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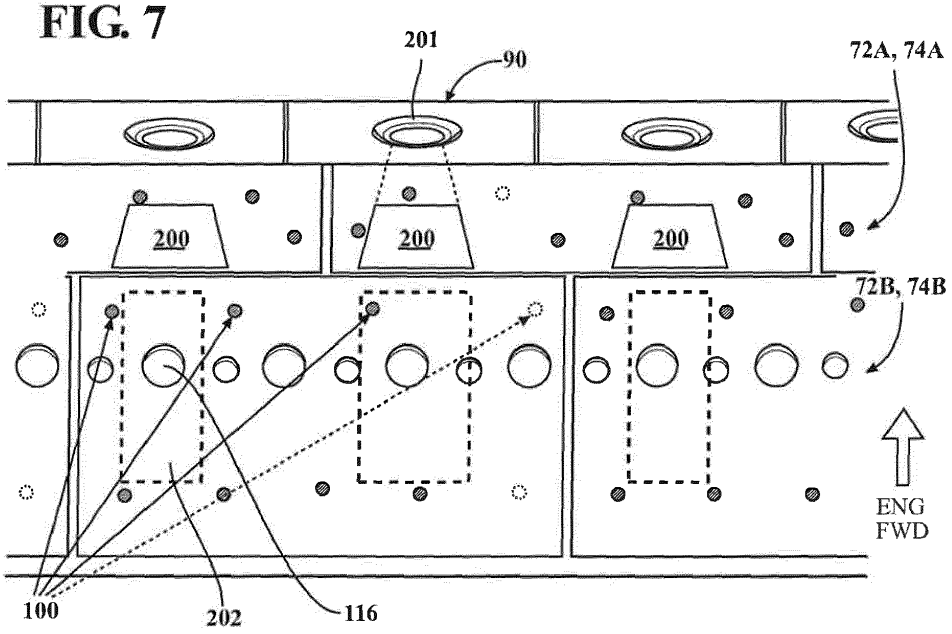
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(54) STUD ARRANGEMENT FOR GAS TURBINE ENGINE COMBUSTOR

(57) A liner panel (72A,B, 74A,B) for use in a combustor of a gas turbine engine, the liner panel (72A,B, 74A,B) includes a stud free zone (200) downstream of a combustor swirler (90). A combustor for a gas turbine engine, the combustor including a liner panel (72A,B, 74A,B) mounted to the support shell via a multiple of studs (100), the liner panel (72A,B, 74A,B) including a stud free zone (200) downstream of each respective combustor swirler (90), the stud free zone (200) including a multiple of film cooling holes.

FIG. 7



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Description

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 section to pressurize an airflow, a combustor section to burn a hydrocarbon fuel in the presence of the pressurized air, and a turbine section to extract energy from the resultant combustion gases.

[0003] Among the engine components, relatively high temperatures are observed in the combustor section such that cooling airflow is provided to meet desired service life requirements. The combustor section typically includes a combustion chamber formed by an inner and outer wall assembly. Each wall assembly includes a support shell lined with heat shields often referred to as liner panels.

[0004] In typical combustor chamber designs, combustor Impingement Film-Cooled Floatwall (IFF) liner panels are typically a curved flat surface on a hot side exposed to the gas path. The opposite, or cold side, has features such as cast in threaded studs to mount the liner panel and a full perimeter rail that contacts the inner surface of the respective liner shell. These features may result in durability issues.

SUMMARY

[0005] From a first aspect the invention provides a liner panel for use in a combustor of a gas turbine engine, the liner panel according to one disclosed non-limiting embodiment of the present disclosure can include a stud free zone downstream of a combustor swirler.

[0006] A further embodiment of the present disclosure may include, wherein the stud free zone is trapezoidal in shape.

[0007] A further embodiment of the present disclosure may include, wherein the stud free zone is defined by a forward liner panel.

[0008] A further embodiment of the present disclosure may include an aft liner panel aft of the forward liner panel.

[0009] A further embodiment of the present disclosure may include an aft stud free zone downstream of the forward liner panel stud free zone.

[0010] A further embodiment of the present disclosure may include at least one major diffusion aperture an aft stud free zone downstream of the forward liner panel stud free zone.

[0011] A further embodiment of the present disclosure may include, wherein the stud free zone is trapezoidal in shape and defined by a forward liner panel.

[0012] A further embodiment of the present disclosure may include, wherein the stud free zone is located toward an aft edge of the forward liner panel.

[0013] A further embodiment of the present disclosure may include, wherein the stud free zone is defined by a truncated triangle with a truncated apex located at combustor swirler.

[0014] A further embodiment of the present disclosure may include, wherein the stud free zone includes a multiple of film cooling holes.

[0015] The invention also provides a combustor for a gas turbine engine which includes a support shell and a liner panel mounted to the support shell via a multiple of studs, the liner panel including a stud free zone downstream of each respective combustor swirler, the stud free zone including a multiple of film cooling holes.

[0016] A further embodiment of the present disclosure may include a forward assembly including a bulkhead support shell, a bulkhead assembly mounted to the bulkhead support shell, and a multiple of the combustor swirlers mounted at least partially through the bulkhead assembly.

[0017] A further embodiment of the present disclosure may include, wherein the forward assembly is mounted to the support shell.

[0018] A further embodiment of the present disclosure may include a multiple of circumferentially distributed bulkhead liner panels secured to the bulkhead support shell around the swirler opening.

[0019] A further embodiment of the present disclosure may include, wherein the stud free zone is defined by a forward liner panel.

[0020] A further embodiment of the present disclosure may include an aft liner panel downstream of the forward liner panel, an aft stud free zone downstream of the forward liner panel stud free zone.

[0021] A method of directing airflow through a wall assembly within a combustor of a gas turbine engine according to another disclosed non-limiting embodiment of the present disclosure can include providing a stud free zone in a forward liner panel downstream of a combustor swirler, the stud free zone including a multiple of film cooling holes.

[0022] A further embodiment of the present disclosure may include locating a dilution passage within an aft stud free zone in an aft liner panel, the aft liner panel downstream of the forward liner panel.

