine assembly. By attaching the booster compressor directly to the fan assembly, a frame can be eliminated. By including a low swirl low pressure (LP) turbine stage, another frame can be eliminated. The gas turbine engine assembly includes a core engine including a high pressure compressor, a combustor, and a high pressure (HP) turbine in a serial flow arrangement. A low swirl LP turbine is positioned axially aft of the core engine and a low pressure LP compressor is positioned axially forward of the core engine. The LP compressor is rotatably coupled to the LP turbine through a gearbox, which may be a speed changing gearbox or a reduction gearbox, and is aligned axially with the gearbox. The LP compressor is positioned radially outward from the gearbox. In various embodiments, the gas turbine engine assembly also includes an engine frame assembly including only two frames, a forward fan frame member and a turbine center frame member. The forward fan frame member is positioned axially between the low pressure compressor and the high pressure compressor and axially aft of the gearbox. The forward fan frame member is configured to support the low pressure compressor and the high pressure compressor. The turbine center frame member is positioned axially between the HP turbine and the LP turbine. In the example embodiment, the gas turbine engine assembly includes a longitudinal centerline and the forward fan frame member and the turbine center frame member are coaxially aligned with the centerline.

[0018] The core engine includes a high pressure rotor shaft and the gas turbine engine assembly includes a low pressure rotor shaft. The turbine rear frame member is configured to rotatably support an aft end portion of the high pressure rotor shaft and an aft end portion of the low pressure rotor shaft. At the forward end, the engine fan assembly is directly coupled

to the low pressure compressor and consequently the fan assembly and the low pressure compressor rotate at the same speed. Because the fan assembly and the low pressure compressor are coupled to the LP turbine through the gearbox, the fan assembly and the low pressure compressor may rotate at a speed that is the same or that is different than a speed of rotation of the LP turbine depending on the configuration of the gearbox. In various embodiments, the fan assembly and the low pressure compressor rotate at a first speed and the LP turbine rotates at a second speed. The first and second speeds can be the same, the first speed can be greater than or less than the second speed depending, in some embodiments, on a configuration of the gearbox.

[0019] A method of assembling a gas turbine engine includes providing a core engine including a high pressure compressor, a combustor, and a turbine coupled together in axial flow communication, coupling a low swirl LP turbine to a first shaft axially aft of the core engine, coupling an input of a gearbox to the first shaft axially forward of the core engine, and coupling a fan assembly and a booster compressor to an output of the gearbox axially forward of the core engine.

[0020] Embodiments described herein disclose a booster compressor to the fan assembly in an integral drive configuration. The boost power is sent through the gearbox from the low swirl LP turbine to the fan and booster as a common spool. Additionally, embodiments described herein disclose including a low swirl low pressure turbine rotor blade stage which eliminates the need for a turbine rear frame or outlet guide vanes to reduce the swirl of exhaust gases. Such a configuration eliminates the need for two frames of the engine and shortens the engine. The engine configuration described herein permits increasing the fan speed such that the booster compressor speed is increased thus reducing loading on the booster compressor and improving pressure ratio possible from the booster compressor. Also increasing fan assembly speed is beneficial because this makes the fan more distortion tolerant or operable. Moreover, improvements in the fan tip speed range combined with lower fan pressure ratio result from the described configuration.

[0021] FIG. 1 is a perspective view of an aircraft 100. In the example embodiment, aircraft 100 includes a fuselage 102 that includes a nose 104, a tail 106, and a hollow, elongate body 108 extending therebetween. Aircraft 100 also includes a wing 110 extending away from fuselage 102 in a lateral direction 112. Wing 110 includes a forward leading edge 114 in a

direction 116 of motion of aircraft 100 during normal flight and an aft trailing edge 118 on an opposing edge of wing 110. Aircraft 100 further includes at least one engine assembly 120, which may be embodied in a gas turbine engine and/or a gas turbine engine of the high bypass turbofan type or the like, configured to drive a bladed rotatable member 122 or fan to generate thrust. Engine assembly 120 is coupled to at least one of wing 110 and fuselage 102, for example, in a pusher configuration (not shown) proximate tail 106.

[0022] FIG. 2 is a schematic cross-sectional view of gas turbine engine assembly 120 in accordance with an exemplary embodiment of the present disclosure. In the example embodiment, gas turbine engine assembly 120 is embodied in a high bypass turbofan jet engine. As shown in FIG. 2, turbofan engine assembly 120 defines an axial direction A (extending parallel to a longitudinal axis 202 provided for reference) and a radial direction R. In general, turbofan 120 includes a fan assembly 204 and a core engine 206 disposed downstream from fan assembly 204.

[0023] In the example embodiment, core engine 206 includes an approximately tubular outer casing 208 that defines an annular inlet 220. Outer casing 208 encases, in serial flow relationship, a compressor section including a booster or low pressure (LP) compressor 222 and a high pressure (HP) compressor 224; a combustion section 226; a turbine section including a high pressure (HP) turbine 228 and a low pressure (LP) turbine 230; and a jet exhaust nozzle section 232. A high pressure (HP) shaft or spool 234 drivingly connects HP turbine 228 to HP compressor 224. A low pressure (LP) shaft or spool 236 drivingly connects LP turbine 230 to LP compressor 222. The compressor section, combustion section 226, turbine section, and nozzle section 232 together define a core air flowpath 237.

