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(54) **METHOD AND APPARATUS FOR ASSEMBLING GAS TURBINE ENGINE COMBUSTORS**

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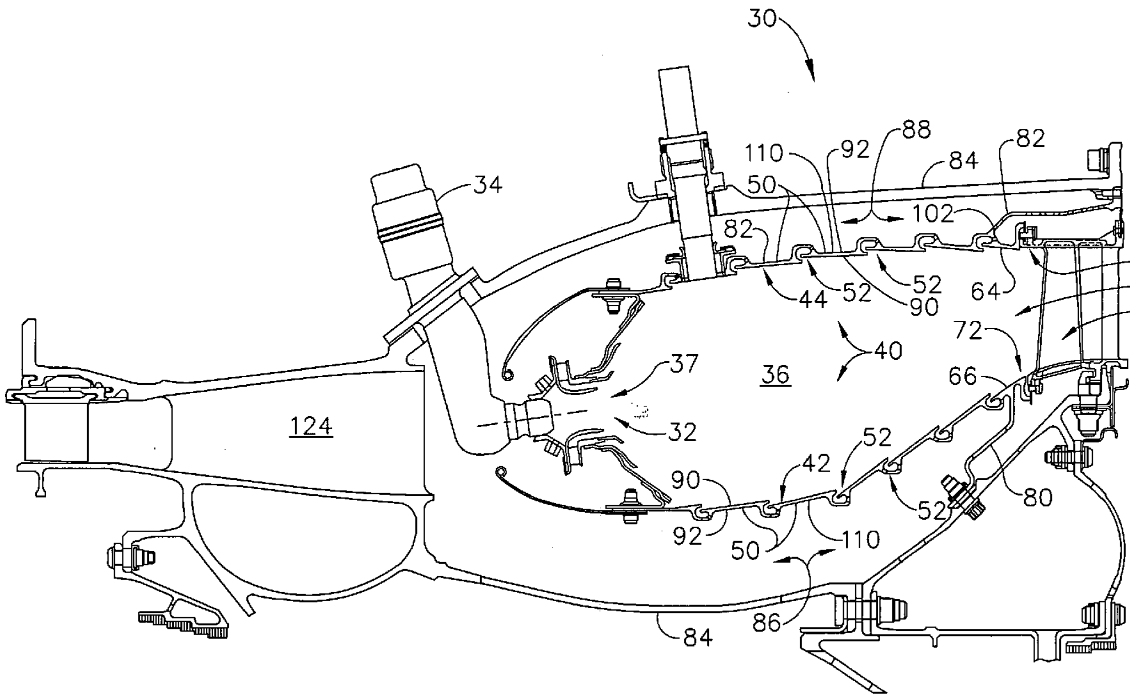
(57) **ABSTRACT**

A method enables the operation of a gas turbine engine. The method comprises channeling airflow into a cooling passageway defined between the combustor casing and an inner liner of the combustor, wherein the inner liner is fabricated from a plurality of panels coupled together, channeling airflow into a cooling passageway defined between the combustor casing and an outer liner of the combustor; wherein the outer liner is fabricated from a plurality of panels coupled together, and channeling dilution airflow into a combustion chamber defined between the inner and outer liners, through a plurality of openings formed within at least one panel within at least one of the inner liner panels and the outer liner panels, wherein the plurality of openings are non-circular.