[0023] A further embodiment of the present disclosure may include defining the stud free zone in the forward liner panel as a trapezoidal shape.

[0024] A further embodiment of the present disclosure may include defining the stud free zone in the forward liner panel as a truncated triangle with a truncated apex located adjacent to the combustion swirler.

[0025] The foregoing features and elements may be combined in various combinations without exclusivity, unless expressly indicated otherwise. These features and elements as well as the operation thereof will become more apparent in light of the following description and the accompanying drawings. It should be appreciated, however, the following description and drawings are in-

tended to be exemplary in nature and non-limiting.

BRIEF DESCRIPTION OF THE DRAWINGS

[0026] 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:

Figure 1 is a schematic cross-section of an example gas turbine engine architecture;
Figure 2 is an expanded longitudinal schematic sectional view of a combustor section according to one non-limiting embodiment that may be used with the example gas turbine engine architectures;
Figure 3 is an exploded partial sectional view of a portion of a combustor wall assembly;
Figure 4 is a perspective cold side view of a portion of a liner panel array;
Figure 5 is a perspective partial sectional view of a combustor;
Figure 6 is a sectional view of a portion of a combustor wall assembly; and
Figure 7 is a perspective view of a liner panel array.

DETAILED DESCRIPTION

[0027] 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 engine architectures might include an augmentor section among other systems or features. The fan section 22 drives air along a bypass flowpath and into the compressor section 24. The compressor section 24 drives air along a core flowpath for compression and communication into the combustor section 26, which then expands and directs the air through the turbine section 28. Although depicted as a turbofan in the disclosed non-limiting embodiment, it should be appreciated 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").

[0028] 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 ("LPC") 44 and a Low Pressure Turbine ("LPT") 46. 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.

[0029] The high spool 32 includes an outer shaft 50 that interconnects a High Pressure Compressor ("HPC") 52 and High Pressure Turbine ("HPT") 54. A combustor 56 is arranged between the HPC 52 and the HPT 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.

[0030] 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 LPT 46 and HPT 54 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 systems 38 within the static structure 36.

[0031] 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. 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 LPC 44 and LPT 46 and render increased pressure in a fewer number of stages.

[0032] A pressure ratio associated with the LPT 46 is pressure measured prior to the inlet of the LPT 46 as related to the pressure at the outlet of the LPT 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, the fan diameter is significantly larger than that of the LPC 44, and the LPT 46 has a pressure ratio that is greater than about five. It should be appreciated, 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.

[0033] In one embodiment, a significant amount of thrust is provided by the bypass flow path 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 (10,668m). 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.

[0034] 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 ("Tram" °R / 518.7°R)$^{0.5}$ (where °R = K x 9/5). 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).

[0035] With reference to Figure 2, the combustor section 26 generally includes a combustor 56 with an outer combustor wall assembly 60, an inner combustor wall assembly 62, and a diffuser case module 64. The outer combustor wall assembly 60 and the inner combustor wall assembly 62 are spaced apart such that a combustion chamber 66 is defined therebetween. The combustion chamber 66 is generally annular in shape to surround the engine central longitudinal axis A.

[0036] The outer combustor liner assembly 60 is spaced radially inward from an outer diffuser case 64A of the diffuser case module 64 to define an outer annular plenum 76. The inner combustor liner assembly 62 is spaced radially outward from an inner diffuser case 64B of the diffuser case module 64 to define an inner annular plenum 78. It should be appreciated that although a particular combustor is illustrated, other combustor types with various combustor liner arrangements will also benefit herefrom. It should be further appreciated that the disclosed cooling flow paths are but an illustrated embodiment and should not be limited only thereto.

[0037] The combustor wall assemblies 60, 62 contain the combustion products for direction toward the turbine section 28. Each combustor wall assembly 60, 62 generally includes a respective support shell 68, 70 which supports one or more liner panels 72, 74 mounted thereto arranged to form a liner array. The support shells 68, 70 may be manufactured by, for example, the hydroforming of a sheet metal alloy to provide the generally cylindrical outer shell 68 and inner shell 70. Each of the liner panels 72, 74 may be generally rectilinear with a circumferential arc. The liner panels 72, 74 may be manufactured of, for example, a nickel based super alloy, ceramic or other temperature resistant material. In one disclosed non-limiting embodiment, the liner array includes a multiple of forward liner panels 72A and a multiple of aft liner panels 72B that are circumferentially staggered to line the outer shell 68. A multiple of forward liner panels 74A and a multiple of aft liner panels 74B are circumferentially staggered to line the inner shell 70.

[0038] 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 a cowl 82, a bulkhead assembly 84, and a multiple of swirlers 90 (one shown). Each of the swirlers 90 is circumferentially aligned with one of a multiple of fuel nozzles 86 (one shown) and the respective hood ports 94 to project through the bulkhead assembly 84.

[0039] The bulkhead assembly 84 includes a bulkhead support shell 96 secured to the combustor walls 60, 62, and a multiple of circumferentially distributed bulkhead liner panels 98 secured to the bulkhead support shell 96 around the swirler opening. The bulkhead support shell 96 is generally annular and the multiple of circumferentially distributed bulkhead liner panels 98 are segmented, typically one to each fuel nozzle 86 and swirler 90.

[0040] The cowl 82 extends radially between, and is secured to, the forwardmost ends of the combustor walls 60, 62. The cowl 82 includes a multiple of circumferentially distributed hood ports 94 that receive one of the respective multiple of fuel nozzles 86 and facilitates the direction of compressed air into the forward end of the combustion chamber 66 through a swirler 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 swirler opening 92 within the respective swirler 90.

[0041] The forward assembly 80 introduces core combustion air into the forward section 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 adjacent structure generate a blended fuel-air mixture that supports stable 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. 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.

[0043] With reference to Figure 3, a multiple of studs 100 extend from each of the liner panels 72, 74 so as to permit a liner array (partially shown in Figure 4) of the liner panels 72, 74 to be mounted to their respective support shells 68, 70 with fasteners 102 such as nuts. That is, the studs 100 project rigidly from the liner panels 72, 74 to extend through the respective support shells 68, 70 and receive the fasteners 102 on a threaded section thereof (Figure 5).

