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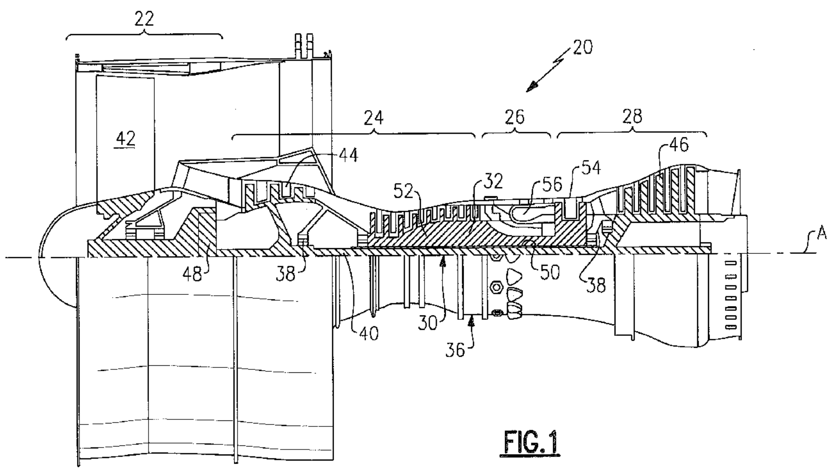

FIG.1

(57) Abstract: A liner for a gas turbine engine includes a cold side having a partition which extends therefrom to define a forward cavity and an aft cavity, each cavity having a plurality of film holes to communicate a coolant from the cold side to the hot side.

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COMBUSTOR SECTION OF A GAS TURBINE ENGINE

Applicant hereby claims priority to U.S. Patent Application No. 61/707,497 filed September 28, 2012, the disclosure of which is herein incorporated by reference.

BACKGROUND

[0001] The present disclosure relates to a gas turbine engine and, more particularly, to a combustor section therefor.

[0002] Gas turbine engines, such as those that power modern commercial and military aircraft, generally include a compressor to pressurize an airflow, a combustor for burning a hydrocarbon fuel in the presence of the pressurized air, and a turbine to extract energy from the resultant combustion gases.

[0003] Combustors are subject to high thermal loads for prolonged periods of time. To alleviate the accompanying thermal stresses, the combustor walls are cooled. In one cooling arrangement a twin wall configuration includes a shell lined with heat shields often referred to as Impingement Film-Cooled Floatwall (IFF) liners which are attached to the shell with studs and nuts. Dilution holes in the shell are aligned with respective dilution holes in the liner for introduction of dilution air. In addition to the dilution holes, relatively smaller air impingement holes direct cooling air between the outer shell and the liners to cool the backside of the liners. This cooling air then exits effusion holes in the liners to form a cooling film on the hot side of the liners.

[0004] Although effective, the twin wall arrangement requires close control of thermal load as the thermal load in a combustor may be non-uniform in some locations such that the combustor may experience differential thermal growth, stress and strain which may negatively effect the usable life of the combustor. One such non-uniform thermal load is caused by vane bow waves immediately upstream of the Nozzle Guide Vanes (NGVs).

SUMMARY

[0005] A liner for use in a combustor of a gas turbine engine according to one disclosed non-limiting embodiment of the present disclosure includes a hot side and a cold side with one or more raised rails that project from said cold side to form partitions that define at least a forward cavity and an aft cavity, each said cavity having a plurality of holes to communicate a coolant from said cold side to said hot side.

[0006] In a further embodiment of the foregoing embodiment, an area of said aft cavity is less than approximately twenty-five percent 25% that of said forward cavity.

[0007] In a further embodiment of any of the foregoing embodiments, an axial length of said aft cavity is less than approximately fifteen percent 15% an axial length of said forward cavity.

[0008] In a further embodiment of any of the foregoing embodiments, the aft cavity is transverse to a longitudinal length of said liner.

[0009] In a further embodiment of any of the foregoing embodiments, the liner includes a plurality of studs which extend from said cold side.

[0010] In a further embodiment of any of the foregoing embodiments, the liner includes a plurality of studs which extend from said cold side, said plurality of studs extend from within said forward cavity.

[0011] In a further embodiment of any of the foregoing embodiments, the liner is mountable within a diffuser case module such that said aft cavity is immediately upstream of a Nozzle Guide Vane (NGV).

[0012] In a further embodiment of any of the foregoing embodiments, the partition is generally rectilinear to define said aft cavity. In the alternative or additionally thereto, in the foregoing embodiment a rail of said partition is adjacent to an edge of said liner.

[0013] A wall for use in a combustor of a gas turbine engine according to another disclosed non-limiting embodiment of the present disclosure includes a shell and a liner mountable to said shell to define a forward cavity and an aft cavity, said aft cavity operable at a pressure different than said forward cavity.

[0014] In a further embodiment of the foregoing embodiment, the aft cavity is operable at a pressure greater than said forward cavity.

[0015] In a further embodiment of any of the foregoing embodiments, the wall includes a plurality of studs which extend from said liner and through said shell.

[0016] In a further embodiment of any of the foregoing embodiments, the wall includes a partition which extends from a cold side of said liner into contact with said shell to define said forward cavity and said aft cavity to define a respective pressure.

[0017] In a further embodiment of any of the foregoing embodiments, the shell defines a plurality of cooling impingement holes to communicate coolant into said forward cavity and said aft cavity.

[0018] In a further embodiment of any of the foregoing embodiments, the shell defines a plurality of cooling impingement holes to communicate coolant into said forward cavity and said aft cavity to generate a desired pressure within each.