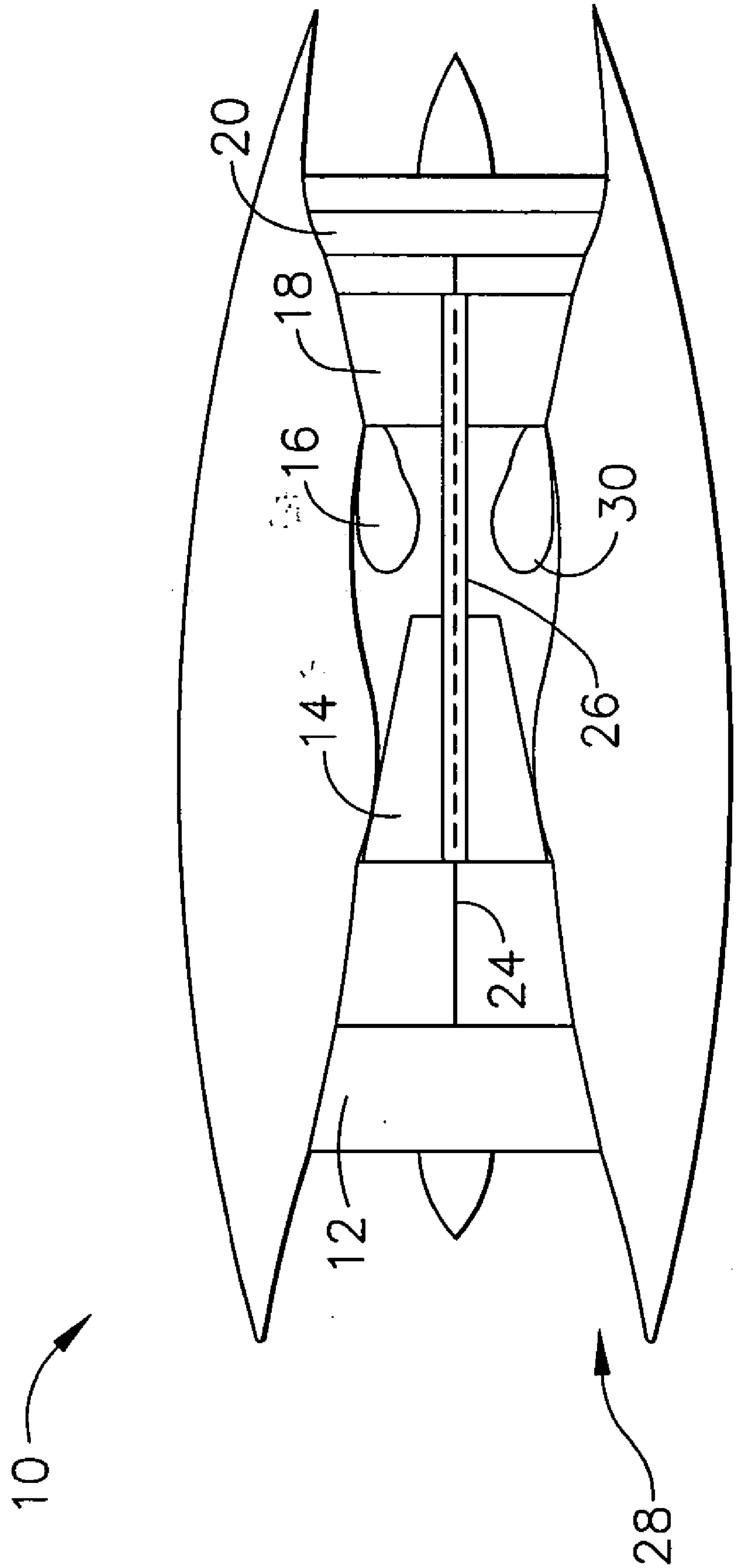


FIG. 1

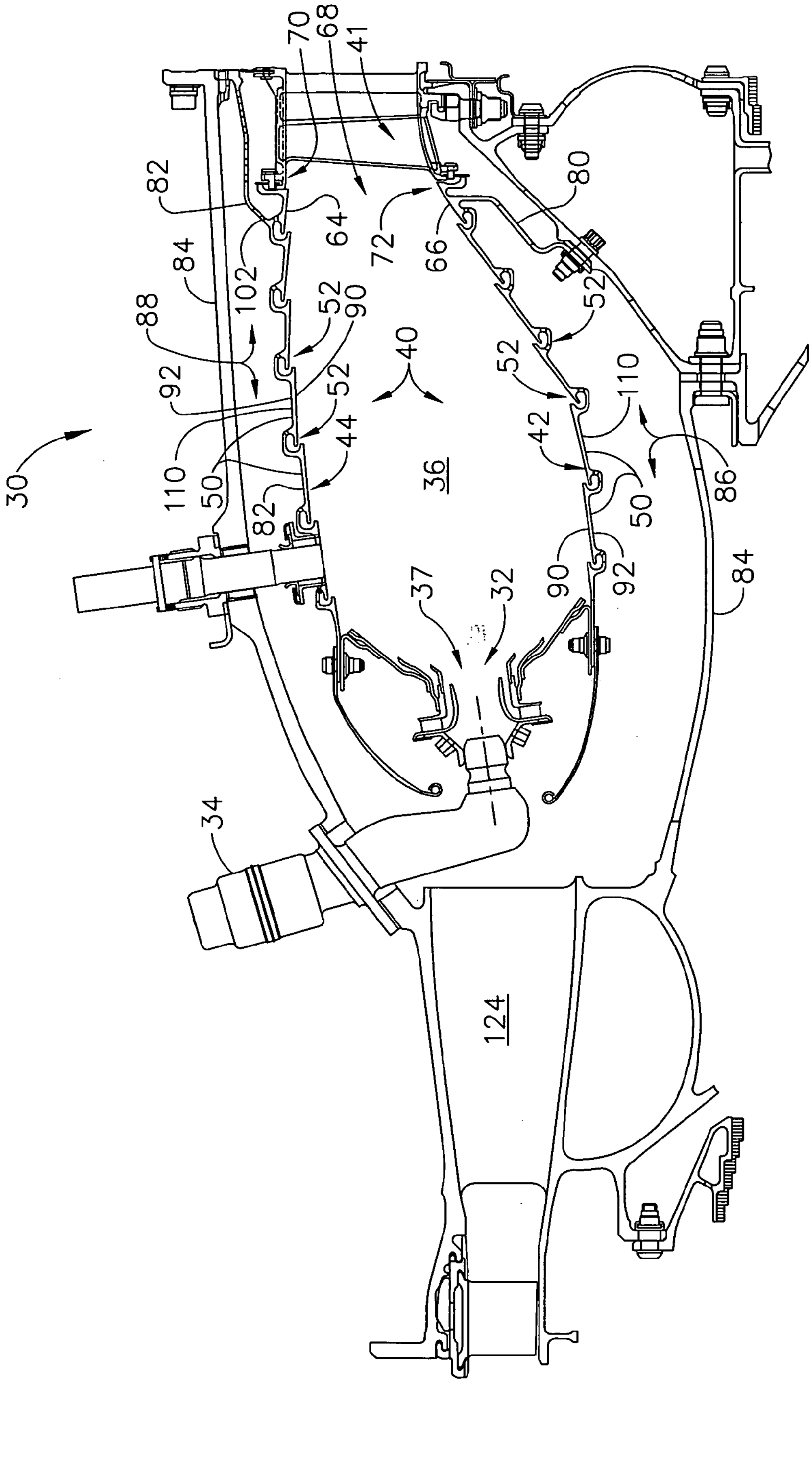

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FIG. 2

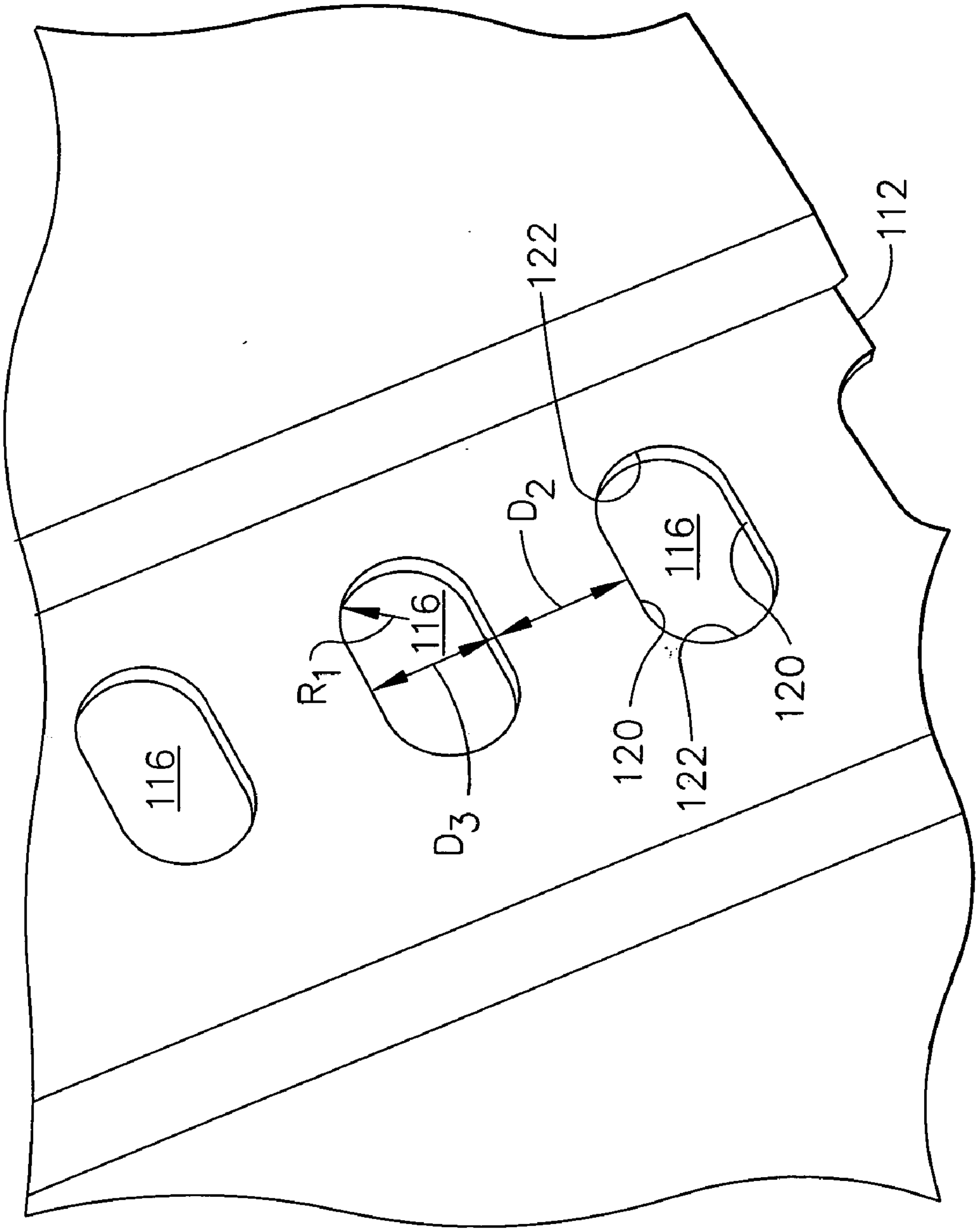

FIG. 3

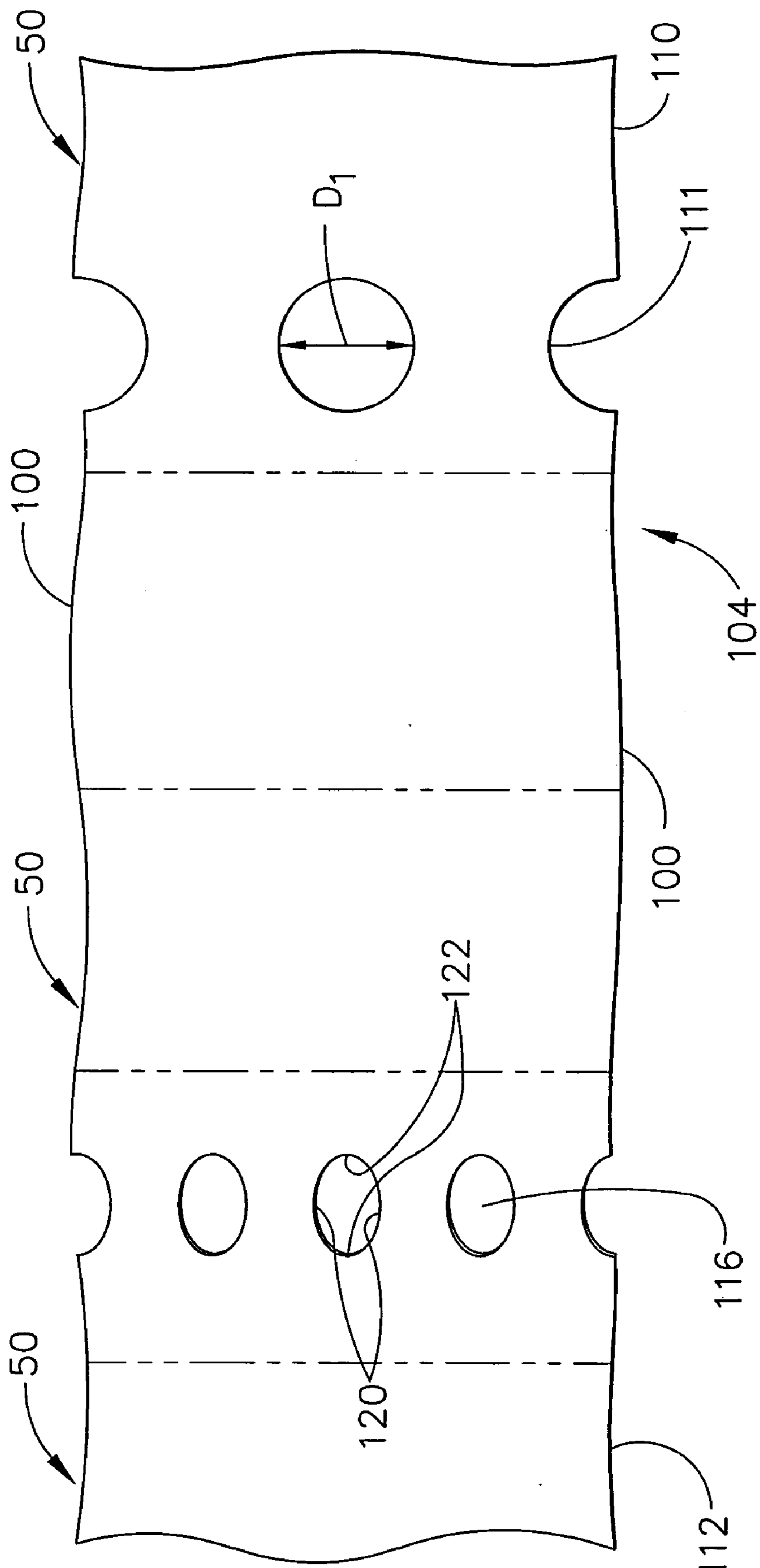

FIG. 4

METHOD AND APPARATUS FOR ASSEMBLING GAS TURBINE ENGINE COMBUSTORS

BACKGROUND OF THE INVENTION

[0001] This invention relates generally to combustors and, more particularly to a method and apparatus for decreasing combustor acoustics.

[0002] At least some known gas turbine engines include a compressor for compressing air which is suitably mixed with a fuel and channeled to a