[0044] A multiple of cooling impingement passages 104 penetrate through the support shells 68, 70 to allow air from the respective annular plenums 76, 78 to enter cavities 106 formed in the combustor walls 60, 62 between the respective support shells 68, 70 and liner panels 72, 74. The impingement passages 104 are generally normal to the surface of the liner panels 72, 74. The air in the cavities 106 provides cold side impingement cooling of the liner panels 72, 74 that is generally defined herein as heat removal via internal convection.

[0045] A multiple of effusion passages 108 penetrate through each of the liner panels 72, 74. The geometry of

the passages, e.g., diameter, shape, density, surface angle, incidence angle, etc., as well as the location of the passages with respect to the high temperature combustion flow also contributes to effusion cooling. The effusion passages 108 allow the air to pass from the cavities 106 defined in part by a cold side 110 of the liner panels 72, 74 to a hot side 112 of the liner panels 72, 74 and thereby facilitate the formation of a thin, relatively cool, film of cooling air along the hot side 112.

[0046] In one disclosed non-limiting embodiment, each of the multiple of effusion passages 108 are typically 0.025" (0.635 mm) in diameter and define a surface angle of about thirty (30) degrees with respect to the cold side 110 of the liner panels 72, 74. The effusion passages 108 are generally more numerous than the impingement passages 104 and promote film cooling along the hot side 112 to sheath the liner panels 72, 74 (Figure 6). Film cooling as defined herein is the introduction of a relatively cooler air at one or more discrete locations along a surface exposed to a high temperature environment to protect that surface in the region of the air injection as well as downstream thereof.

[0047] The combination of impingement passages 104 and effusion passages 108 may be referred to as an Impingement Film Floatwall (IFF) assembly. A multiple of dilution passages 116 are located in the liner panels 72, 74 each along a common axis D. For example only, the dilution passages 116 are located in a circumferential line W (shown partially in Figure 4). Although the dilution passages 116 are illustrated in the disclosed non-limiting embodiment as within the aft liner panels 72B, 74B, the dilution passages may alternatively be located in the forward liner panels 72A, 72B or in a single liner panel which replaces the fore/aft liner panel array. Further, the dilution passages 116 although illustrated in the disclosed non-limiting embodiment as integrally formed in the liner panels, it should be appreciated that the dilution passages 116 may be separate components. Whether integrally formed or separate components, the dilution passages 116 may be referred to as grommets.

[0048] With reference to Figure 4, in one disclosed non-limiting embodiment, each of the liner panels 72A, 72B, 74A, 74B in the liner panel array includes a perimeter rail 120 formed by a forward circumferential rail 122, an aft circumferential rail 124, and axial rails 126A, 126B, that interconnect the forward and aft circumferential rail 122, 124. The perimeter rail 120 seals each liner panel with respect to the respective support shell 68, 70 to form the impingement cavity 106 therebetween. That is, the forward and aft circumferential rail 122, 124 are located at relatively constant curvature shell interfaces while the axial rails 126 extend across an axial length of the respective support shell 68, 70 to complete the perimeter rail 120 that seals the liner panels 72, 74 to the respective support shell 68, 70.

[0049] A multiple of studs 100 are located adjacent to the respective forward circumferential rail 122 and the aft circumferential rail 124. Each of the studs 100 may be at least partially surrounded by posts 130 to at least partially support the fastener 102 and provide a standoff between each liner panels 72B, 74B and respective support shell 68, 70.

[0050] With reference to Figure 7, the quantity and location of the multiple of studs 100 is typically based on structural analysis and symmetry of the studs 100 relative to the liner to facilitate proper sealing of the panel rail to the inner combustor shell. The conventional position would often locate one or more of the multiple of studs 100 downstream of the combustor swirlers 90. As this area may have relatively high metal temperatures, durability issues may result from the lack of effusion cooling as this issue may be more significant at the aft section of the forward row of liners.

[0051] In one embodiment, the multiple of studs 100 of the forward liner panels 72A, 74A are not located within a stud free zone 200 defined downstream of each of the combustion swirlers 90. Each stud free zone 200 is defined as an essentially truncated triangular shape.

[0052] In one embodiment, the stud free zone 200 is defined by a truncated triangle with a truncated apex 201 located at the combustor swirler 90. In other words, the stud free zone 200 is a trapezoidal shaped zone located at the aft edge of the forward liner panels 72A, 74A. That is, to increase durability, the studs 100 are specifically moved away from each zone 200 directly aft of the respective combustor swirlers 90 in the aft section of the forward liner panels 72A, 74A as this area has the relatively hottest surface metal temperatures. The stud free zone 200 facilitates a more efficient distribution of film cooling holes in the hottest areas of the segment as the studs no longer hinder location of film cooling holes 108 (Figure 3). In one example, the stud free zone 200 extends from about 0.75 inches (19 mm) to 1.7 inches (43 mm) from the respective combustor swirler 90 and the sides between the fore and aft lines are at about 20 degrees.

[0053] For the aft liner panels 72B, 74B, the forward most row of studs 100A are also intentionally moved out away from an aft liner panel stud free zone 202 downstream of the stud free zone 200 and the respective combustor swirler 90. As with the forward liner panels 72A, 74A, this permits a more advantageous distribution of cooling holes around the area of the liner segments which typically have the hottest metal temperatures. Further, at least one dilution passage 116 may be located within the aft liner panel stud free zone 202. The stud free zone 200 in the aft liner panels 72B, 74B, defines a rectangle of about 1.5 inches (38 mm) by 2.8 inches (71mm) and is located 1.8 inches (45mm) behind the respective combustor swirlers 90.

[0054] The use of the terms "a" and "an" and "the" and similar references in the context of description (especially in the context of the following claims) are to be construed to cover both the singular and the plural, unless otherwise indicated herein or specifically contradicted by context. The modifier "about" used in connection with a quantity

is inclusive of the stated value and has the meaning dictated by the context (e.g., it includes the degree of error associated with measurement of the particular quantity). All ranges disclosed herein are inclusive of the endpoints, and the endpoints are independently combinable with each other. It should be appreciated 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.

