US00RE49382E

(19) **United States**
(12) **Reissued Patent**
Virkler

(10) **Patent Number: US RE49,382 E**
(45) **Date of Reissued Patent: Jan. 24, 2023**

(54) **HIGH PRESSURE ROTOR DISK**

(71) Applicant: **Raytheon Technologies Corporation**, Farmington, CT (US)

(72) Inventor: **Scott D. Virkler**, Ellington, CT (US)

(73) Assignee: **Raytheon Technologies Corporation**, Farmington, CT (US)

(21) Appl. No.: **17/011,472**

(22) Filed: **Sep. 3, 2020**

Related U.S. Patent Documents

Reissue of:

(64) Patent No.: **10,119,400**
Issued: **Nov. 6, 2018**
Appl. No.: **13/713,257**
Filed: **Dec. 13, 2012**

U.S. Applications:

(60) Provisional application No. 61/707,009, filed on Sep. 28, 2012.

(51) **Int. Cl.**
F01D 5/02 (2006.01)
F01D 5/30 (2006.01)

(52) **U.S. Cl.**
CPC ***F01D 5/02*** (2013.01); *F01D 5/3007* (2013.01); *F05D 2200/00* (2013.01); *F05D 2240/20* (2013.01); *F05D 2240/24* (2013.01)

(58) **Field of Classification Search**
CPC F01D 5/00; F01D 5/02; F01D 5/30; F01D 5/3007; F05D 2240/00; F05D 2240/20; F05D 2240/24; F05D 2240/242; F05D 2200/00; F05D 2200/14; F05D 2200/30
See application file for complete search history.

(56) **References Cited**

U.S. PATENT DOCUMENTS

2,080,425	A	*	5/1937	Lysholm	F01D 25/246 415/138
4,784,572	A		11/1988	Novotny et al.	
4,836,750	A		6/1989	Modafferi et al.	
4,844,694	A	*	7/1989	Naudet	F16B 35/041 415/199.5
4,850,187	A	*	7/1989	Siga et al.	C22C 38/44 60/39.37
5,067,876	A	*	11/1991	Moreman, III	F01D 5/3007 416/219 R
5,292,385	A		3/1994	Kington	
5,632,600	A		5/1997	Hull	

(Continued)

FOREIGN PATENT DOCUMENTS

EP	1058772 B1	10/2002
EP	1424465 A1	6/2004

(Continued)

OTHER PUBLICATIONS

English translation of JP-S61234207-A retrieved from EPO's Espacenet website.*

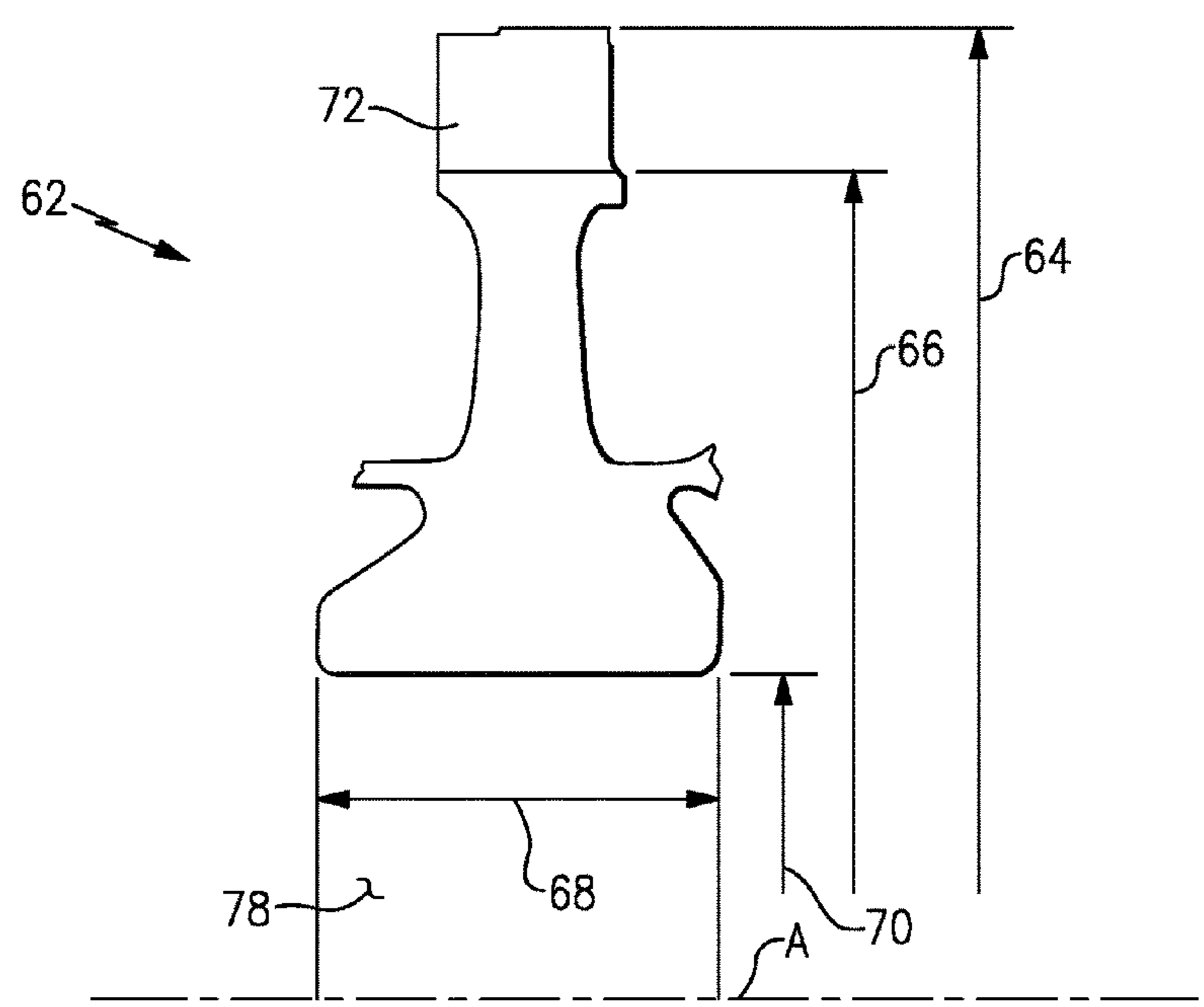
(Continued)