[0024] In the example embodiment, fan assembly 204 includes a variable pitch fan 238 having a plurality of fan blades 240 coupled to a disk 242 in a spaced apart relationship. Although a variable pitch fan is shown in FIG. 2, other fan configurations are anticipated including a configuration as shown in FIG. 3 without a variable pitch fan. Fan blades 240 extend radially outwardly from disk 242. Each fan blade 240 is rotatable relative to disk 242 about a pitch axis P by virtue of fan blades 240 being operatively coupled to a suitable pitch change mechanism (PCM) 244 configured to vary the pitch of fan blades 240. In other embodiments, pitch change mechanism (PCM) 244 is configured to collectively vary the pitch of fan blades 240 in unison. Fan blades 240, disk 242, and pitch change mechanism 244 are

together rotatable about longitudinal axis 202 by LP shaft 236 across a power gearbox 246. Power gearbox 246 includes a plurality of gears for adjusting the rotational speed of fan 238 relative to LP shaft 236 to a more efficient rotational fan speed.

[0025] Disk 242 is covered by rotatable front hub 248 aerodynamically contoured to promote an airflow through the plurality of fan blades 240. Additionally, fan assembly 204 includes an annular fan casing or outer nacelle 250 that circumferentially surrounds fan 238 and/or at least a portion of core engine 206. In the example embodiment, nacelle 250 is configured to be supported relative to core engine 206 by a plurality of circumferentially-spaced outlet guide vanes 252 coupled to a forward fan frame member 259. Moreover, a downstream section 254 of nacelle 250 may extend over an outer portion of core engine 206 so as to define a bypass airflow passage 256 therebetween. Gas turbine engine assembly 120 includes an engine frame assembly 257 including, in one embodiment, only two frames, forward fan frame member 259 and a turbine center frame member 261. As used herein, a frame member supports a bearing and may incorporate an aerodynamic fairing to swirl or de-swirl the air through gas turbine engine assembly 120 during operation. In various other embodiments, turbine rear frame member 255 is positioned aft of the LP turbine. Forward fan frame member 259 is positioned axially between low pressure compressor 222 and high pressure compressor 224 and axially aft of gearbox 246. Forward fan frame member 259 is configured to support LP compressor 222 and HP compressor 224. Turbine center frame member 261 is positioned axially between HP turbine 228 and LP turbine 230. In the example embodiment, gas turbine engine assembly 120 includes a longitudinal axis 202 and forward fan frame member 259 and turbine center frame member 261 are coaxially aligned with the centerline. In various embodiments, turbine rear frame member 255 is added to provide additional support to LP turbine 230. Accordingly, gas turbine engine assembly 120 includes a three frame engine frame assembly in some embodiments.

[0026] Core engine 206 includes a high pressure rotor shaft 234 and gas turbine engine assembly 120 includes a low pressure rotor shaft 236. Turbine center frame member 261 is configured to rotatably support an aft end portion 239 of HP turbine 228 and a forward end portion 241 of LP turbine 230.

[0027] During operation of turbofan engine assembly 120, a volume of air 258 enters turbofan 120 through an associated inlet 260 of nacelle 250 and/or fan assembly 204. As

volume of air 258 passes across fan blades 240, a first portion 262 of volume of air 258 is directed or routed into bypass airflow passage 256 and a second portion 264 of volume of air 258 is directed or routed into core air flowpath 237, or more specifically into LP compressor 222. A ratio between first portion 262 and second portion 264 is commonly referred to as a bypass ratio. The pressure of second portion 264 is then increased as it is routed through high pressure (HP) compressor 224 and into combustion section 226, where it is mixed with fuel and burned to provide combustion gases 266.

[0028] Combustion gases 266 are routed through HP turbine 228 where a portion of thermal and/or kinetic energy from combustion gases 266 is extracted via sequential stages of HP turbine stator vanes 268 that are coupled to outer casing 208 and HP turbine rotor blades 270 that are coupled to HP shaft or spool 234, thus causing HP shaft or spool 234 to rotate, which then drives a rotation of HP compressor 224. Combustion gases 266 are then routed through LP turbine 230 where a second portion of thermal and kinetic energy is extracted from combustion gases 266 via sequential stages of LP turbine stator vanes 272 that are coupled to outer casing 208 and LP turbine rotor blades 274 that are coupled to LP shaft or spool 236, which drives a rotation of LP shaft or spool 236 and LP compressor 222 and/or rotation of fan 238.

[0029] Combustion gases 266 are subsequently routed through jet exhaust nozzle section 232 of core engine 206 to provide propulsive thrust. Simultaneously, the pressure of first portion 262 is substantially increased as first portion 262 is routed through bypass airflow passage 256 before it is exhausted from a fan nozzle exhaust section 276 of turbofan 120, also providing propulsive thrust. HP turbine 228, LP turbine 230, and jet exhaust nozzle section 232 at least partially define a hot gas path 278 for routing combustion gases 266 through core engine 206.

[0030] Turbofan engine assembly 120 is depicted in FIG. 1 by way of example only, and that in other exemplary embodiments, turbofan engine assembly 120 may have any other suitable configuration including for example, a turboprop engine.