[0055] Although the different non-limiting embodiments have specific illustrated components, the embodiments of this invention are not limited to those particular combinations. It is possible to use some of the components or features from any of the non-limiting embodiments in combination with features or components from any of the other non-limiting embodiments.

[0056] It should be appreciated that like reference numerals identify corresponding or similar elements throughout the several drawings. It should also be appreciated that although a particular component arrangement is disclosed in the illustrated embodiment, other arrangements will benefit herefrom.

[0057] Although particular step sequences are shown, described, and claimed, it should be appreciated that steps may be performed in any order, separated or combined unless otherwise indicated and will still benefit from the present disclosure.

[0058] 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 appreciated 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

1. A liner panel (72, 74) for use in a combustor (56) of a gas turbine engine (20), the liner panel (72, 74) comprising a stud free zone (200) downstream of a combustor swirler (90).

2. A combustor (56) for a gas turbine engine (20) comprising:

   a support shell (68, 70); and
   a liner panel (72, 74) as recited in claim 1 mounted to the support shell (68, 70) via a multiple of studs (100), the liner panel (72,74) including a stud free zone (200) downstream of each respective combustor swirler (90).

3. The combustor (56) or liner panel (72, 74) as recited in claim 1 or 2, wherein the stud free zone (200) is trapezoidal in shape.

4. The combustor (56) or liner panel (72, 74) as recited in any preceding claim, wherein the stud free zone (200) is defined by a forward liner panel (72A, 74A).

5. The combustor (56) or liner panel (72, 74) as recited in claim 4, wherein the stud free zone (200) is located toward an aft edge of the forward liner panel (72A, 74A).

6. The combustor (56) or liner panel (72, 74) as recited in claim 4 or 5, further comprising an aft liner panel (72B, 74B) aft of the forward liner panel (72A, 74A).

7. The combustor (56) or liner panel (72, 74) as recited in claim 6, further comprising an aft stud free zone (202) downstream of the forward liner panel stud free zone (200).

8. The combustor (56) or liner panel (72, 74) as recited in claim 7, further comprising at least one major diffusion aperture (116) in aft stud free zone (202) downstream of the forward liner panel stud free zone (200).

9. The combustor (56) or liner panel (72, 74) as recited in any preceding claim, wherein the stud free zone (200) is defined by a truncated triangle with a truncated apex (201) located at combustor swirler (90).

10. The combustor (56) or liner panel (72, 74) as recited in any preceding claim, wherein the stud free zone (200) includes a multiple of film cooling holes (108).

11. The combustor (56) as recited in any of claims 2 to 10, further comprising a forward assembly (80) including a bulkhead support shell (96), a bulkhead assembly (84) mounted to the bulkhead support shell (96), and a multiple of the combustor swirlers (90) mounted at least partially through the bulkhead assembly (84).

12. The combustor (56) as recited in claim 11, wherein the forward assembly (80) is mounted to the support shell (68, 70).

13. The combustor (56) as recited in claim 11 or 12, further comprising a multiple of circumferentially distributed bulkhead liner panels (98) secured to the bulkhead support shell (96) around a swirler opening (92).

14. A method of directing airflow through a wall assembly (60, 62) within a combustor (56) of a gas turbine engine (20), comprising providing a stud free zone

(200) in a forward liner panel (72A, 74A) downstream of a combustor swirler (90), the stud free zone (200) including a multiple of film cooling holes (108).

15. The method as recited in claim 14, further comprising locating a dilution passage (116) within an aft stud free zone (202) in an aft liner panel (72B, 74B), the aft liner panel (72B, 74B) downstream of the forward liner panel (72A, 74A).