Primary Examiner — Peter C English
(74) *Attorney, Agent, or Firm* — Carlson, Gaskey & Olds, P.C.

(57) **ABSTRACT**

A rotor disk for a gas turbine engine is disclosed and formed to enable operation at high rotational speeds in a high temperature environment. The rotor disk is formed to include a bore, a live rim diameter and an outer diameter related to each other according to defined relationships.

23 Claims, 2 Drawing Sheets



(56) **References Cited**

U.S. PATENT DOCUMENTS

5,860,789	A	1/1999	Sekihara et al.	
6,183,641	B1 *	2/2001	Conrad et al.	A47L 5/22 209/12.1
6,382,903	B1	5/2002	Caruso et al.	
6,749,400	B2	6/2004	Dougherty et al.	
6,893,226	B2	5/2005	Phipps	
7,241,111	B2	7/2007	Harding et al.	
7,578,656	B2	8/2009	Higgins et al.	
7,819,632	B2	10/2010	Hoell et al.	
2007/0059158	A1 *	3/2007	Alvanos et al.	F02C 7/18 415/115
2007/0258813	A1 *	11/2007	Klutz	F01D 5/087 416/90 R
2009/0053058	A1 *	2/2009	Kohlenberg et al.	F02K 1/72 415/227
2009/0208339	A1 *	8/2009	Cherolis	F01D 5/3007 416/219 R
2012/0291449	A1 *	11/2012	Adams	F02C 7/36 60/793

FOREIGN PATENT DOCUMENTS

EP	1927722 A1		6/2008
JP	S61234207 A	*	10/1986

OTHER PUBLICATIONS

Petition for Inter Partes Review of U.S. Pat. No. 10,119,400. *General Electric Company, Petitioner*, v. *United technologies Corporation*, Patent Owner. IPR2020-00346. Dec. 23, 2019.
Declaration of Reza Abhari, Ph.D. In re U.S. Pat. No. 10,119,400. Executed Dec. 11, 2019. pp. 1-96.
Curriculum Vitae of Reza S. Abhari. pp. 1-24.
Alver, A. S. and Wong, J. K. (1975). Improved turbine disk design to increase reliability of airciall jet engines. NASA/CR-134985. pp. 1-64.
NASA Technical Reports Server (NTRS) for Alver NASA/CR-134985.
Fledderjohn, K.R. (1983). The TFE731-5: Evolution of a decade of business jet service. SAE 1983 Transactions. Section 3—vol. 92. Society of Automotive Engineers, Inc. Warrendale, PA. pp. 3.146-3.157.
Howe, D. C. and Marchant, R. D. (1988). Energy Efficient Engine. High-pressure compressor test hardware detailed design report. NASA/CR-180580. pp. 1-233.
NASA Technical Reports Server (NTRS) for Howe NASA/CR-180850.
Barack, W. N. and DOMAS P. A. (1976). An improved turbine disk design to increase reliablity of aircraft jet engines. NASA/CR-135033 pp. 1-131.
Roberts, R., Fiorentino, A. J., and Diehl, L. A. (1977). Pollution reduction Technology Program for class T4 (JT8D) engines. NASA Document ID 19780003124. pp. 59-89.
NASA Technical Reports Server (NTRS) for Roberts NASA Document ID 19780003124.
Declaration of Ingrid Hsieh-Yee, PHD, Under37 C.F.R. § 1.68. Executed Dec. 3, 2019. IPR2020-00346. pp. 1-77.
Mattingly, J.D. (1996). Elements of gas turbine propulsion. New York, New York: McGraw-Hill, Inc. pp. 1-15, 60-62, 85-87, 95-104, 121-123, 223-234, 323-326, 462-479, 517-520, 563-565, 673-675, 682-684, 697-699, 703-705, 302-805, 862-864.
Muktinutalapati, N. R. (2011). Materials for gas turbines—An overview. Advances in Gas Turbine Technology, pp. 293-314.
Declaration of Courtney H. Bailey. In re U.S. Pat. No. 8,511,605 B2. Executed Jul. 19, 2016. pp. 1-4.
What is the NTRS?—NASA Scientific and Technical Information (STI) Program, https://sti.nasa.gov/what-is-the-ntrs/#.X0zufXIKiUk.
Patent Owner's Preliminary Response. *General Electric Company, Petitioner*, v. *Raytheon Technologies Corporation, Patent Owner*. IPR2020-00346. Apr. 15, 2020.
Mattingly, J.D. (1996). Elements of gas turbine propulsion. New York, New York: McGraw-Hill, Inc. pp. 392-411, 615-756.
AirInsight. Breakthrough: The Market Changing Pratt & Whitney Geared Turbofan Engine. Jul. 2014. pp. 1-58.
Pratt & Whitney GTF Engine Revenue Flight Hours Triple in 2019. Dec. 18, 2019. https://newsroom.aviator.aero/pratt-whitney-gtf-engine-revenue-flight-hours-triple-in-2019/.
Decision. Denying Institution of Inter Partes Review 35 U.S.C. § 314. *General Electric Company, Petitioner*, v. *Raytheon Technologies Corporation, Patent Owner*. IPR2020-00346. Jun. 23, 2020.
International Preliminary Report on Patentability for International Application No. PCT/US2013/061319, dated Apr. 9, 2015.
International Preliminary Report on Patentability for PCT Application No. PCT/US2013/061319, dated Apr. 9, 2015.
International Search Report and Written Opinion for International Application No. PCT/US2013/061319 dated Jul. 3, 2014.