[0019] In a further embodiment of any of the foregoing embodiments, the liner is mountable within a diffuser case module such that said aft cavity is immediately upstream of a Nozzle Guide Vane (NGV) and directly adjacent a mount flange of said shell.

[0020] A method of controlling a thermal load in a combustor of a gas turbine according to another disclosed non-limiting embodiment of the present disclosure includes generating a pressure in an aft cavity of a combustor wall different than a pressure in a forward cavity of the combustor wall, the aft cavity immediately upstream of a Nozzle Guide Vane (NGV).

[0021] In a further embodiment of the foregoing embodiment, the method includes generating a higher pressure in the aft cavity relative to said forward cavity.

BRIEF DESCRIPTION OF THE DRAWINGS

[0022] Various features will become apparent to those skilled in the art from the following detailed description of the disclosed non-limiting embodiment. The drawings that accompany the detailed description can be briefly described as follows:

[0023] Figure 1 is a schematic cross-section of a gas turbine engine;

[0024] Figure 2 is a partial longitudinal schematic sectional view of a combustor section according to one non-limiting embodiment that may be used with the gas turbine engine shown in Figure 1;

[0025] Figure 3 is an exploded view of a wall of the combustor;

[0026] Figure 4 is a cold side view of an outer liner of a wall of the combustor according to one non-limiting embodiment;

[0027] Figure 5 is a cold side view of an outer liner of a wall of the combustor according to another non-limiting embodiment; and

[0028] Figure 6 is an expanded schematic sectional view of an aft edge of the liner illustrating an aft cavity upstream of a Nozzle Guide Vane (NGV).

DETAILED DESCRIPTION

[0029] Figure 1 schematically illustrates a gas turbine engine 20. The gas turbine engine 20 is disclosed herein as a two-spool turbo fan that generally incorporates a fan section 22, a compressor section 24, a combustor section 26 and a turbine section 28. Alternative engines might include an augmentor section (not shown) among other systems or features. The fan section 22 drives air along a bypass flowpath while the compressor section 24 drives air along a core flowpath for compression and communication into the combustor section 26 then expansion through the turbine section 28. Although depicted as a turbofan in the disclosed non-limiting embodiment, 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 such as a turbojets, turboshafts, and three-spool (plus fan) turbofans wherein an intermediate spool includes an intermediate pressure compressor ("IPC") between a Low Pressure Compressor ("LPC") and a High Pressure Compressor ("HPC"), and an intermediate pressure turbine ("IPT") between the high pressure turbine ("HPT") and the Low pressure Turbine ("LPT").

[0030] The engine 20 generally includes a low spool 30 and a high spool 32 mounted for rotation about an engine central longitudinal axis A relative to an engine static structure 36 via several bearing structures 38. The low spool 30 generally includes an inner shaft 40 that interconnects a fan 42, a low pressure compressor 44 ("LPC") and a low pressure turbine 46 ("LPT"). The inner shaft 40 drives the fan 42 directly or through a geared architecture 48 to drive the fan 42 at a lower speed than the low spool 30. An exemplary reduction transmission is an epicyclic transmission, namely a planetary or star gear system.

[0031] The high spool 32 includes an outer shaft 50 that interconnects a high pressure compressor 52 ("HPC") and high pressure turbine 54 ("HPT"). A combustor 56 is arranged between the high pressure compressor 52 and the high pressure turbine 54. The inner shaft 40 and the outer shaft 50 are concentric and rotate about the engine central longitudinal axis A which is collinear with their longitudinal axes.

[0032] Core airflow is compressed by the LPC 44 then the HPC 52, mixed with the fuel and burned in the combustor 56, then expanded over the HPT 54 and the LPT 46. The turbines 54, 46 rotationally drive the respective low spool 30 and high spool 32 in response to the expansion. The main engine shafts 40, 50 are supported at a plurality of points by bearing structures 38 within the static structure 36. It should be understood that various bearing structures 38 at various locations may alternatively or additionally be provided.

[0033] In one non-limiting example, the gas turbine engine 20 is a high-bypass geared aircraft engine. In a further example, the gas turbine engine 20 bypass ratio is greater than about six (6:1). The geared architecture 48 can include an epicyclic gear train, such as a planetary gear system or other gear system. The example epicyclic gear train has a gear reduction ratio of greater than about 2.3, and in another example is greater than about 2.5:1. The geared turbofan enables operation of the low spool 30 at higher speeds which can increase the operational efficiency of the low pressure compressor 44 and low pressure turbine 46 and render increased pressure in a fewer number of stages.

[0034] A pressure ratio associated with the low pressure turbine 46 is pressure measured prior to the inlet of the low pressure turbine 46 as related to the pressure at the outlet of the low pressure turbine 46 prior to an exhaust nozzle of the gas turbine engine 20. In one non-limiting embodiment, the bypass ratio of the gas turbine engine 20 is greater than about ten (10:1), the fan diameter is significantly larger than that of the low pressure compressor 44, and the low pressure turbine 46 has a pressure ratio that is greater than about five (5:1). It should be understood, however, that the above parameters are only exemplary of one embodiment of a geared architecture engine and that the present disclosure is applicable to other gas turbine engines including direct drive turbofans.

[0035] In one embodiment, a significant amount of thrust is provided by the bypass flow path B due to the high bypass ratio. The fan section 22 of the gas turbine engine 20 is designed for a particular flight condition - typically cruise at about 0.8 Mach and about 35,000 feet. This flight condition, with the gas turbine engine 20 at its best fuel consumption, is also known as bucket cruise Thrust Specific Fuel Consumption (TSFC). TSFC is an industry standard parameter of fuel consumption per unit of thrust.