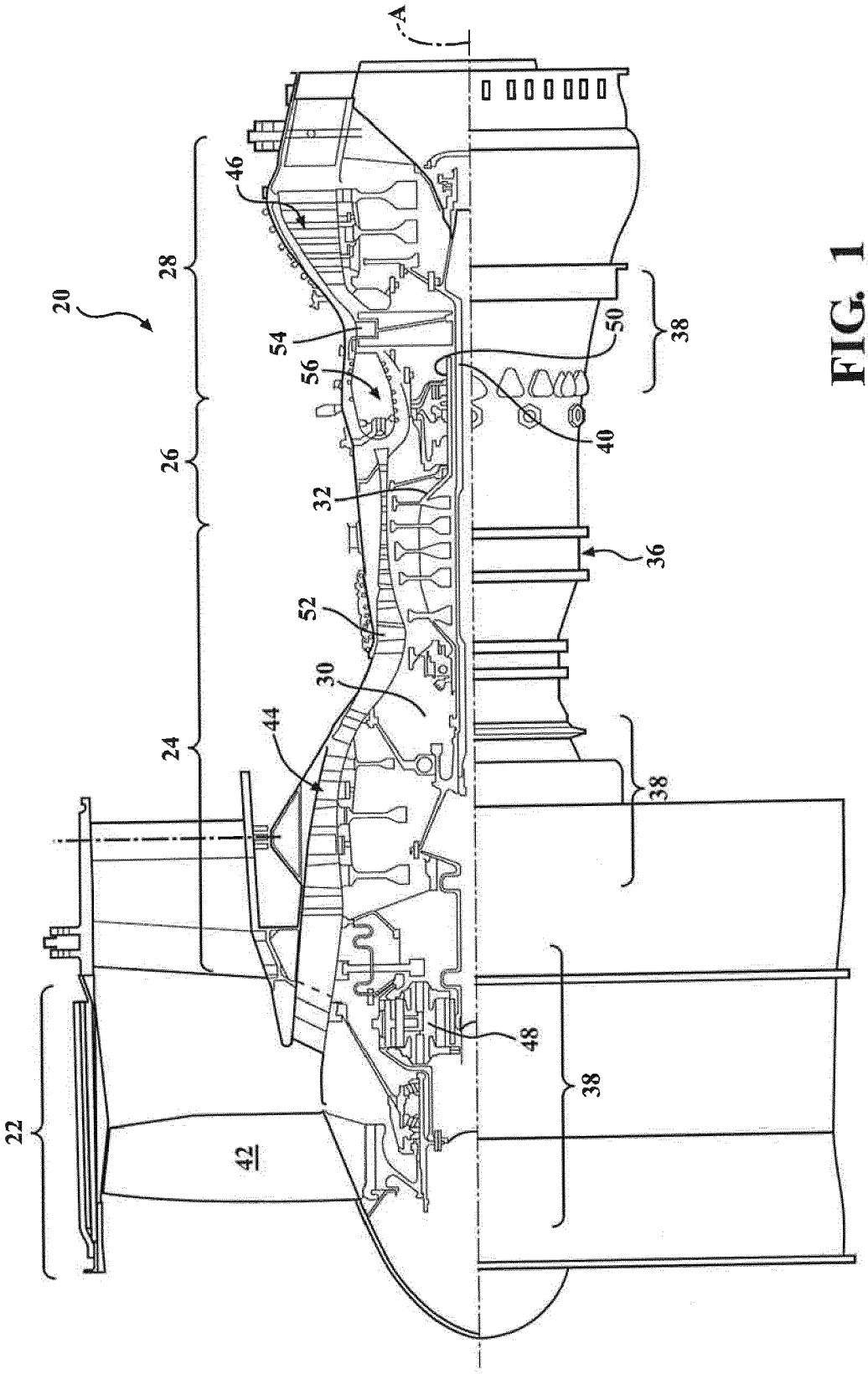


FIG. 1

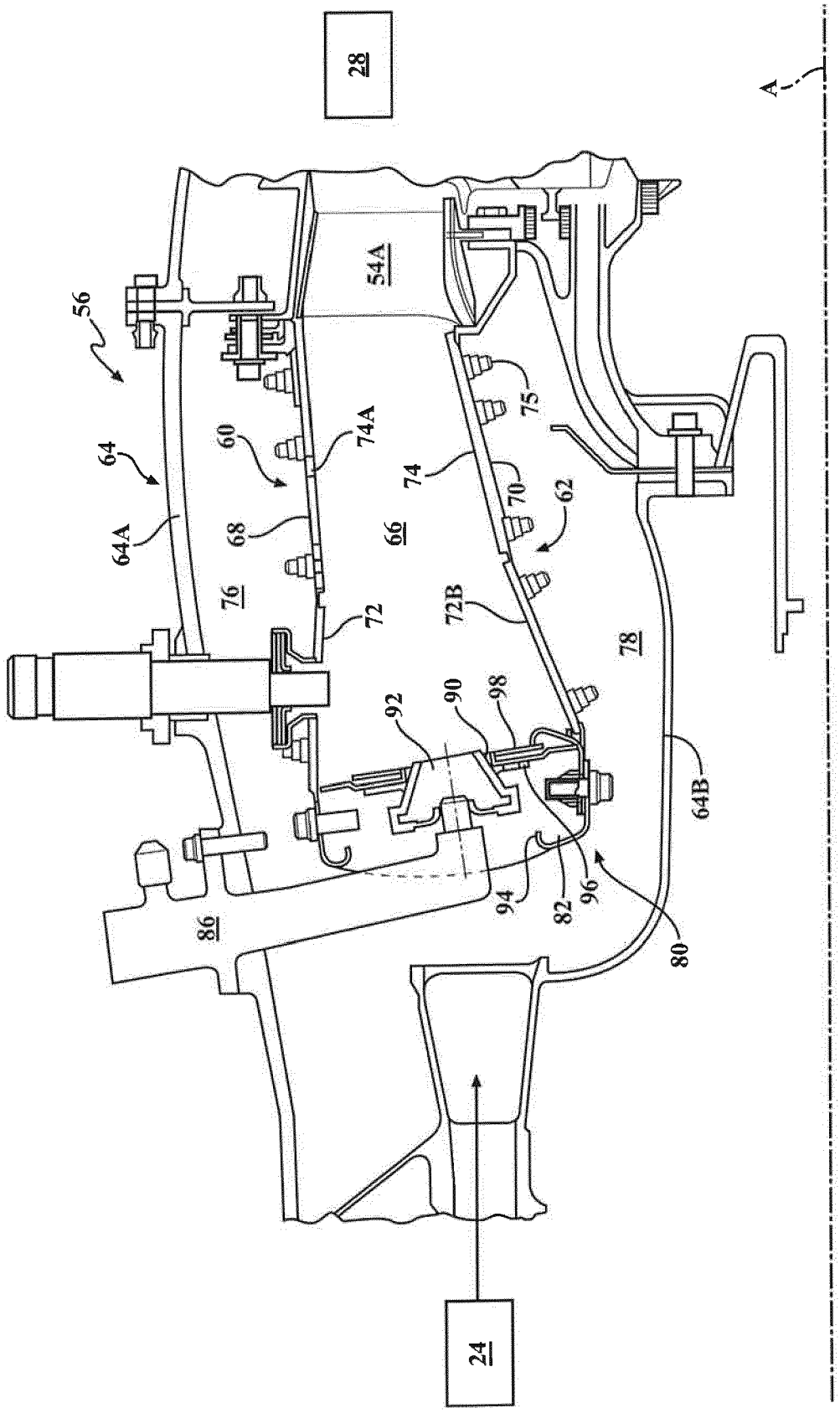

28
A
54A
56
64
64A
60
74A
74
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96
80
86
24

FIG. 2

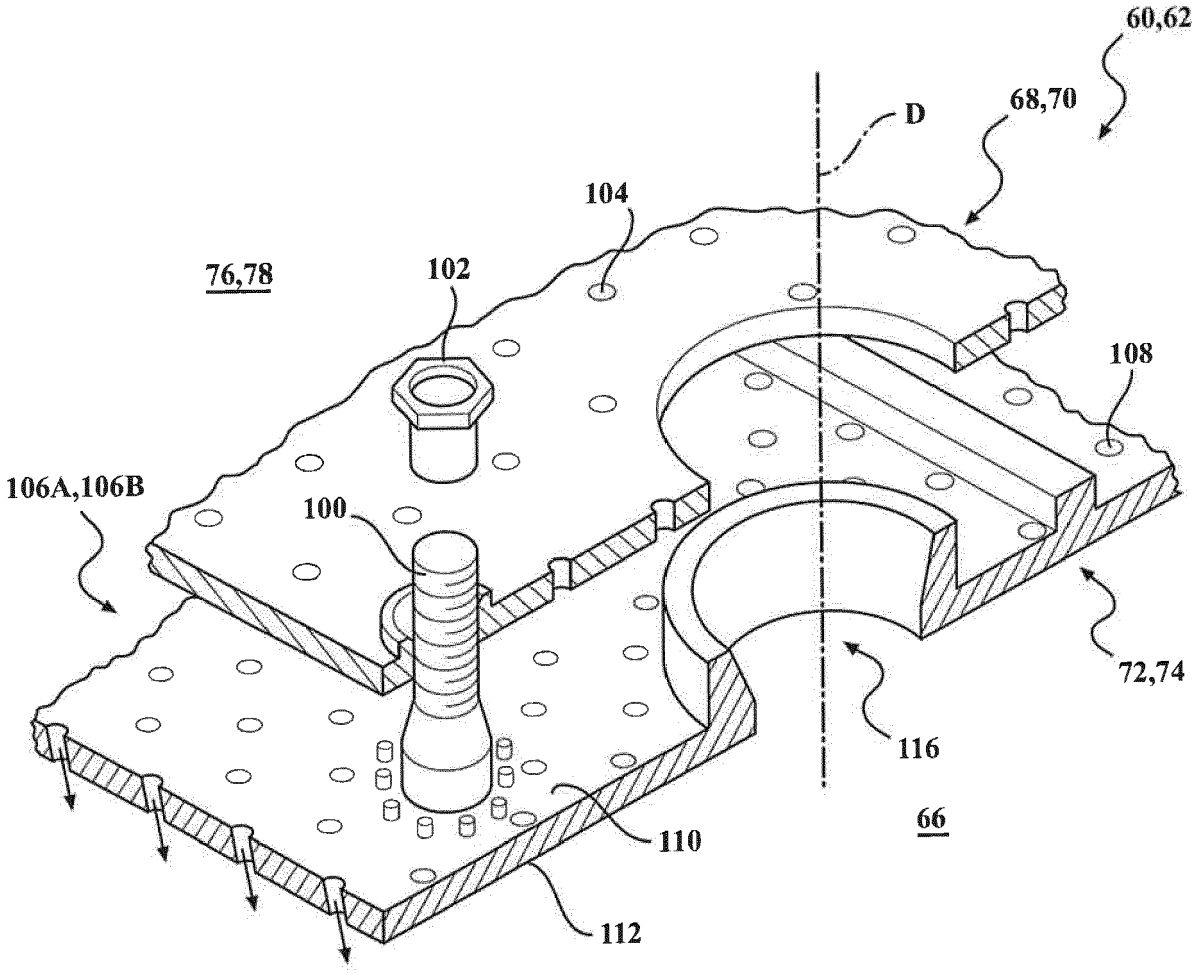


FIG. 3

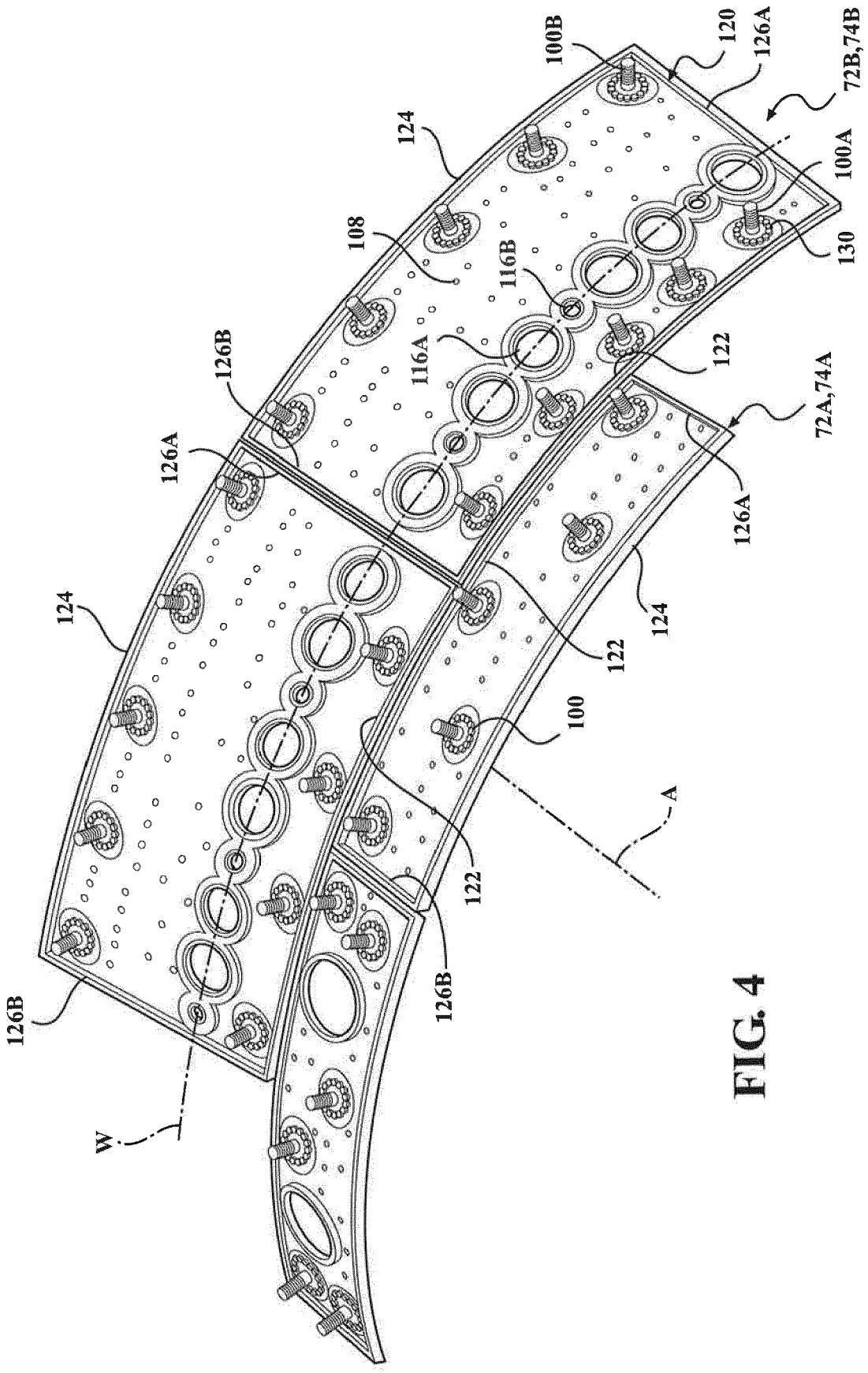

100B
120
126A
72B,74B
124
100A
108
130
116B
122
126B
116A
72A,74A
126A
126A
124
124
122
100
A
122
126B
126B
W

FIG. 4

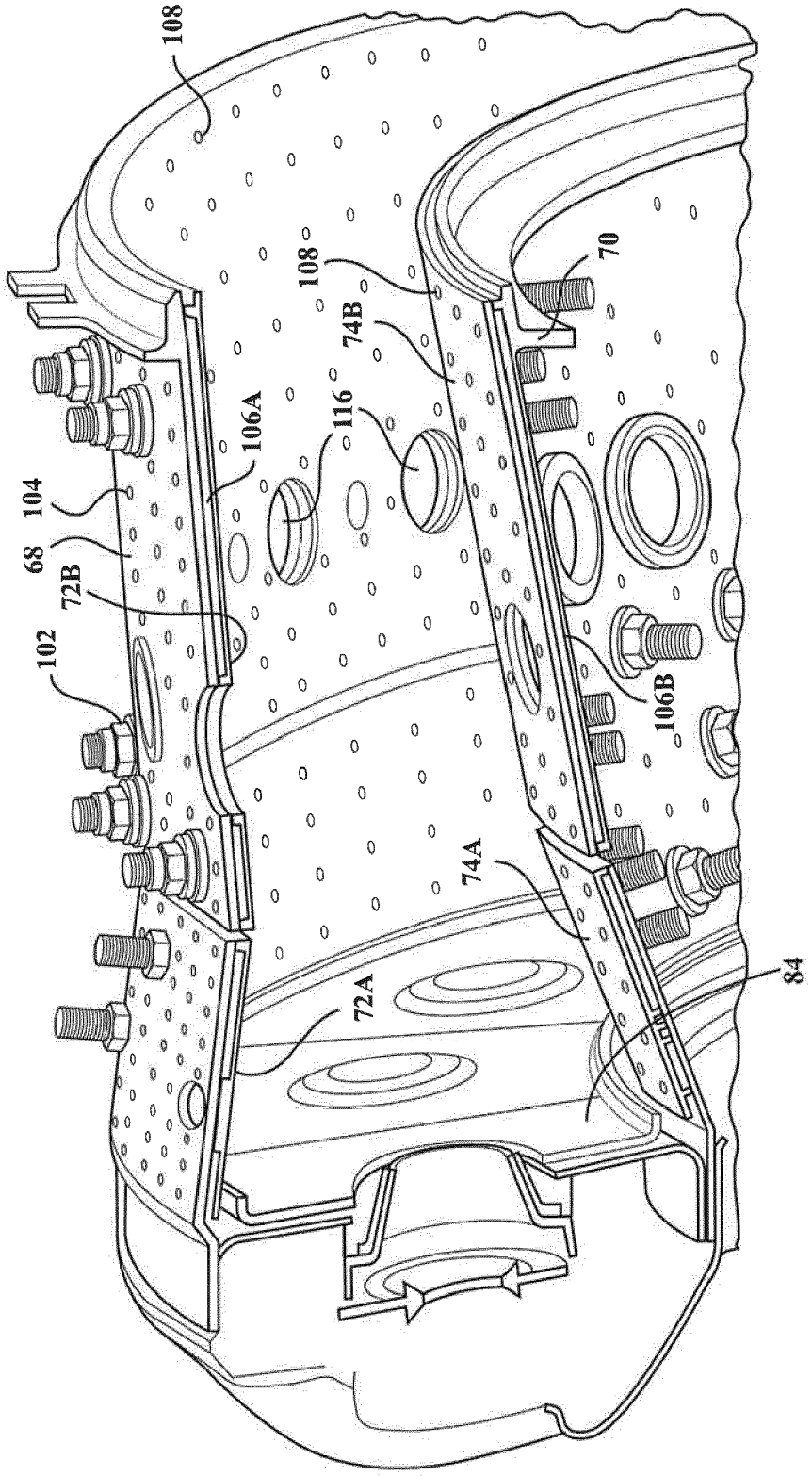


FIG. 5

FIG. 6

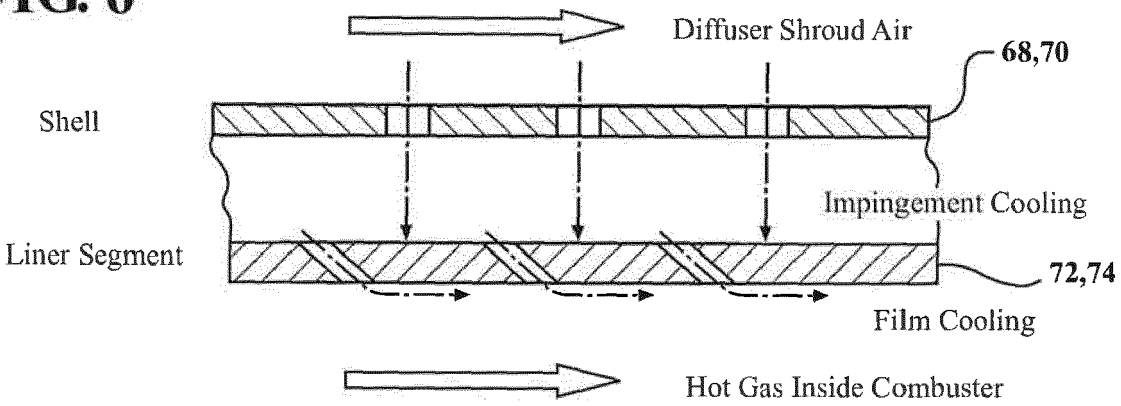
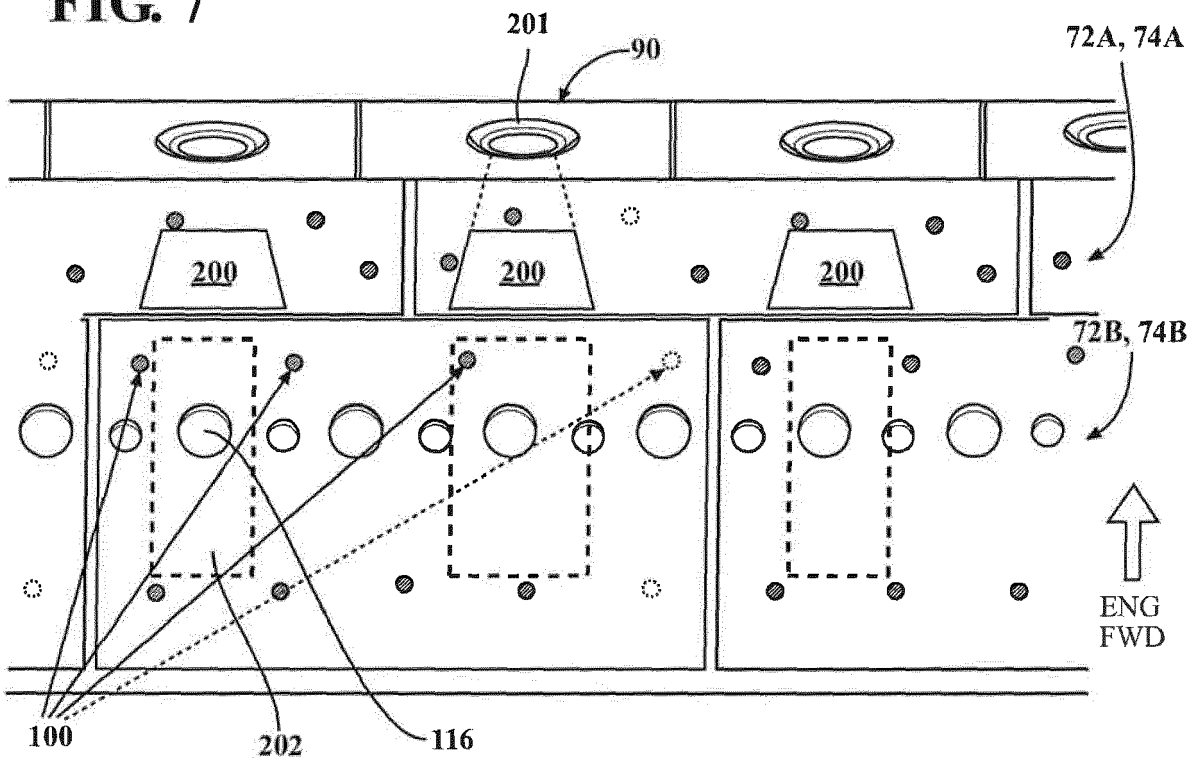

Diffuser Shroud Air
68,70
Shell
Impingement Cooling
Liner Segment
72,74
Film Cooling
Hot Gas Inside Combuster

FIG. 7


201
90
72A, 74A
200
200
200
72B, 74B
ENG
FWD
100
202
116