[0031] FIG. 3 is another schematic cross-sectional view of the turbofan engine assembly 120 (shown in FIGS. 1 and 2). In the example embodiment, gearbox 246 is positioned axially aligned and radially inward of LP compressor 222. Forward fan frame member 259 is positioned axially between LP compressor 222 and HP compressor 224. Such relative positions permits eliminating a frame member that is typically found in other gas turbine

engines of similar size and configuration. In the example embodiment, epicyclic gear train 246 is embodied in, for example, an epicyclic gear and a compound gear. Forward fan frame member 259 provides support for fan assembly 204, LP compressor 222, gearbox 246, and a forward end portion 247 of HP compressor 224.

[0032] In some embodiments, turbine center frame member 261 supports aft end portion 239 of HP turbine 228 and a forward end portion 241 of LP turbine 230. In various embodiments, turbine rear frame member 255 solely supports an aft end portion 243 of LP turbine 230. Accordingly, in some embodiments, turbofan engine assembly 120 includes three frames wherein turbine rear frame member includes an airfoil portion configured to deswirl exhaust gases exiting LP turbine 230. In other embodiments, turbofan engine assembly 120 includes only two frames, forward fan frame member 259 and turbine center frame member 261. Because the two-frame embodiment of turbofan engine assembly 120 does not include aft frame aft 255, the deswirling function typically provided by aft frame aft 255 is provided for elsewhere, for example, by the addition of a stage to LP turbine 230. This final stage is configured to deswirl the exhaust gases channeled from the preceding stages of LP turbine 230. In the exemplary embodiment, LP turbine 230 includes a low swirl LP turbine last stage configured to deswirl the exhaust gases.

[0033] FIG. 4 is a schematic cross-sectional view of an aft portion of gas turbine engine 120 in accordance with an exemplary embodiment of the present disclosure. LP turbine 230 includes four stages of LP turbine rotor blades 402, 404, 406, and 408 coupled to LP shaft 236 and four stages of LP turbine stator vanes 409, 410, 412, and 414. In alternative embodiments, LP turbine 230 may include more or fewer stages of LP turbine rotor blades, such as one, two, three, or five LP turbine rotor blades, or any other suitable number of LP turbine rotor blades that enables LP turbine 230 to function as described herein. In alternative embodiments, LP turbine 230 may include more or fewer stages of LP turbine stator vanes, such as one, two, three, or five LP turbine stator vanes, or any other suitable number of LP turbine stator vanes that enables LP turbine 230 to function as described herein. During operation, combustion gases 266 are routed sequentially to a first LP turbine stator vane 409, a first LP turbine rotor blade stage 402, a second LP turbine stator vane 410, a second LP turbine rotor blade stage 404, a third LP turbine stator vane 412, a third LP turbine rotor blade stage 406, a fourth LP turbine stator vane 414, and a fourth LP turbine rotor blade stage 408.

[0034] Combustion gases 266 include LP turbine stator vane velocities 415, 418, 422, and 426 and LP turbine rotor blade velocities 416, 420, 424, and 428. LP turbine stator vane velocities 418, 422, and 426 and LP turbine rotor blade velocities 416, 420, 424, and 428 each include an axial component and a circumferential component. LP turbine stator vane velocities 415, 418, 422, and 426 projects combustion gases 266 to LP turbine rotor blade velocities 416, 420, 424, and 428 such that the circumferential components of LP turbine rotor blade velocities 416, 420, 424, and 428 are reduced. The circumferential component of LP turbine rotor blade velocities 428 is reduced such that the direction of LP turbine rotor blade velocities 428 deviates from axial direction A by less than or equal to ten degrees at the point combustion gases 266 exit fourth LP turbine rotor blade stage 408. Fourth LP turbine rotor blade stage 408 is a low swirl LP turbine stage which does not require an outlet guide vane or turning vanes to reduce the swirl of exhaust gases.

[0035] FIG. 5 is a flow diagram of a method 500 of constructing a gas turbine engine, such as, gas turbine engine 120 (shown in FIG. 1). Method 500 includes providing 502 core turbine engine 206 including HP compressor 224, combustion section 226, and HP turbine 228 coupled together in serial flow communication. Method 500 also includes coupling 504 core turbine engine 206 to forward fan frame member 259 positioned axially forward of core engine 206. Method 500 further includes coupling 506 core turbine engine 206 to turbine center frame member 261. Turbine center frame member 261 is coupled to core turbine engine 206 axially aft of HP turbine 228. Method 500 also includes coupling 508 LP turbine 230 to LP shaft 236 axially aft of turbine center frame member 261. Method 500 further includes coupling 510 an input of power gearbox 246 to LP shaft 236 axially forward of forward fan frame member 259. Method 500 also includes coupling 512 fan assembly 206 and LP compressor 222 to an output of power gearbox 246 axially forward of forward fan frame member 259.

[0036] The above described embodiments of a method and system of a reduced frame gas turbine engine assembly provides a cost effective and reliable means for reducing the length, weight, and cost of the gas turbine engine assembly. More specifically, the methods and systems described herein facilitate optimizing the fan and booster speed independent of the LP turbine speed to allow an optimized pressure ratio and performance from the fan and booster. Also increasing fan assembly speed is beneficial to make the fan more distortion tolerant or operable. Moreover, improvements in the fan tip speed range combined with lower fan

pressure ratio result from the described configuration. As a result, the methods and systems described herein facilitate improving the fan tip speed range and permitting a lower fan pressure ratio in a shorter, lighter engine in a cost effective and reliable manner.

[0037] Although specific features of various embodiments of the disclosure may be shown in some drawings and not in others, this is for convenience only. In accordance with the principles of the disclosure, any feature of a drawing may be referenced and/or claimed in combination with any feature of any other drawing.