[0036] Fan Pressure Ratio is the pressure ratio across a blade of the fan section 22 without the use of a Fan Exit Guide Vane system. The low Fan Pressure Ratio according to one non-limiting embodiment of the example gas turbine engine 20 is less than 1.45. Low Corrected Fan Tip Speed is the actual fan tip speed divided by an industry standard temperature correction of (“T” / $518.7^{0.5}$) in which “T” represents the ambient temperature in degrees Rankine. The Low Corrected Fan Tip Speed according to one non-limiting embodiment of the example gas turbine engine 20 is less than about 1150 fps (351 m/s).

[0037] With reference to Figure 2, the combustor 56 generally includes an outer combustor wall 60, an inner combustor wall 62 and a diffuser case module 64. The outer combustor wall 60 and the inner combustor wall 62 are spaced apart such that a combustion chamber 66 is defined therebetween. The combustion chamber 66 is generally annular in shape. The outer combustor wall 60 is spaced radially inward from an outer diffuser case 64-O of the diffuser case module 64 to define an outer annular plenum 76. The inner combustor wall 62 is spaced radially outward from an inner diffuser case 64-I of the diffuser case module 64 to define an inner annular plenum 78. It should be understood that although a particular combustor is illustrated, other combustor types with various combustor liner arrangements will also benefit herefrom. It should be further understood that the disclosed cooling flow paths are but an illustrated embodiment and should not be limited only thereto.

[0038] The combustor walls 60, 62 contain the combustion products for direction toward the turbine section 28. Each combustor wall 60, 62 generally includes a respective support shell 68, 70 which supports one or more liners 72, 74 mounted to a hot side of the respective support shell 68, 70. The liners 72, 74, often referred to as Impingement Film Float (IFF) wall panels define a generally rectilinear liner array which form the annular combustor chamber 66. Each of the liners 72, 74 may be generally rectilinear and manufactured of, for example, a nickel based super alloy, ceramic or other temperature resistant material. In one disclosed non-limiting embodiment, the array includes a multiple of forward liners 72A and a multiple of aft liners 72B that line the hot side of the outer shell 68 and a multiple of forward liners 74A and a multiple of aft liners 74B that line the hot side of the inner shell 70.

[0039] The combustor 56 further includes a forward assembly 80 immediately downstream of the compressor section 24 to receive compressed airflow therefrom. The forward assembly 80 generally includes an annular hood 82, a bulkhead assembly 84, a multiple of fuel nozzles 86 (one shown) and a multiple of fuel nozzle guides 90 (one shown). Each of the fuel nozzle guides 90 is circumferentially aligned with one of the hood ports 94 to project through the bulkhead assembly 84. Each bulkhead assembly 84 includes a bulkhead support shell 96 secured to the combustor walls 60, 62, and a multiple of circumferentially distributed bulkhead heatshields segments 98 secured to the bulkhead support shell 96 around the central opening 92.

[0040] The annular hood 82 extends radially between, and is secured to, the forwardmost ends of the combustor walls 60, 62. The annular hood 82 includes a multiple of circumferentially distributed hood ports 94 that accommodate the respective fuel nozzle 86 and introduce air into the forward end of the combustion chamber 66 through a central opening 92. Each fuel nozzle 86 may be secured to the diffuser case module 64 and project through one of the hood ports 94 and through the central opening 92 within the respective fuel nozzle guide 90.

[0041] The forward assembly 80 introduces core combustion air into the forward end of the combustion chamber 66 while the remainder enters the outer annular plenum 76 and the inner annular plenum 78. The multiple of fuel nozzles 86 and surrounding structure generate a blended fuel-air mixture that supports combustion in the combustion chamber 66.

[0042] Opposite the forward assembly 80, the outer and inner support shells 68, 70 are mounted to a first row of Nozzle Guide Vanes (NGVs) 54A in the HPT 54. In one disclosed non-limiting embodiment, thirty-two (32) NGVs 54A are located immediately downstream of the combustor 56. The NGVs 54A in one disclosed non-limiting embodiment, are the first static vane structure upstream of a first turbine rotor in the turbine section 28. The NGVs 54A are static engine components which direct core airflow combustion gases onto the turbine blades of the first turbine rotor in the turbine section 28 to facilitate the conversion of pressure energy into kinetic energy. The core airflow combustion gases are also accelerated by the NGVs 54A because of their convergent shape and are typically given a "spin" or a "swirl" in the direction of turbine rotor rotation. The turbine rotor blades absorb this energy to drive the turbine rotor at high speed. The NGVs 54A generate bow waves along a leading edge 54L which may shorten combustor 56 service life.

[0043] With reference to Figure 3, studs 100 extend from the liners 72, 74 to mount the liners 72, 74 to the respective shells 68, 70 with fasteners 102 such as nuts. That is, the studs 100 project rigidly from the liners 72, 74 and through the respective shells 68, 70 to receive and removable engage the fasteners 102 at a threaded distal end. Cooling impingement holes 104 penetrate through the shells 68, 70 to allow a coolant from the respective annular plenums 76, 78 to enter cavities 106A, 106B (Figure 4) formed in the combustor walls 60, 62 between the respective shells 68, 70 and liners 72, 74. Cooling film or effusion holes 108 penetrate each of the liners 72, 74 to allow the cooling air to pass from the cavities 106A, 106B along a cold side 110 of the liners 72, 74 to a hot side 112 of the liner 72, 74 to promote the formation of a film or blanket of cooling air over the hot side 112.