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EUROPEAN SEARCH REPORT

Application Number
EP 17 20 0015

DOCUMENTS CONSIDERED TO BE RELEVANT

Category	Citation of document with indication, where appropriate, of relevant passages	Relevant to claim	CLASSIFICATION OF THE APPLICATION (IPC)
X	US 2015/362192 A1 (CUNHA FRANK J [US] ET AL) 17 December 2015 (2015-12-17) * paragraph [0049] - paragraph [0074]; figures 2-15 *	1-15	INV. F23R3/00
X	US 2010/095680 A1 (RUDRAPATNA NAGARAJA S [US] ET AL) 22 April 2010 (2010-04-22) * paragraph [0022] - paragraph [0031]; figures 1-6 *	1-5,9-14	
X	US 4 614 082 A (STERMAN ALBERT P [US] ET AL) 30 September 1986 (1986-09-30) * figures 1-9 *	1-7,9-14	
X	US 2010/095679 A1 (RUDRAPATNA NAGARAJA S [US] ET AL) 22 April 2010 (2010-04-22) * paragraph [0022] - paragraph [0035]; figures 1-7 *	1-15	
A	US 2006/277921 A1 (PATEL BHAWAN [CA] ET AL) 14 December 2006 (2006-12-14) * paragraph [0015] - paragraph [0024]; figures 1-4 *	1,14	TECHNICAL FIELDS SEARCHED (IPC) F23R

The present search report has been drawn up for all claims

Place of search	Date of completion of the search	Examiner
Munich	26 February 2018	Theis, Gilbert

CATEGORY OF CITED DOCUMENTS

X : particularly relevant if taken alone
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ANNEX TO THE EUROPEAN SEARCH REPORT ON EUROPEAN PATENT APPLICATION NO. EP 17 20 0015

This annex lists the patent family members relating to the patent documents cited in the above-mentioned European search report. The members are as contained in the European Patent Office EDP file on
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26-02-2018

Patent document cited in search report	Publication date	Patent family member(s)	Publication date
US 2015362192 A1	17-12-2015	EP 2946092 A1	25-11-2015
		US 2015362192 A1	17-12-2015
		WO 2014113007 A1	24-07-2014
US 2010095680 A1	22-04-2010	EP 2180256 A2	28-04-2010
		US 2010095680 A1	22-04-2010
US 4614082 A	30-09-1986	NONE	
US 2010095679 A1	22-04-2010	NONE	
US 2006277921 A1	14-12-2006	CA 2546881 A1	10-12-2006
		US 2006277921 A1	14-12-2006

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For more details about this annex : see Official Journal of the European Patent Office, No. 12/82