* cited by examiner

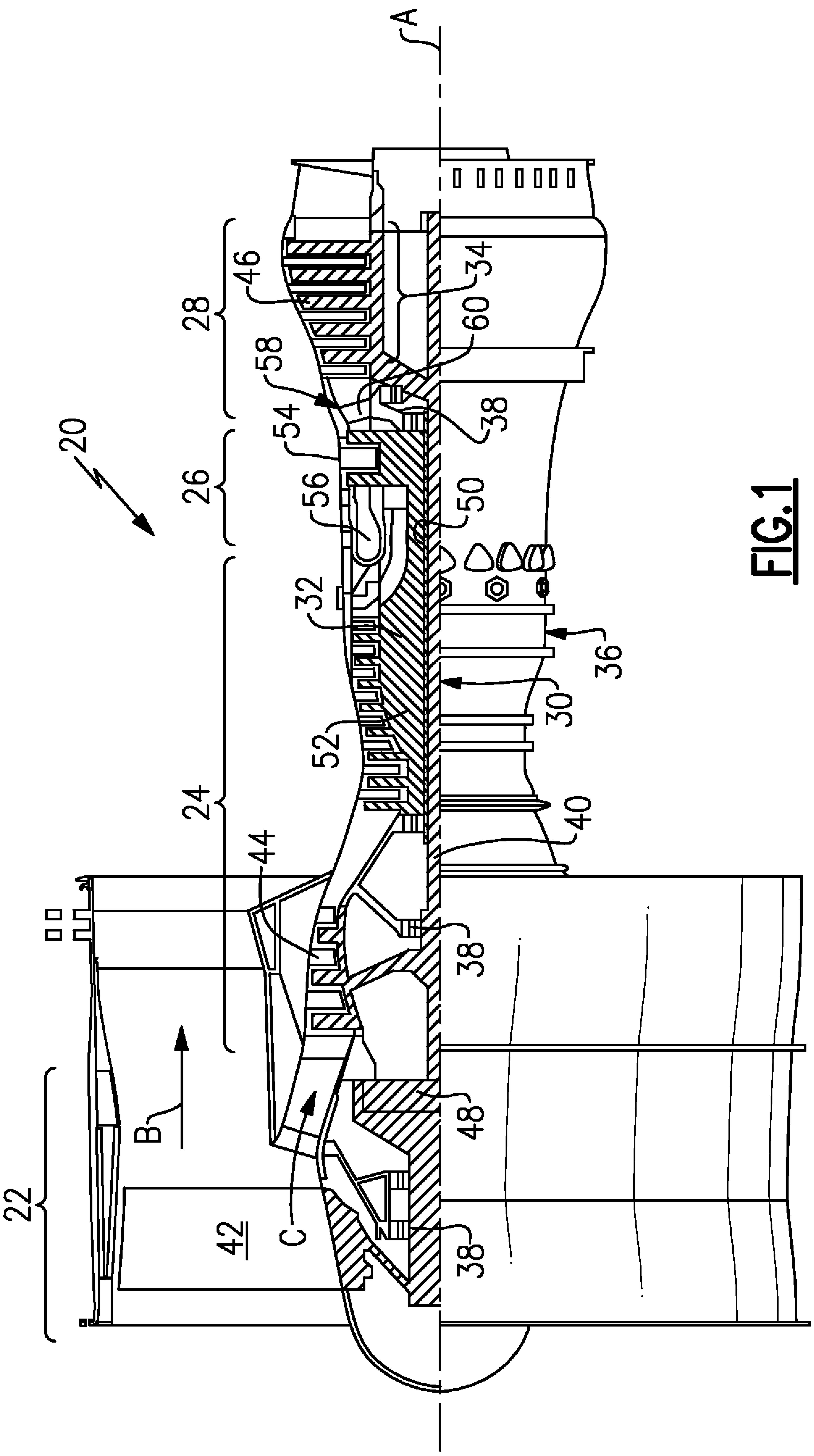


FIG. 1

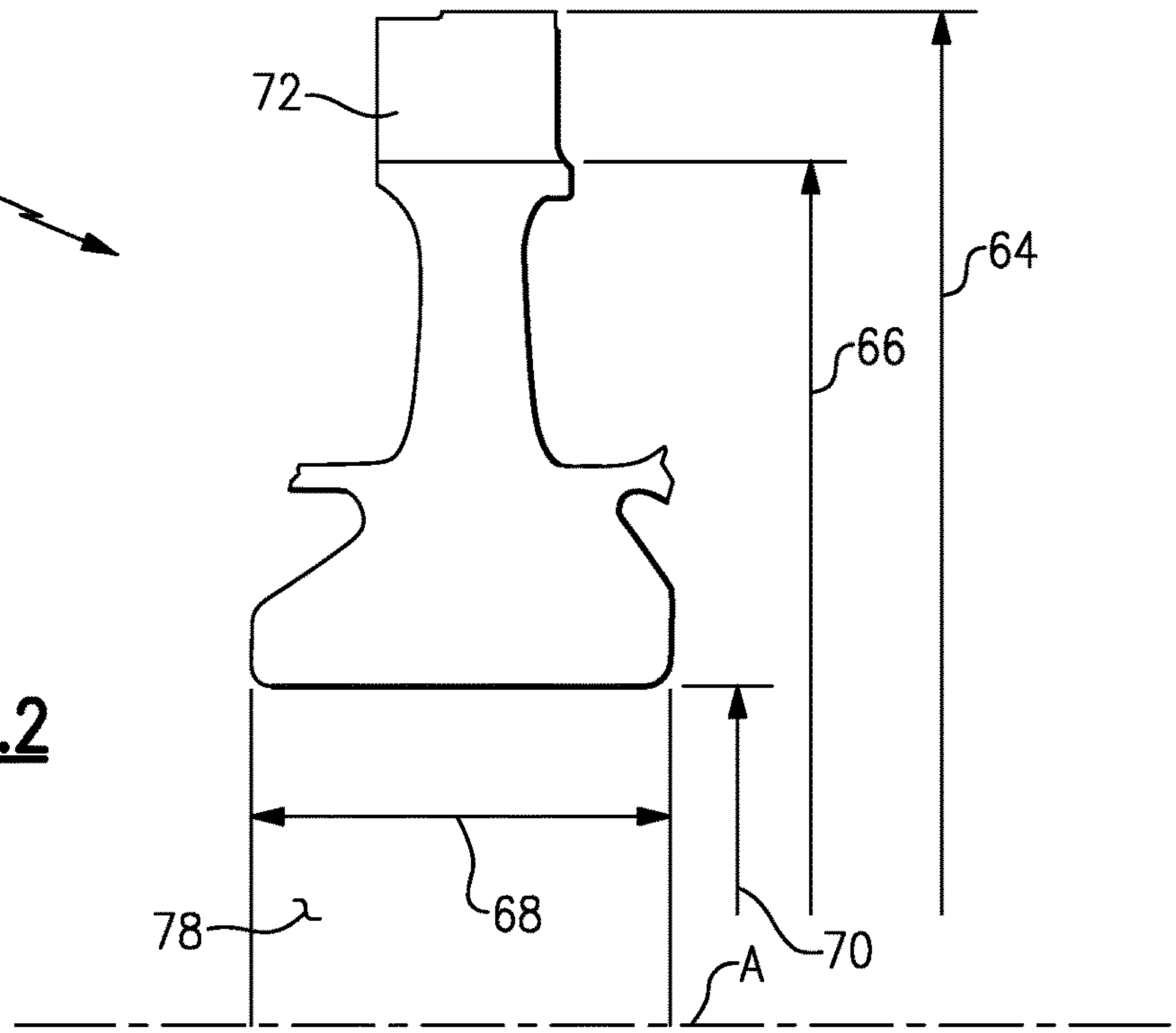


FIG.2

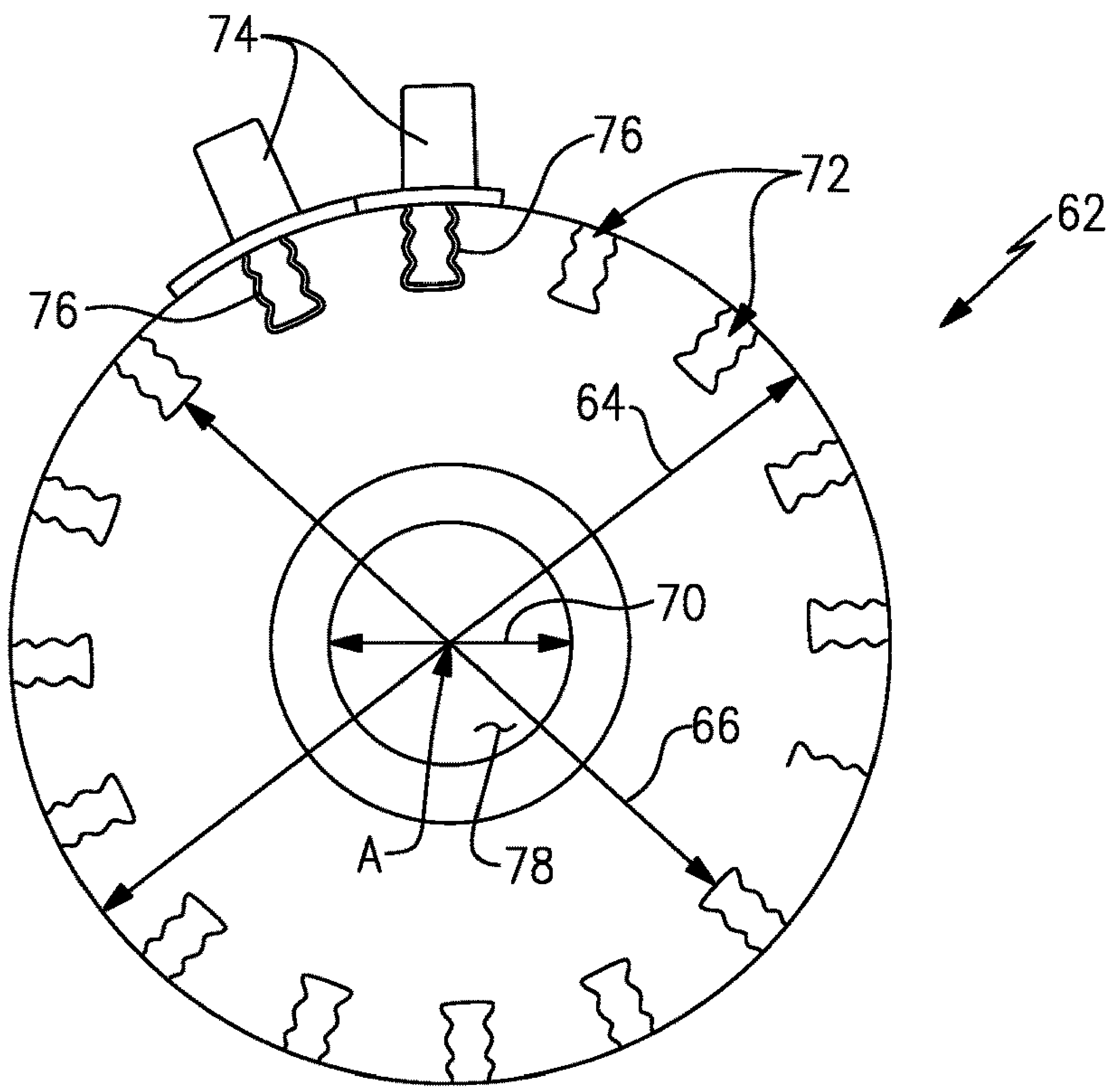


FIG.3

HIGH PRESSURE ROTOR DISK

Matter enclosed in heavy brackets [ ] appears in the original patent but forms no part of this reissue specification; matter printed in italics indicates the additions made by reissue; a claim printed with strikethrough indicates that the claim was canceled, disclaimed, or held invalid by a prior post-patent action or proceeding.

CROSS REFERENCE TO RELATED [APPLICATION] *APPLICATIONS*

This application *is a broadening reissue of U.S. Pat. No. 10,119,400 (filed as U.S. application Ser. No. 13/713,257), which* claims priority to U.S. Provisional Application No. 61/707,009 filed on Sep. 28, 2012.

BACKGROUND

A gas turbine engine typically includes a fan section, and a core engine section including 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.

Turbine and compressor rotor disks operate at high speeds and support blades. Exhaust gases produced in the combustor drive a rotor disk within the turbine section and thereby rotation of a corresponding rotor disk within the compressor section. The turbine disk is attached to drive a shaft that in turn drives the compressor or the fan section.