[0044] The cavities 106A, 106B are defined by partitions 114 that project from the cold side 110 to contact the respective shells 68, 70 to form enclosed spaces in the walls 60, 62 when the liners 72, 74 are mounted to the shells 68, 70. One or more raised rails 116 that extend from the cold side 110 of the liners 72, 74, define the partitions 114 which in the disclosed non-limiting embodiment define a forward cavity 106A and an aft cavity 106B.

[0045] One or more raised rails 116 project laterally from the cold side 110 of the liners 72, 74 to contact or come in close proximity to the respective shells 68, 70. The rails 116 generally divide the cold side 110 into partitions 114. Each cavity 106A, 106B is defined by respective partitions 114 of the cold side 110 and the shell 68 or shell 70, and generally becomes an enclosed space when the liners 72, 74 are mounted to the respective shells 68, 70.

[0046] With reference to Figure 4, the aft cavity 106B is transverse to a longitudinal length of the liner 72B. The aft cavity 106B in one disclosed non-limiting embodiment defines an area less than approximately twenty-five percent (25%) that of the forward cavity 106A and an axial length less than approximately fifteen percent (15%) of an axial length of the forward cavity 106A. It should be appreciated that the aft cavity 106B is sized to facilitate coolant flow in front of the leading edge 54L of the NGVs 54A. In this disclosed non-limiting embodiment, the plurality of studs 100 extend from within the forward cavity 106A of the cold side 110 such that the aft cavity 106B may be unobstructed, however, other arrangements with, for example, various cooling hole and stud arrangements (Figure 5) will also benefit therefrom.

[0047] With reference to Figure 6, the forward and aft segregation of the liners 72, 74 facilitates maintenance of the pressure in the aft cavity 106B at a pressure different than that of the upstream forward cavity 106A. Although described and illustrated in detail herein with respect to the outer liner 72B and the outer annular plenum 76, it should be appreciated that the inner liner 74B and the inner annular plenum 78 may be generally similar and thus need not be described in detail. The aft cavity 106B is axially located radially inboard of a mount flange 118 of the support shell 68, 70 which abuts a location structure 120 of the NGVs 54A. That is, the aft cavity 106B is immediately upstream of the leading edge 54L of the NGVs 54A.

[0048] The outer annular plenum 76 in the disclosed non-limiting embodiment operates at a pressure of approximately 500 psia (3447 kPa) and is referred to herein as P3. As further perspective, P1 is a pressure in front of the fan section 22; P2 is a pressure at the leading edge of the fan 42; P2.5 is between the LPC 44 and the HPC 52; P3 is the pressure aft of the

HPC 44; P4 is the pressure in the combustion chamber 66; P4.5 is the pressure between the HPT 54 and the LPT 46; and P5 is the pressure aft of the LPT 46 (Figure 1).

[0049] The air supply to the aft cavity 106B is tailored through control of the size and number of Cooling impingement holes 104 to the aft cavity 106B to vary the pressure drop of the aft cavity 106B separate from the balance of the liner 72B. Control of the pressure within the aft cavity 106B facilitates reduction of bow wave distress. For example, the pressure within the forward cavity 106A is approximately 97%P3 while the pressure in the aft cavity 106B is approximately 98%P3 as compared to P4 which is approximately 96%P3. The pressure within the aft cavity 106B may be anywhere between P4 and P3, but is maintained in the disclosed non-limiting embodiment above the pressure in the forward cavity 106A to maintain a higher total pressure to counteract the bow wave in front of the leading edge 54L of the NGVs 54A and thereby minimize non-uniform thermal load.

[0050] It should be understood that relative positional terms such as "forward," "aft," "upper," "lower," "above," "below," and the like are with reference to the normal operational attitude of the vehicle and should not be considered otherwise limiting.

[0051] It should be understood that like reference numerals identify corresponding or similar elements throughout the several drawings. It should also be understood that although a particular component arrangement is disclosed in the illustrated embodiment, other arrangements will benefit herefrom.

[0052] Although particular step sequences are shown, described, and claimed, it should be understood that steps may be performed in any order, separated or combined unless otherwise indicated and will still benefit from the present disclosure.

[0053] The foregoing description is exemplary rather than defined by the limitations within. Various non-limiting embodiments are disclosed herein, however, one of ordinary skill in the art would recognize that various modifications and variations in light of the above teachings will fall within the scope of the appended claims. It is therefore to be understood that within the scope of the appended claims, the disclosure may be practiced other than as specifically described. For that reason the appended claims should be studied to determine true scope and content.

CLAIMS

What is claimed is:

1. A liner for use in a combustor of a gas turbine engine comprising:

a hot side; and

a cold side with one or more raised rails that project from said cold side to form partitions that define at least a forward cavity and an aft cavity, each said cavity having a plurality of holes to communicate a coolant from said cold side to said hot side.

2. The liner as recited in claim 1, wherein an area of said aft cavity is less than approximately twenty-five percent 25% that of said forward cavity.

3. The liner as recited in claim 1, wherein an axial length of said aft cavity is less than approximately fifteen percent 15% an axial length of said forward cavity.

4. The liner as recited in claim 1, wherein said aft cavity is transverse to a longitudinal length of said liner.

5. The liner as recited in claim 1, further comprising a plurality of studs which extend from said cold side.

6. The liner as recited in claim 1, further comprising a plurality of studs which extend from said cold side, said plurality of studs extend from within said forward cavity.

7. The liner as recited in claim 1, wherein said liner is mountable within a diffuser case module such that said aft cavity is immediately upstream of a Nozzle Guide Vane (NGV).