combustor wherein the mixture is ignited for generating hot combustion gases. At least some known combustors include a dome assembly, a cowling, and inner and outer liners to channel the combustion gases to a turbine, which extracts energy from the combustion gases for powering the compressor, as well as producing useful work to propel an aircraft in flight or to power a load, such as an electrical generator. The liners are coupled to the dome assembly with the cowling, and extend downstream from the cowling to define the combustion chamber. An outer support is coupled radially outward from the outer liner such that an outer cooling passage is defined radially outward from the outer liner, and an inner support is coupled radially inward from the inner liner such that an inner cooling passage is defined therebetween.

[0003] At least some known liners include a plurality of panels that are serially connected together between the upstream and aft ends of each liner such that the panels define the combustion chamber. At least some known panels are formed with primary airflow openings or secondary airflow openings. Known primary airflow openings are formed with a first diameter that is sized to enable sufficient air to enter the combustion chamber to facilitate complete oxidation of the fuel within the chamber. Known secondary airflow openings are typically formed with a smaller diameter than that of the primary airflow openings, and are positioned downstream from the primary airflow openings. The secondary airflow openings are sized to facilitate channeling airflow into the combustion chamber to facilitate diluting the combustion gases generated therein. However, the number of secondary openings that may be formed within a given panel is usually limited by structural considerations, and as such, the amount of dilution airflow that may be provided to the combustion chamber may be limited.