[0038] This written description uses examples to disclose the embodiments, including the best mode, and also to enable any person skilled in the art to practice the embodiments, including making and using any devices or systems and performing any incorporated methods. The patentable scope of the disclosure is defined by the claims, and may include other examples that occur to those skilled in the art. Such other examples are intended to be within the scope of the claims if they have structural elements that do not differ from the literal language of the claims, or if they include equivalent structural elements with insubstantial differences from the literal language of the claims.

WHAT IS CLAIMED IS:

1. A gas turbine engine assembly comprising:
 - a core engine comprising a high pressure compressor, a combustor, and a high pressure turbine in a serial flow arrangement;
 - a low pressure turbine positioned axially aft of said core engine and comprising a plurality of stages of stator vanes and rotor blades;
 - a low pressure compressor positioned axially forward of said core engine and rotatably coupled to said low pressure turbine through a gearbox, said low pressure compressor aligned axially with the gearbox and positioned radially outward from said gearbox;
 - a fan assembly directly coupled to said low pressure compressor such that said fan assembly and said low pressure compressor rotate at a same speed; and
 - an engine frame assembly comprising:
 - a forward fan frame member positioned axially between said low pressure compressor and said high pressure compressor, and positioned axially aft of the gearbox; and
 - a turbine center frame member positioned axially between said high pressure turbine and said low pressure turbine.
2. The gas turbine engine assembly of Claim 1, wherein said gas turbine engine assembly comprising a longitudinal centerline, said forward fan frame member and said turbine center frame member are coaxially aligned with said centerline.
3. The gas turbine engine assembly of Claim 1, wherein said core engine comprises a high pressure rotor shaft and further comprises a low pressure rotor shaft, said turbine center frame member is configured to rotatably support an aft end portion of said high pressure rotor shaft and an aft end portion of said low pressure rotor shaft.
4. The gas turbine engine assembly of Claim 1, wherein said forward fan frame member is configured to support said low pressure compressor.
5. The gas turbine engine assembly of Claim 1, wherein said engine frame assembly comprises only two frames.

6. The gas turbine engine assembly of Claim 4, wherein said fan assembly and said low pressure compressor rotate at a first speed, said low pressure turbine rotates at a second speed, said first speed less than said second speed.

7. The gas turbine engine assembly of Claim 1, wherein said gas turbine engine assembly comprises a longitudinal centerline, a last stage of rotor blades of said plurality of stages of stator vanes and rotor blades of said low pressure turbine comprises a low swirl outlet rotor blade stage, wherein said low swirl outlet rotor blade stage directs said exhaust stream substantially parallel to said longitudinal centerline.

8. The gas turbine engine assembly of Claim 7, wherein a last stator vane of said plurality of stator vanes and rotor blades of said low pressure turbine directs said exhaust stream to said low swirl outlet rotor blades with a first velocity, said first velocity comprises a first axial component and a first circumferential component, said low swirl outlet rotor blades directs said exhaust stream with a second velocity, said second velocity comprises a second axial component and a second circumferential component, wherein said first circumferential component is greater than said second circumferential component.

9. The gas turbine engine assembly of Claim 8, wherein said second velocity deviates from said longitudinal centerline by less than or equal to ten degrees.

10. The gas turbine engine assembly of Claim 9, wherein said low swirl outlet stage does not include a plurality of outlet guide vane or turning vanes configured to reduce the swirl from the exhaust stream.

11. A method of assembling a two-frame gas turbine engine, the method comprising:

providing a core gas turbine engine including a high pressure compressor, a combustor, and a high pressure turbine coupled together in serial flow communication;

coupling the core gas turbine engine to a forward fan frame member positioned axially forward of the core engine;

coupling the core gas turbine engine to a turbine center frame member, the turbine center frame member coupled to the core gas turbine engine axially aft of the high pressure turbine;

coupling a low pressure turbine to a first shaft axially aft of the turbine center frame member;

coupling an input of a gearbox to the first shaft axially forward of the forward fan frame member; and

coupling a fan assembly and a low pressure compressor to an output of the gearbox axially forward of the forward fan frame member.

12. The method of Claim 11, wherein coupling a low pressure turbine to a first shaft axially aft of said turbine center frame member comprises coupling a low pressure turbine to a first shaft axially aft of said turbine center frame member, said low pressure turbine comprises a plurality of stages, wherein a last stage of said plurality of stages comprises a low swirl outlet stage.

13. The method of Claim 11, wherein coupling a low pressure turbine to a first shaft axially aft of said turbine center frame member comprises coupling a low pressure turbine to a first shaft axially aft of said turbine center frame member, said low pressure turbine is unsupported on an at end portion of said low pressure turbine.

14. The method of Claim 11, wherein coupling a low pressure turbine to a first shaft axially aft of said turbine center frame member comprises coupling a low pressure turbine to a first shaft axially aft of said turbine center frame member, said low pressure turbine generates a low swirl exhaust stream without outlet guide vanes.