8. The liner as recited in claim 1, wherein said partition is generally rectilinear to define said aft cavity.

9. The liner as recited in claim 8, wherein a rail of said partition is adjacent to an edge of said liner.

10. A wall for use in a combustor of a gas turbine engine comprising:

a shell; and

a liner mountable to said shell to define a forward cavity and an aft cavity, said aft cavity operable at a pressure different than said forward cavity.

11. The wall as recited in claim 10, wherein said aft cavity is operable at a pressure greater than said forward cavity.

12. The wall as recited in claim 10, further comprising a plurality of studs which extend from said liner and through said shell.

13. The wall as recited in claim 10, further comprising a partition which extends from a cold side of said liner into contact with said shell to define said forward cavity and said aft cavity to define a respective pressure.

14. The wall as recited in claim 10, wherein said shell defines a plurality of Cooling impingement holes to communicate coolant into said forward cavity and said aft cavity.

15. The wall as recited in claim 10, wherein said shell defines a plurality of Cooling impingement holes to communicate coolant into said forward cavity and said aft cavity to generate a desired pressure within each.

16. The wall as recited in claim 10, wherein said liner is mountable within a diffuser case module such that said aft cavity is immediately upstream of a Nozzle Guide Vane (NGV) and directly adjacent a mount flange of said shell.

17. A method of controlling a thermal load in a combustor of a gas turbine engine comprising:

generating a pressure in an aft cavity of a combustor wall different than a pressure in a forward cavity of the combustor wall, the aft cavity immediately upstream of a Nozzle Guide Vane (NGV).

18. The method as recited in claim 17, further comprising:

generating a higher pressure in the aft cavity relative to said forward cavity.