Engine manufactures continuously seek improvements to thermal, weight and propulsive efficiencies. Improvements to engine architectures have enabled higher speeds and operation at increased temperatures. Accordingly, it is desirable to develop rotor disks that perform at higher speeds and greater temperatures.

SUMMARY

A gas turbine engine according to an exemplary embodiment of this disclosure, among other possible things includes a compressor section including a high pressure compressor and a low pressure compressor. A combustor is in fluid communication with the compressor section. A turbine section is in fluid communication with the combustor. The turbine section includes a high pressure turbine driving the high pressure compressor and a low pressure turbine driving the low pressure compressor. At least one of the high pressure turbine and the high pressure compressor includes a disk having a bore diameter (D) related to a bore width (W) according to a ratio (D/W) between about 1.25 and about 1.65.

In a further embodiment of the foregoing gas turbine engine, the ratio (D/W) is between about 1.35 and about 1.55.

In a further embodiment of any of the foregoing gas turbine engines, the ratio (D/W) is about 1.45.

In a further embodiment of any of the foregoing gas turbine engines, the ratio (D/W) is equal to 1.45.

In a further embodiment of any of the foregoing gas turbine engines, the disk includes an outer diameter (OD) related to the bore diameter (D) according to a ratio (OD/D) that is between about 2.95 and about 3.25.

In a further embodiment of any of the foregoing gas turbine engines, the ratio (OD/D) is between about 3.04 and 3.20.

In a further embodiment of any of the foregoing gas turbine engines, the ratio (OD/D) is about 3.15.

In a further embodiment of any of the foregoing gas turbine engines, the ratio (OD/D) is equal to 3.15.

In a further embodiment of any of the foregoing gas turbine engines, the disk includes a live rim diameter (d) related to the bore diameter (D) according to a ratio (d/D) that is between about 2.25 and about 3.00.

In a further embodiment of any of the foregoing gas turbine engines, the ratio (d/D) is between about 2.50 and about 2.75.

In a further embodiment of any of the foregoing gas turbine engines, the ratio (d/D) is about 2.69.

In a further embodiment of any of the foregoing gas turbine engines, the ratio (d/D) is equal to about 2.69.

A rotor disk for a gas turbine engine according to an exemplary embodiment of this disclosure, among other possible things includes an outer diameter (OD) related to a bore diameter (D) according to a ratio (OD/D) that is between about 2.95 and about 3.25.