15. A gas turbine engine assembly configured to drive a bladed rotatable member of a fan assembly, said engine assembly comprising:

a core engine comprising a high pressure compressor, a combustor, and a high pressure turbine in a serial flow arrangement;

a low pressure turbine positioned axially aft of said core engine and comprising a plurality of stages of stator vanes and rotor blades, a last stage of rotor blades of said plurality of stages of stator vanes and rotor blades of said low pressure turbine comprises a low swirl outlet rotor blade stage;

a low pressure compressor positioned axially forward of said core engine and rotatably coupled to said low pressure turbine through a gearbox, said low pressure compressor aligned

axially with said gearbox, said low pressure compressor positioned radially outward from the gearbox; and

an engine frame assembly comprising a forward fan frame member positioned axially between said low pressure compressor and said high pressure compressor, said gearbox positioned axially forward of said forward frame, said gearbox positioned radially inward.

16. The gas turbine engine assembly of Claim 15, wherein said gas turbine engine assembly further comprises a turbine center frame member positioned between said high pressure turbine and said low pressure turbine, said forward fan frame member and said turbine center frame member coaxially aligned with respect to said longitudinal centerline.

17. The gas turbine engine assembly of Claim 15, wherein said core engine comprises a high pressure rotor shaft and further comprises a low pressure rotor shaft, said turbine center frame member is configured to rotatably support an aft end portion of said high pressure rotor shaft and an aft end portion of said low pressure rotor shaft.

18. The gas turbine engine assembly of Claim 15, wherein said gas turbine engine assembly comprises a longitudinal centerline, wherein said low swirl outlet rotor blade stage directs said exhaust stream substantially parallel to said longitudinal centerline.

19. The gas turbine engine assembly of Claim 18, wherein a last stator vane of said plurality of stator vanes and rotor blades of said low pressure turbine directs said exhaust stream to said low swirl outlet rotor blades with a first velocity, said first velocity comprises a first axial component and a first circumferential component, said low swirl outlet rotor blades directs said exhaust stream with a second velocity, said second velocity comprises a second axial component and a second circumferential component, wherein said first circumferential component is greater than said second circumferential component.

20. The gas turbine engine assembly of Claim 19, wherein said second velocity deviates from said longitudinal centerline by less than or equal to ten degrees.