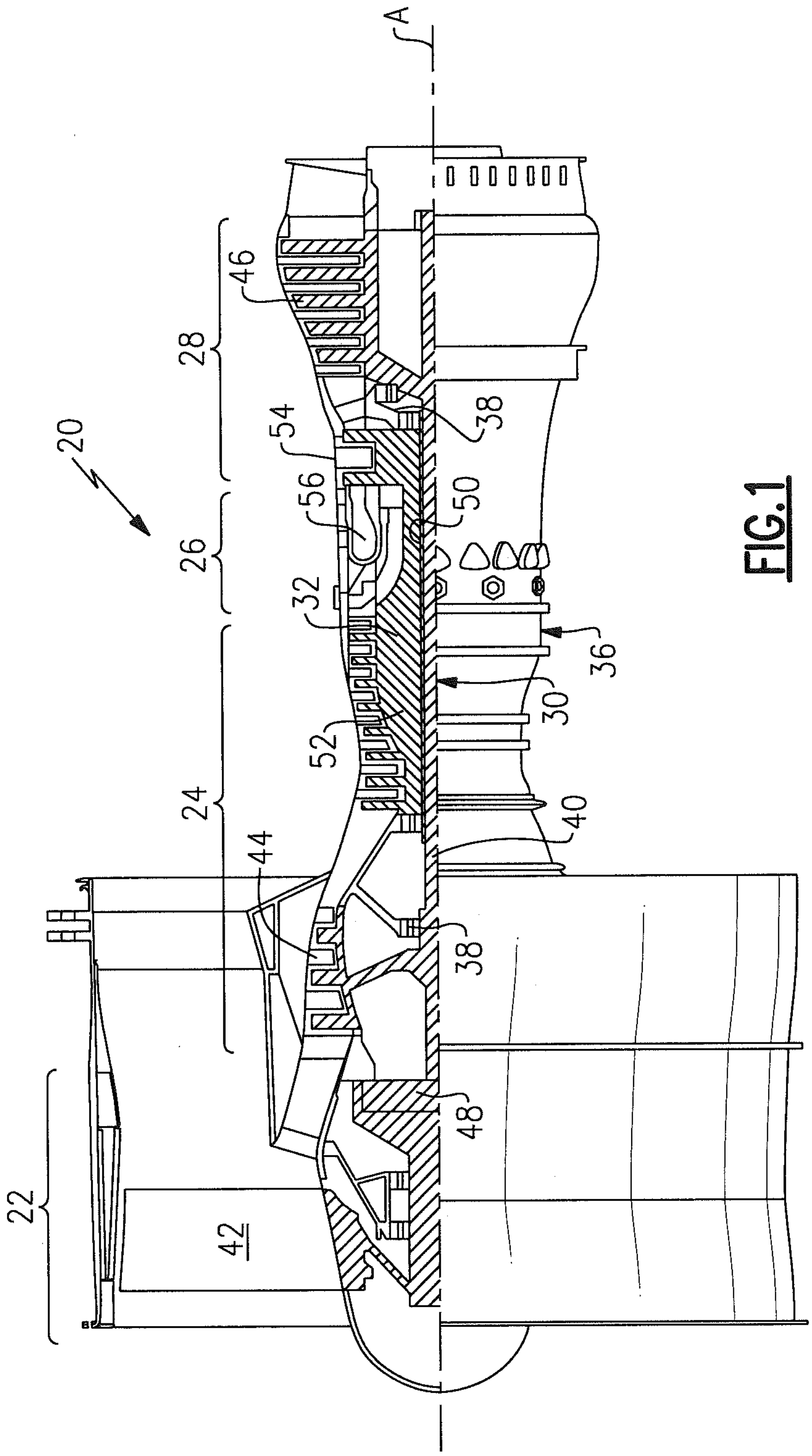

FIG.1

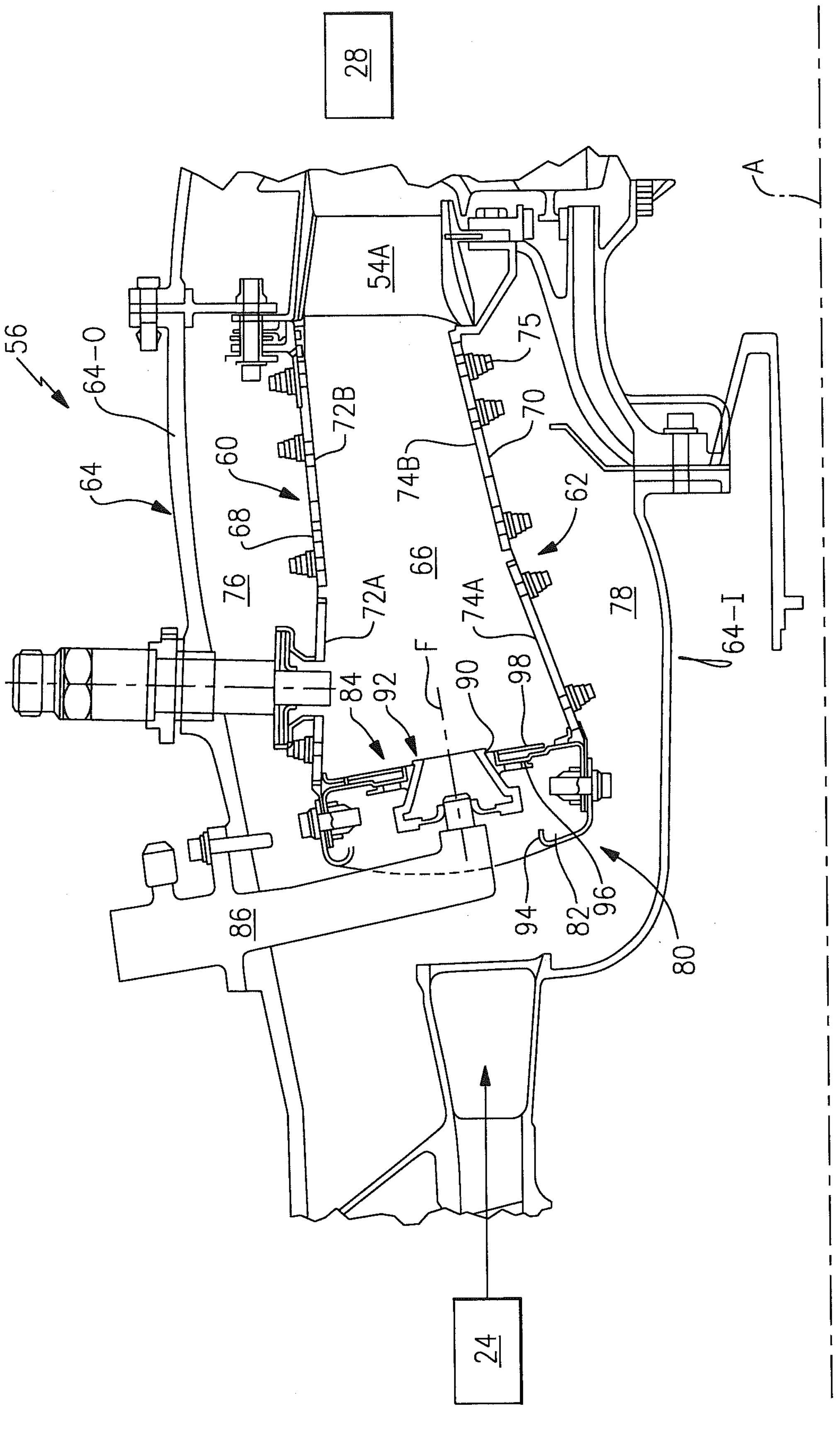
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FIG.2