In a further embodiment of the foregoing rotor disk, the ratio (OD/D) is between about 3.04 and 3.20.

In a further embodiment of any of the foregoing rotor disks, the ratio (OD/D) is about 3.15.

In a further embodiment of any of the foregoing rotor disks, the bore diameter (D) is related to a bore width (W) according to a ratio (D/W) between about 1.25 and about 1.65.

In a further embodiment of any of the foregoing rotor disks, the ratio (D/W) is between about 1.53 and about 1.55.

In a further embodiment of any of the foregoing rotor disks, the ratio (D/W) is about 1.45.

In a further embodiment of any of the foregoing rotor disks, the disk includes a live rim diameter (d) related to the bore diameter (D) according to a ratio (d/D) that is between about 2.25 and about 3.00.

In a further embodiment of any of the foregoing rotor disks, the ratio (d/D) is between about 2.50 and about 2.75.

In a further embodiment of any of the foregoing rotor disks, the ratio (d/D) is about 2.69.

A method of fabricating a rotor disk for a gas turbine engine according to an exemplary embodiment of this disclosure, among other possible things includes forming a bore which includes a bore diameter (D) and a live rim diameter (d) with a ratio (d/D) of the live rim diameter (d) to the bore diameter (D) being between about 2.25 and about 3.00, forming at least one lug for mounting a blade at a live rim diameter (d), and forming an outer diameter (OD).

In a further embodiment of the foregoing method, includes forming the disk to include a ratio (OD/D) of the outer diameter (OD) to the bore diameter (D) between about 2.95 and about 3.25.

In a further embodiment of any of the foregoing methods, includes forming a bore including a bore diameter (D) and a bore width (W) in a direction parallel to an axis of intended rotation. The bore diameter (D) is related to the bore width (W) according to a ratio (D/W) that is between about 1.25 and 1.65.

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.

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

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

FIG. **2** is a cross-section of an example rotor disk for a gas turbine engine;

FIG. **3** is a front view of the example rotor disk for a gas turbine engine.

DETAILED DESCRIPTION

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**.

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 and where a low spool enables a low pressure turbine to drive a fan via a gearbox, an intermediate spool that enables an intermediate pressure turbine to drive a first compressor of the compressor section, and a high spool that enables a high pressure turbine to drive a high pressure compressor of the compressor section.

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.

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.

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.

The example low pressure turbine **46** has a pressure ratio that is greater than about five (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.

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**.

Airflow through the core flow path 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.

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.

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.

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 fuel consumption—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 point.

"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.

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

The example gas turbine engine includes the fan **42** that comprises in one non-limiting embodiment less than about twenty six (26) fan blades. In another non-limiting embodiment, the fan section **22** includes less than about twenty (20) fan blades. Moreover, in one disclosed embodiment the low pressure turbine **46** includes no more than about six (6) turbine rotors schematically indicated at **34**. In another non-limiting example embodiment the low pressure turbine **46** includes about three (3) turbine rotors. A ratio between the number of fan blades **42** and the number of low pressure turbine rotors is between about 3.3 and about 8.6. 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 rotors **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.

As appreciated, although an engine for mounting and powering an aircraft is described and shown, the present disclosure may also provide benefits to land based and industrial gas turbine engines.

Referring to FIGS. **2** and **3**, with continued reference to FIG. **1**, an example rotor disk **62** is shown and includes a plurality of lugs **72** for supporting blades **74** (FIG. **3**). The example rotor disk **62** is provided as part of the high pressure turbine **54**. However, the example rotor disk may also be part of the high pressure compressor **52**.

The rotor disk **62** supports the turbine blades **74** that are driven by high speed exhaust gases generated in the combustor section **26**. The example turbine disk **62** includes an outer diameter **64**, a live rim diameter **66** and a bore **78** having a bore diameter **70**. The bore **78** includes a width **68** in a direction parallel to the axis A.

The bore diameter **70** is that diameter between an inner most surface of the bore **78** about the axis A. The live rim disk diameter **66** is the diameter that extends between bottom and radially inward surfaces of the disk lugs **72**. The turbine blades **74** are supported within the disk lugs **72** by corresponding mating profiles disposed at an interface **76**.

The bore width **68** of the rotor disk **62** in this example is the greatest width on the main body of the rotor disk **62**. The greatest width of the main body of the rotor disk **62** does not include additional widths associated with appendages, arms or other structures that extend from the main body of the rotor disk **62**. The example bore width **68** is disposed at a distance spaced apart from the axis A determined to provide desired performance properties and to accommodate high rotational speeds encountered during operation. The example distance is defined as the bore diameter **70** (D).

The speed at which the high pressure turbine rotor disk **62** operates is accommodated at least in part by a relationship between the live rim diameter **66** (d) to the bore diameter **70** (D) defined by a ratio of the live rim diameter **66** to the bore diameter **70** (i.e., d/D). In the disclosed example embodiment the ratio is between about 2.25 and about 3.00. In another disclosed example embodiment, the ratio d/D is between about 2.50 and about 2.75. In another disclosed example embodiment the ratio d/D is about 2.69.