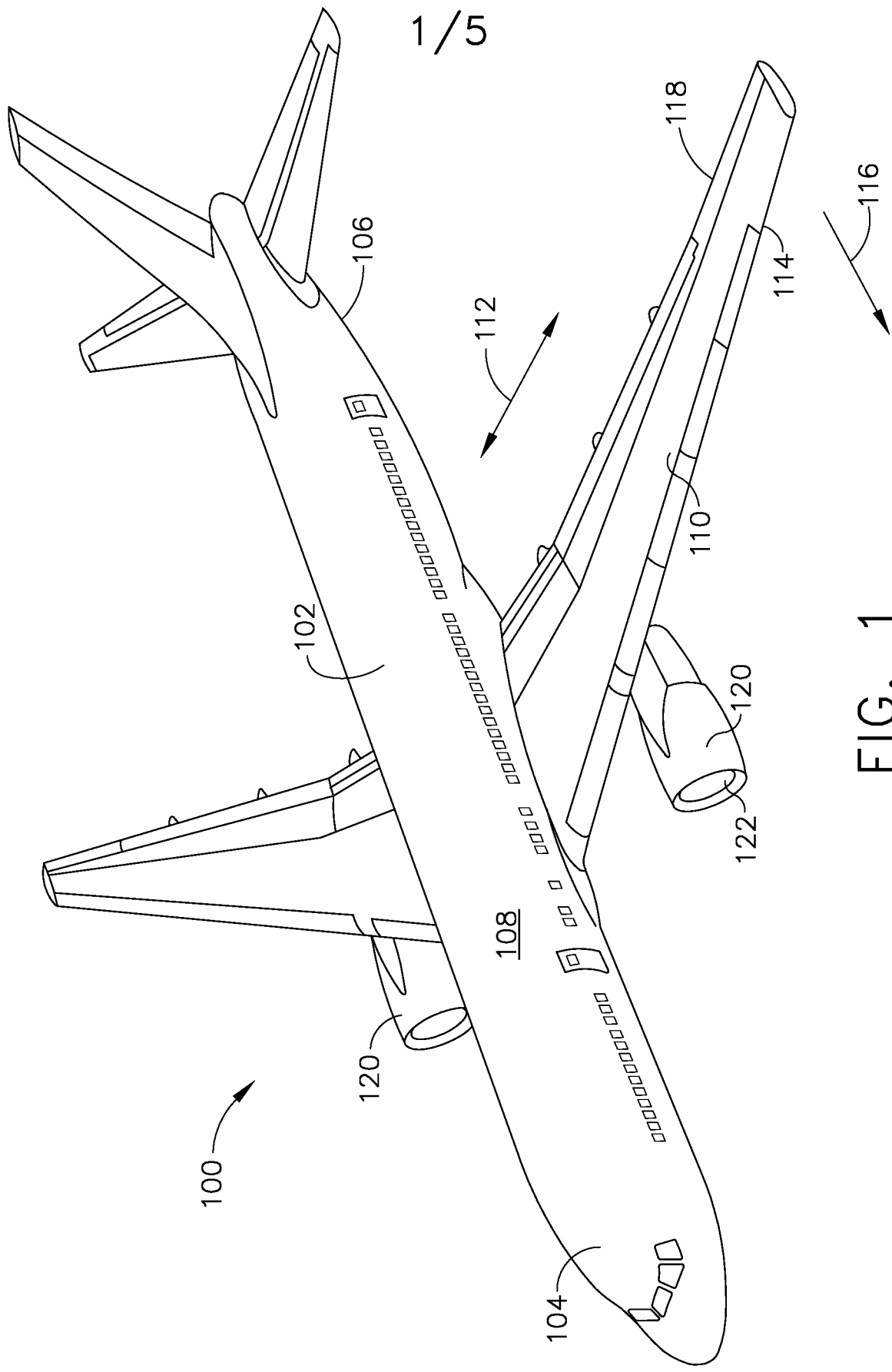


FIG. 1

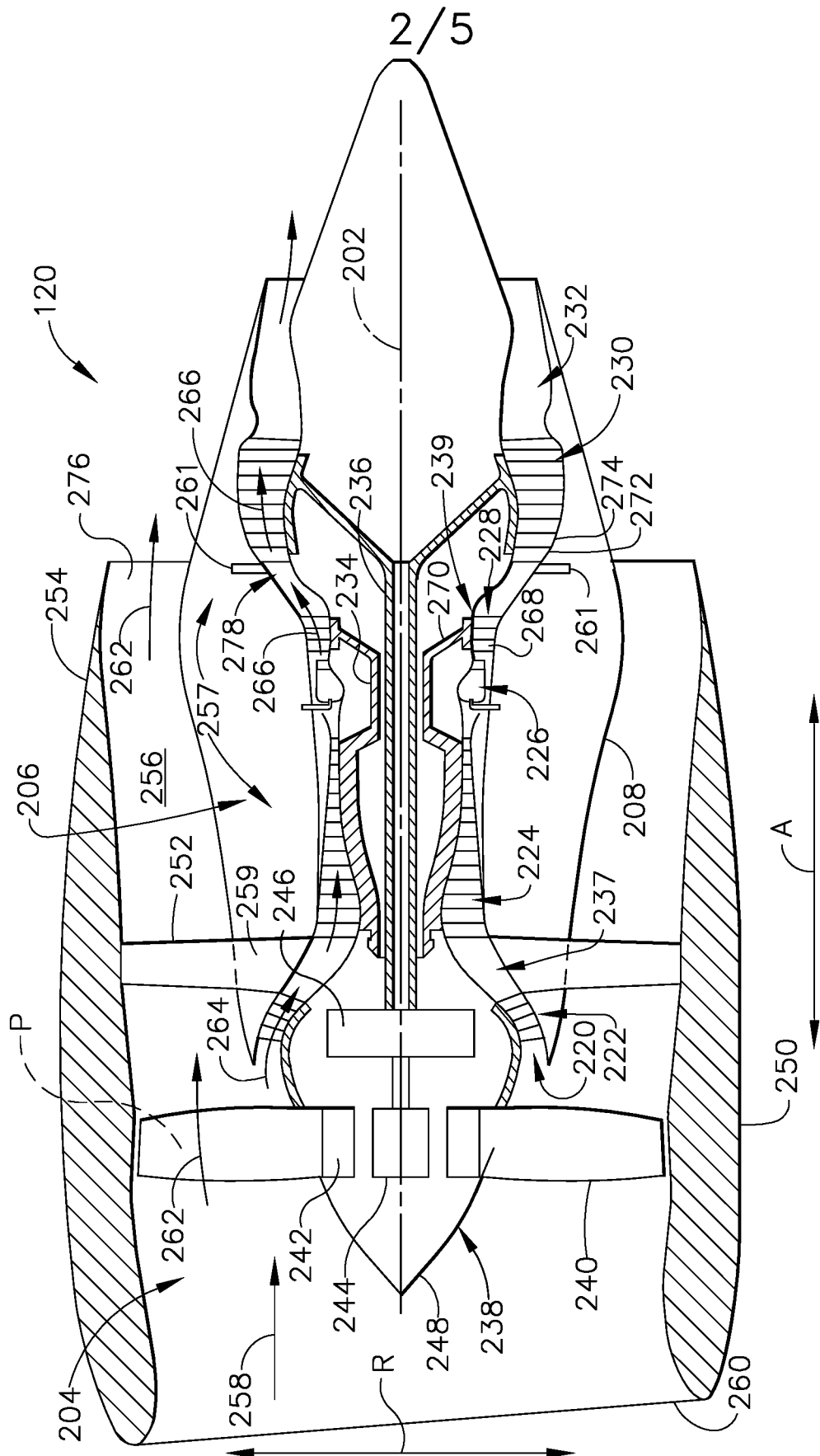


FIG. 2