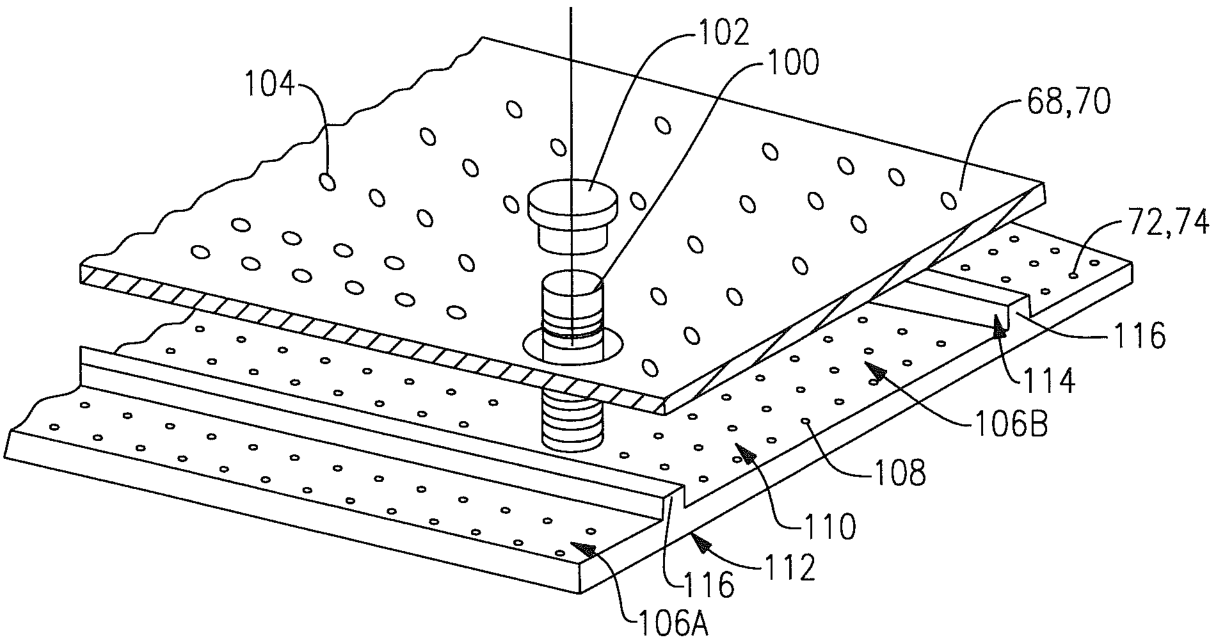

FIG.3

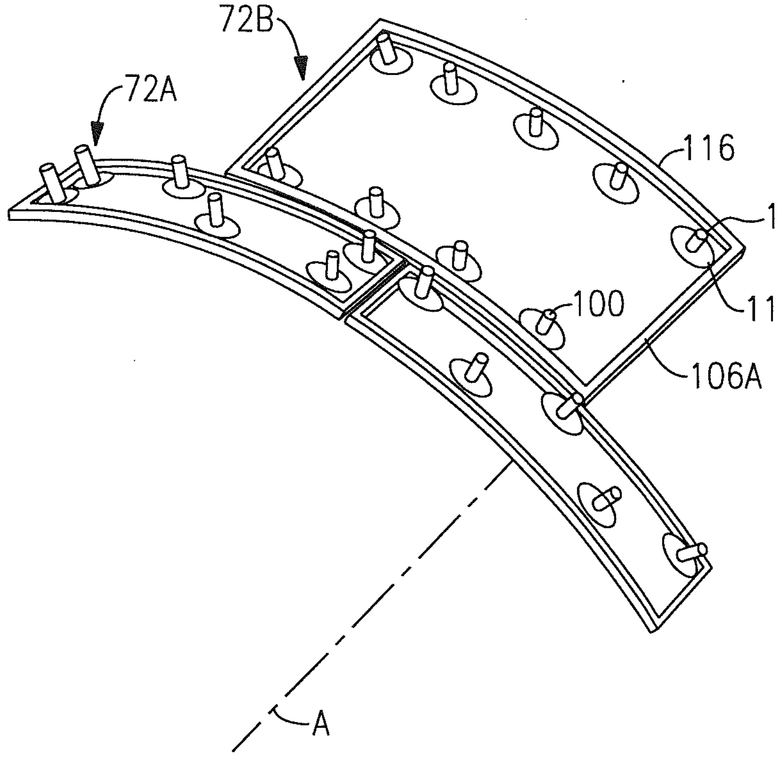

FIG.4

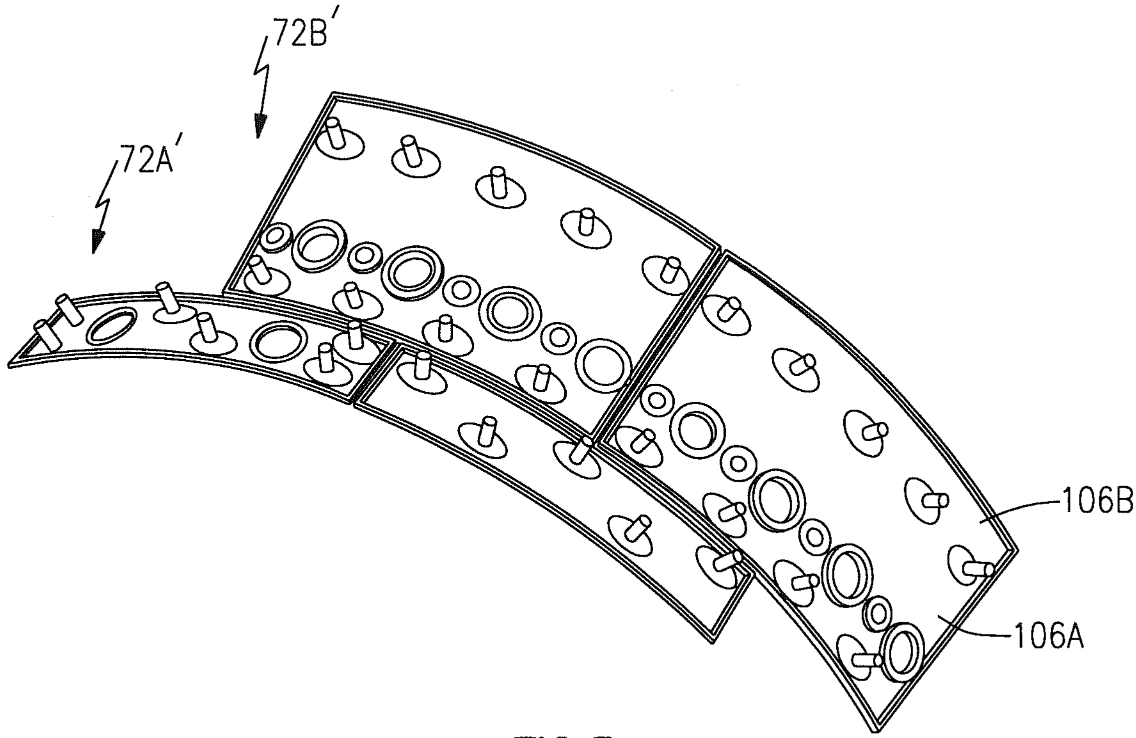

FIG.5

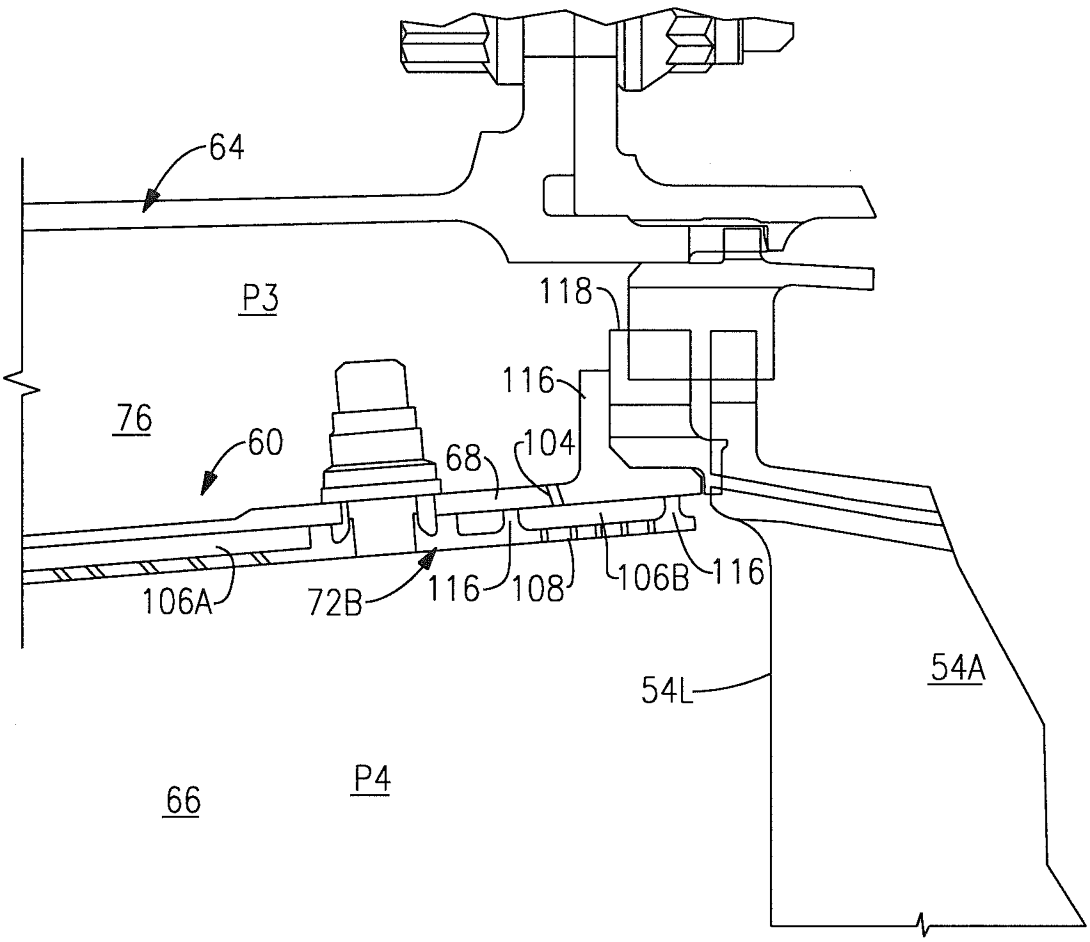

FIG.6

INTERNATIONAL SEARCH REPORT

International application No.
PCT/US2013/062661

A. CLASSIFICATION OF SUBJECT MATTER

F02C 7/22(2006.01)i, F23R 3/60(2006.01)i, F02C 7/00(2006.01)i

According to International Patent Classification (IPC) or to both national classification and IPC

B. FIELDS SEARCHED

Minimum documentation searched (classification system followed by classification symbols)
F02C 7/22; F02C 7/12; F02G 3/00; F02C 7/18; F23R 3/44; F01D 9/00; F02C 1/00; F23R 3/60; F02C 7/00

Documentation searched other than minimum documentation to the extent that such documents are included in the fields searched
Korean utility models and applications for utility models
Japanese utility models and applications for utility models

Electronic data base consulted during the international search (name of data base and, where practicable, search terms used)
eKOMPASS(KIPO internal) & keywords: gas turbine engine, combustor, liner, partition, cavity, shell, stud, and nozzle guide vane

C. DOCUMENTS CONSIDERED TO BE RELEVANT

Category*	Citation of document, with indication, where appropriate, of the relevant passages	Relevant to claim No.
X Y	US 2010-0095679 A1 (RUDRAPATNA et al.) 22 April 2010 See abstract, paragraphs [0025]-[0030],[0034]-[0036], and figures 2-5.	1-9 10-18
Y	US 7093439 B2 (PACHECO-TOUGAS et al.) 22 August 2006 See abstract, column 10, lines 37-49, and figures 1,9-10.	10-18
A	US 2009-0293488 A1 (COUGHLAN, III et al.) 03 December 2009 See abstract, paragraph [0019], and figures 1-1B.	1-18
A	US 2010-0095678 A1 (HAWIE et al.) 22 April 2010 See abstract, paragraphs [0002],[0019], and figures 2-3.	1-18
A	US 2011-0185739 A1 (BRONSON et al.) 04 August 2011 See abstract, claim 1, and figures 1-3.	1-18

☐ Further documents are listed in the continuation of Box C. ☒ See patent family annex.

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Date of the actual completion of the international search	Date of mailing of the international search report
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Information on patent family members

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