The disk **62** includes the bore width **68** (W). The bore width **68** (W) is the width at the bore **78** parallel to the axis A. In one non-limiting embodiment, a relationship between the bore width **68** (W) and the bore diameter **70** (D) is defined by a ratio of D/W. In a disclosed example the ratio D/W is between about 1.25 and about 1.65. In another disclosed embodiment the ratio of D/W is between about 1.35 and about 1.55. In a further embodiment the ratio of D/W is about 1.45.

The outer diameter **64** (OD) is related to the bore diameter **70** (D) by a ratio of the outer diameter **64** (OD) to the bore diameter **70** (D) (i.e., OD/D). In one example the ratio OD/D is between about 2.95 and 3.25. In another embodiment the ratio (OD/D) is between about 3.04 and 3.20. In a further embodiment the ratio (OD/D) is about 3.15.

The example rotor disk for a gas turbine engine is fabricated by forming a bore including the bore diameter (D) and the bore width (W) in a direction parallel to an axis of intended rotation according to the above disclosed ratio. Additional processing steps are performed to form at least one lug **72** at the live rim diameter **66** (d). Further, the outer diameter **64**(OD) is formed according to the above defined ratios. Fabrication further includes forming the rotor disk to include the ratio (OD/D) of the outer diameter (OD) to the bore diameter (D) as disclosed above. The fabrication further includes forming the rotor disk to include the ratio (d/D) of the live rim diameter (d) to the bore diameter (D) as disclosed above.

The example rotor disk **62** is fabricated from a material capable of withstanding rotational speeds and temperatures encountered during operation of the gas turbine engine. The rotor disk **62** can be formed from any material or combination of materials such as for example nickel-based alloys and carbon steels. Moreover, it is within the contemplation of this disclosure that the example rotor disk **62** may be fabricated utilizing known fabrication and machining processes verified to enable operation of a completed rotor disk **62** within desired performance parameters within the gas turbine engine. For example, the rotor disk **62** may be fabricated as a casting or forging followed by machining to obtain the desired shape and features.

The example rotor disk includes relationships that enable performance at high rotational speeds in the high temperature environment of the high pressure turbine **54**.

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 compressor section including a high pressure compressor and a low pressure compressor;~~

~~a combustor in fluid communication with the compressor section;~~

~~a turbine section in fluid communication with the combustor, wherein the turbine section includes a high pressure turbine driving the high pressure compressor and a low pressure turbine driving the low pressure compressor, wherein at least one of the high pressure turbine and the high pressure compressor includes a disk having a bore diameter (D) related to a bore width (W) according to a ratio (D/W) between 1.25 and 1.65.~~

~~**2**. The gas turbine engine as recited in claim **1**, wherein the ratio (DAY) is between 1.35 and 1.55.~~

[3. The gas turbine engine as recited in claim **1**, wherein the ratio (D/W) is 1.45.**]**

4. **[**The gas turbine engine as recited in claim **1]** *A gas turbine engine comprising:*

a compressor section including a high pressure compressor and a low pressure compressor;

a combustor in fluid communication with the compressor section; and *a turbine section in fluid communication with the combustor, wherein the turbine section includes a high pressure turbine driving the high pressure compressor and a low pressure turbine driving the low pressure compressor, wherein at least one of the high pressure turbine and the high pressure compressor includes a disk having a bore diameter (D) related to a bore width (W) according to a ratio (D/W) between 1.25 and 1.65, and* wherein the disk includes an outer diameter (OD) related to the bore diameter (D) according to a ratio (OD/D) that is between 2.95 and 3.25.

5. The gas turbine engine as recited in claim **4**, wherein the ratio (OD/D) is between 3.04 and 3.20.

6. The gas turbine engine as recited in claim **4**, wherein the ratio (OD/D) is 3.15.

7. **[**The gas turbine engine as recited in claim **1]** *A gas turbine engine comprising:*

a compressor section including a high pressure compressor and a low pressure compressor;

a combustor in fluid communication with the compressor section; and

a turbine section in fluid communication with the combustor, wherein the turbine section includes a high pressure turbine driving the high pressure compressor and a low pressure turbine driving the low pressure compressor, wherein at least one of the high pressure turbine and the high pressure compressor includes a disk having a bore diameter (D) related to a bore width (W) according to a ratio (D/W) between 1.25 and 1.65, and wherein the disk includes a live rim diameter (d) related to the bore diameter (D) according to a ratio (d/D) that is between 2.25 and 3.00.

8. The gas turbine engine as recited in claim **7**, wherein the ratio (d/D) is between 2.50 and 2.75.

9. The gas turbine engine as recited in claim **7**, wherein the ratio (d/D) is 2.69.

~~**10**. A rotor disk for a gas turbine engine comprising:~~

~~an outer diameter (OD) related to a bore diameter (D) according to a ratio (OD/D) that is between 2.95 and 3.25.~~

[11. The rotor disk as recited in claim **10**, wherein the ratio (OD/D) is between 3.04 and 3.20.**]**

[12. The rotor disk as recited in claim **10**, wherein the ratio (OD/D) is 3.15.**]**

[13. The rotor disk as recited in claim **10**, wherein the bore diameter (D) is related to a bore width (W) according to a ratio (D/W) between 1.25 and 1.65.**]**

[14. The rotor disk as recited in claim **13**, wherein the ratio (D/W) is between 1.53 and 1.55.**]**

[15. The rotor disk as recited in claim **13**, wherein the ratio (DAY) is 1.45.**]**

~~**16**. The rotor disk as recited in claim **10**, wherein the disk includes a live rim diameter (d) related to the bore diameter (D) according to a ratio (d/D) that is between 2.25 and 3.00.~~