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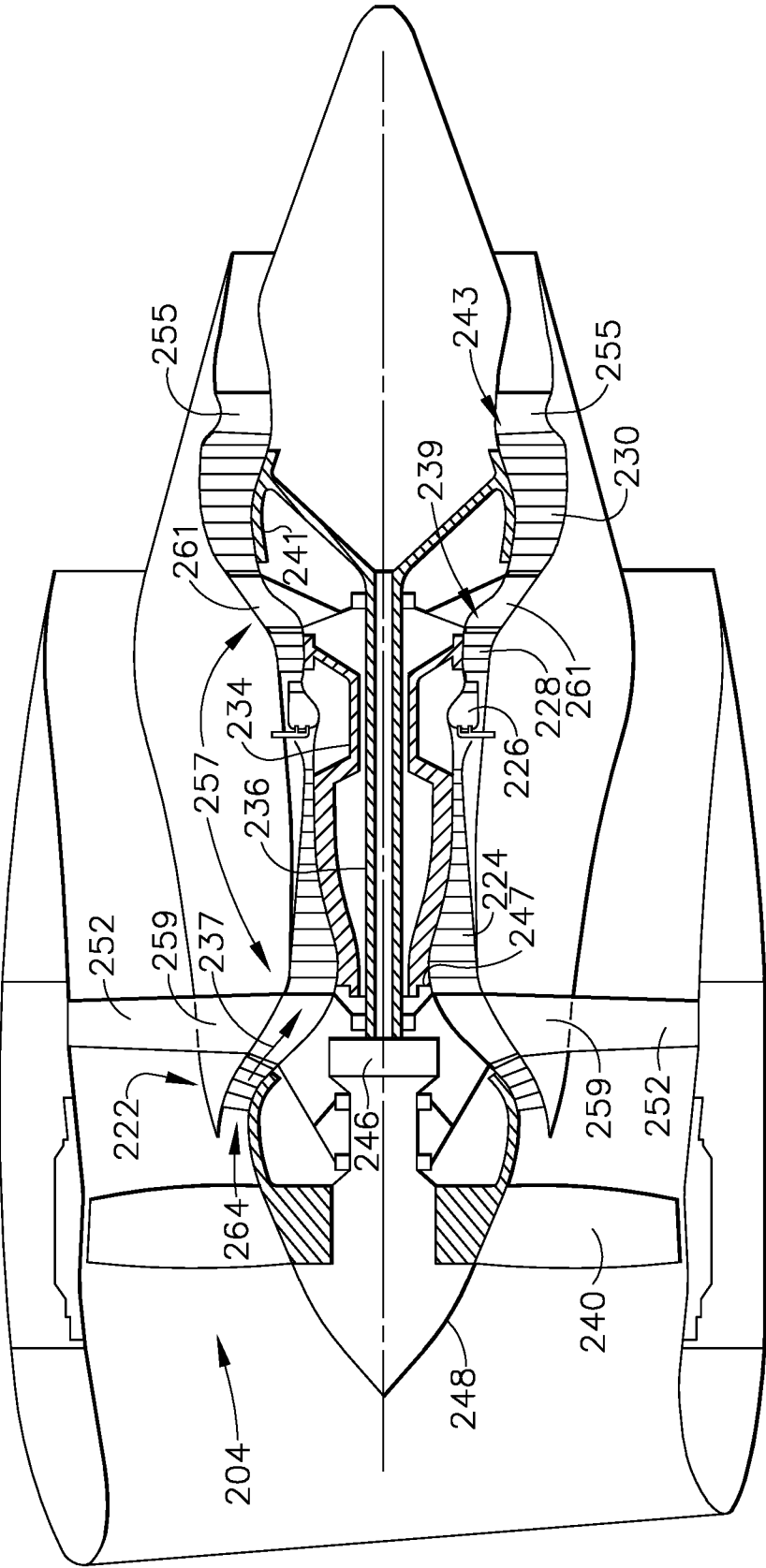