[17. The rotor disk as recited in claim **16**, wherein the ratio (d/D) is between 2.50 and 2.75.**]**

[18. The rotor disk as recited in claim **16**, wherein the ratio (d/D) is 2.69.**]**

~~**19**. A method of fabricating a rotor disk for a gas turbine engine comprising:~~

~~forming a bore including a bore diameter (D) and a live rim diameter (d) with a ratio (d/D) of the live rim diameter (d) to the bore diameter (D) being between 2.25 and 3.00~~

~~forming at least one lug for mounting a blade at the live rim diameter (d); and~~

~~forming an outer diameter (OD).~~

~~**20**. The method as recited in claim **19**, including forming the disk to include a ratio (OD/D) of the outer diameter (OD) to the bore diameter (D) between 2.95 and 3.25.~~

[21. The method as recited in claim **19**, including forming a bore including a bore diameter (D) and a bore width (W) in a direction parallel to an axis of intended rotation, wherein the bore diameter (D) is related to the bore width (W) according to a ratio (D/W) that is between 1.25 and 1.65.**]**

22. The gas turbine engine as set forth in claim 4, wherein the ratio (D/W) is between 1.35 and 1.55.

23. The gas turbine engine as recited in claim 4, wherein the ratio (D/W) is 1.45.

24. The gas turbine engine as set forth in claim 7, wherein the ratio (D/W) is between 1.35 and 1.55.

25. The gas turbine engine as recited in claim 7, wherein the ratio (D/W) is 1.45.

26. A gas turbine engine, comprising:

a high speed shaft mounted for rotation about a longitudinal axis;

a low speed shaft coupled to a fan of a fan section, the low speed shaft mounted for rotation concentrically with the high speed shaft, the low speed shaft configured to rotate at a lower speed than the high speed shaft;

a rotor disk of a turbine attached to the high speed shaft, the rotor disk of the turbine having a first bore diameter (D_1) *related to a first bore width* (W_1) *according to a first ratio* (D_1/W_1) *between 1.25 and 1.65; and*

a rotor disk of a compressor attached to the high speed shaft, the rotor disk of the compressor having a second bore diameter (D_2) *related to a compressor outer diameter* (OD_2) *according to a second ratio* (OD_2/D_2) *between 2.95 and 3.25;*

wherein the fan has less than 26 fan blades; and

further comprising a speed change device, wherein a low pressure turbine attached to the low speed shaft is configured to drive the fan through the speed change device at a lower speed than the low speed shaft.

27. The gas turbine engine as recited in claim 26, wherein the fan section has a low fan pressure ratio measured across the fan blades alone of less than 1.45 at a flight condition of 0.8 Mach and 35,000 ft., wherein the fan section has a low corrected fan tip speed less than 1,150 ft./sec at the flight condition of 0.8 Mach and 35,000 ft., and wherein the low corrected fan tip speed is an actual fan tip speed divided by $((Tram\ ^\circ R)/(518.7^\circ R))^{0.5}$.

28. The gas turbine engine as set forth in claim 26, wherein the turbine is a two stage turbine.

29. The gas turbine engine as set forth in claim 26, wherein the ratio (D_1/W_1) *is between 1.35 and 1.55.*

30. The gas turbine engine as recited in claim 26, wherein the ratio (D_1/W_1) *is 1.45.*

31. The gas turbine engine as recited in claim 26, wherein the ratio (OD_2/D_2) *is between 3.04 and 3.20.*

32. The gas turbine engine as recited in claim 26, wherein the ratio (OD_2/D_2) *is 3.15.*

33. A gas turbine engine, comprising:

a high speed shaft mounted for rotation about a longitudinal axis;

a low speed shaft coupled to a fan of a fan section, the low speed shaft mounted for rotation concentrically with the high speed shaft, the low speed shaft configured to rotate at a lower speed than the high speed shaft;

a rotor disk of a turbine attached to the high speed shaft, the rotor disk of the turbine having a first bore diameter (D_1) *related to a first bore width* (W_1) *according to a first ratio* (D_1/W_1) *between 1.25 and 1.65;*

a rotor disk of a compressor attached to the high speed shaft, the rotor disk of the compressor having a second bore diameter (D_2) related to a compressor outer diameter (OD_2) according to a second ratio (OD_2/D_2) between 2.95 and 3.25;

wherein the fan has less than 26 fan blades; and

wherein the fan section has a low fan pressure ratio measured across the fan blades alone of less than 1.45 at a flight condition of 0.8 Mach and 35,000 ft., wherein the fan section has a low corrected fan tip speed less than 1,150 ft./sec at the flight condition of 0.8 Mach and 35,000 ft., and wherein the low corrected fan tip speed is an actual fan tip speed divided by $((Tram\ ° R)/(518.7° R))^{0.5}$.

34. The gas turbine engine as set forth in claim 33, wherein the turbine is a two stage turbine.

35. The gas turbine engine as set forth in claim 33, wherein the ratio (D_1/W_1) is between 1.35 and 1.55.

36. The gas turbine engine as recited in claim 33, wherein the ratio (D_1/W_1) is 1.45.

37. The gas turbine engine as recited in claim 33, wherein the ratio (OD_2/D_2) is between 3.04 and 3.20.

38. The gas turbine engine as recited in claim 33, wherein the ratio (OD_2/D_2) is 3.15.

* * * * *