FIG. 3

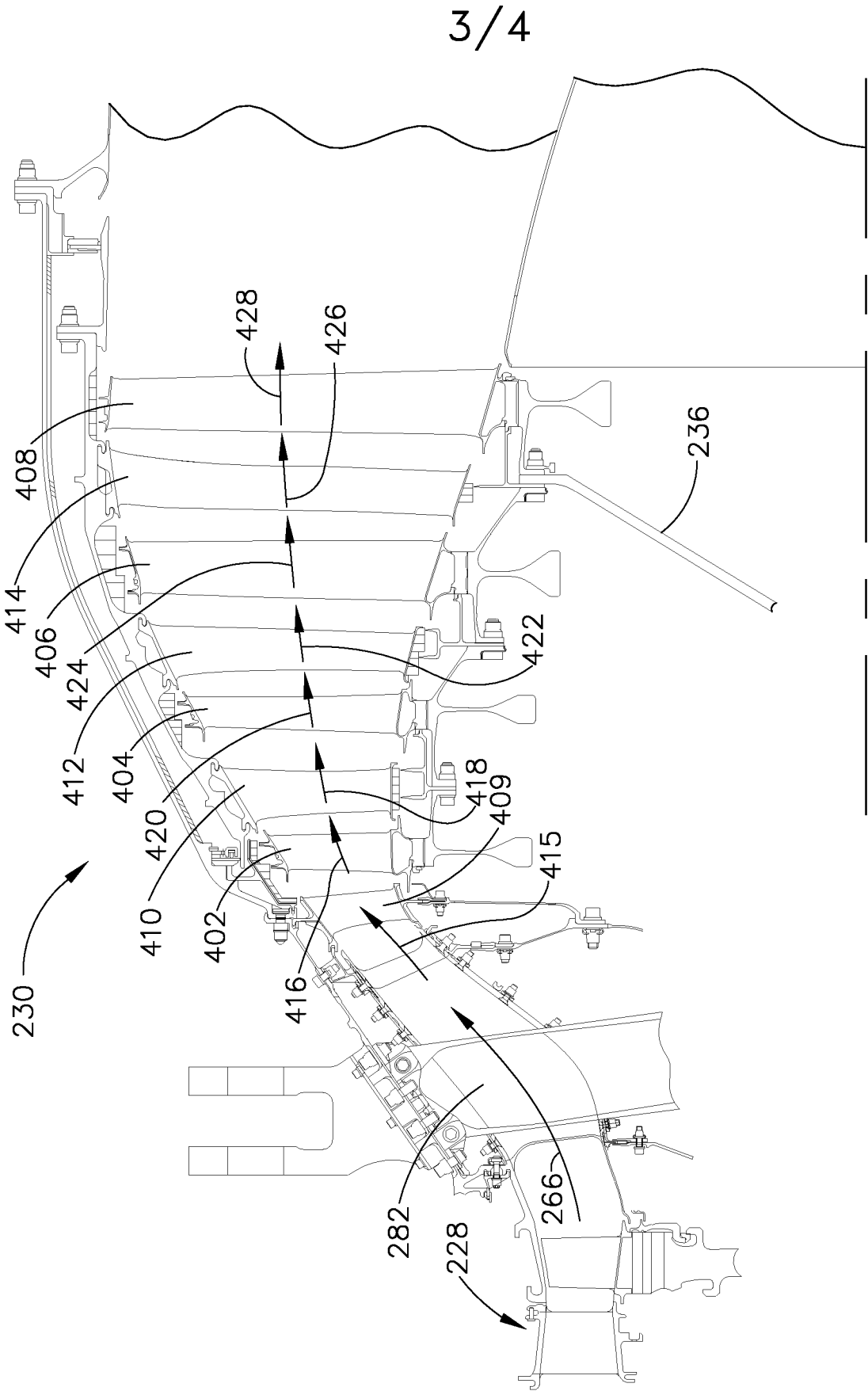


FIG. 4

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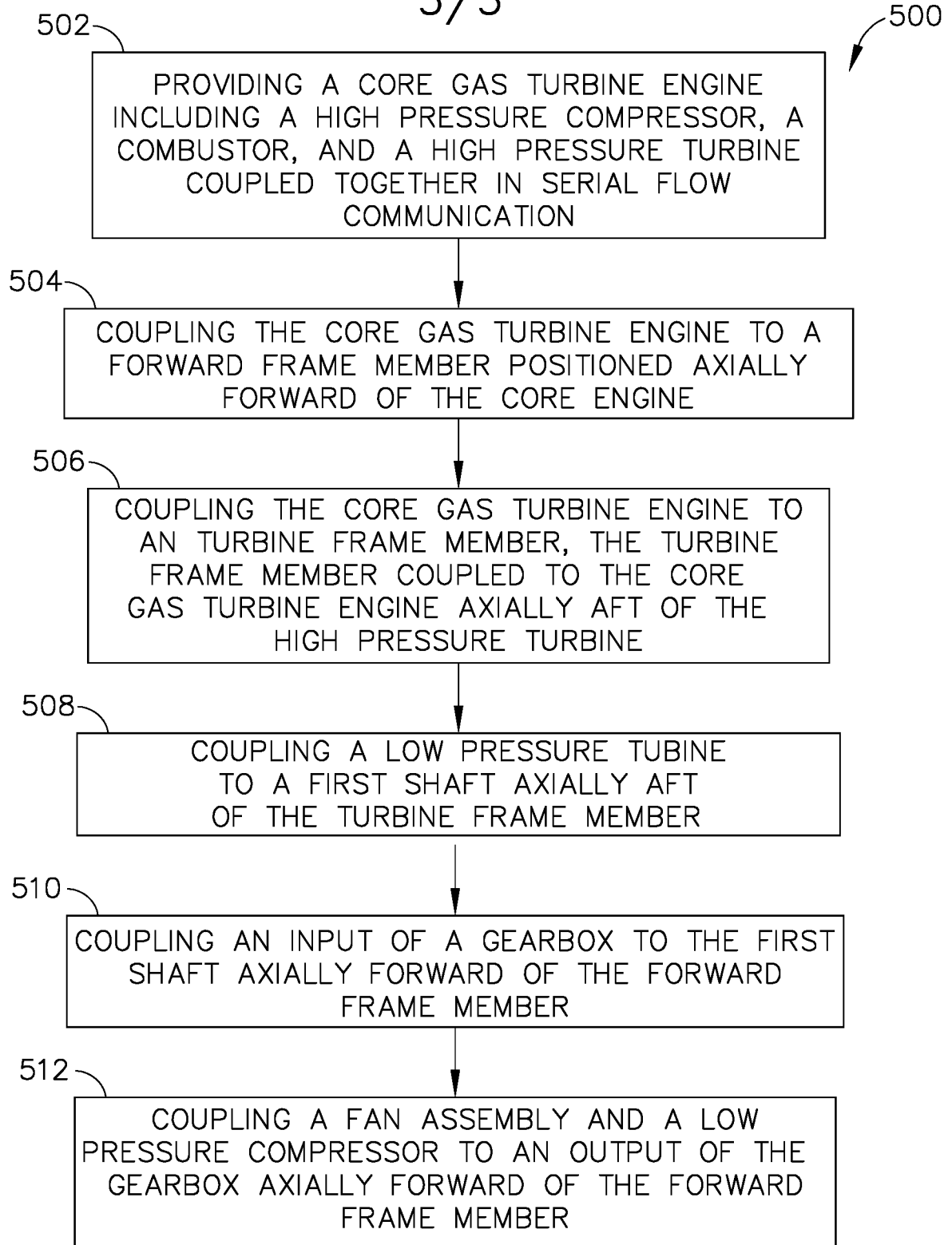


FIG. 5