BRIEF DESCRIPTION OF THE INVENTION

[0004] In one aspect, a method for operating a gas turbine engine is provided. The method comprises channeling airflow into a cooling passageway defined between the combustor casing and an inner liner of the combustor, wherein the inner liner is fabricated from a plurality of panels coupled together, channeling airflow into a cooling passageway defined between the combustor casing and an outer liner of the combustor; wherein the outer liner is fabricated from a plurality of panels coupled together, and channeling dilution airflow into a combustion chamber defined between the inner and outer liners, through a plurality of openings formed within at least one panel within at least one of the inner liner panels and the outer liner panels, wherein the plurality of openings are non-circular.

[0005] In another aspect, a combustor for a gas turbine engine is provided. The combustor includes an inner liner, an outer liner, and a combustion chamber defined therebetween. The inner and outer liners each include a plurality of panels coupled together. At least one of the plurality of inner liner panels and the plurality of outer liner panels includes a plurality of openings extending therethrough for channeling dilution airflow into the combustion chamber. The plurality of openings are non-circular.

[0006] In a further aspect, a gas turbine engine is provided. The gas turbine engine includes a combustor including an inner liner, an outer liner, and a combustion chamber defined therebetween. The inner and outer liners each include a plurality of panels coupled together. At least one of the plurality of inner liner panels and the plurality of outer liner panels includes a plurality of openings extending therethrough for channeling dilution airflow into the combustion chamber. The plurality of openings are non-circular.

BRIEF DESCRIPTION OF THE DRAWINGS

[0007] **FIG. 1** is a schematic illustration of an exemplary gas turbine engine;

[0008] **FIG. 2** is a cross-sectional view of a combustor that may be used with the gas turbine engine;

[0009] **FIG. 3** is an enlarged perspective view of a portion of a liner used with the combustor shown in **FIG. 2** and taken along area **3**; and

[0010] **FIG. 4** is a plan view of a portion of the liner used with the combustor shown in **FIG. 2** and taken along area **4**.

DETAILED DESCRIPTION OF THE INVENTION

[0011] **FIG. 1** is a schematic illustration of an exemplary gas turbine engine **10** including a low pressure compressor **12**, a high pressure compressor **14**, and a combustor **16**. Engine **10** also includes a high pressure turbine **18**, and a low pressure turbine **20** arranged in a serial, axial flow relationship. Compressor **12** and turbine **20** are coupled by a first shaft **24**, and compressor **14** and turbine **18** are coupled by a second shaft **26**. In one embodiment, gas turbine engine **10** is an LMS100 engine commercially available from General Electric Company, Cincinnati, Ohio.

[0012] In operation, air flows through low pressure compressor **12** from an upstream side **28** of engine **10**. Compressed air is supplied from low pressure compressor **12** to high pressure compressor **14**. Highly compressed air is then delivered to combustor assembly **16** where it is mixed with fuel and ignited. Combustion gases are channeled from combustor assembly **16** to drive turbines **18** and **20**.

[0013] **FIG. 2** is a cross-sectional view of a combustor **30** that may be used with gas turbine engine **10**. **FIG. 3** is an enlarged perspective view of a portion of a liner **40** used with combustor **30** and taken along area **3**. **FIG. 4** is a plan view of a portion of liner **40** used with combustor **30** shown in **FIG. 2** and taken along area **4**. Combustor **30** includes a dome assembly **32**. A fuel injector **34** extends into dome assembly **32** and injects atomized fuel through dome assembly **32** into a combustion zone or chamber **36** of combustor **30** to form an air-fuel mixture that is ignited downstream of fuel injector **34**.

[0014] Combustor dome assembly **32** defines an upstream end of combustion zone **36** and includes a plurality of mixer assemblies **37** that are spaced circumferentially around combustor dome assembly **32** for delivering a mixture of fuel and air to combustion zone **36**. In the exemplary embodiment, combustor dome assembly **32** is a single annular combustor (SAC) that includes one annular combustor dome. However, it should be understood that in alternative embodiments combustor dome assembly **32** may include any number of combustor domes. For example, in one embodiment, combustor dome assembly **32** is a dual annular combustor (DAC), and, in another embodiment, combustor dome assembly **32** is a triple annular combustor.

[0015] Combustion zone **36** is defined by combustor liners **40** that shield components external to combustor **30** from heat generated within combustion zone **36**. Combustion zone **36** extends from dome assembly **32** downstream to a turbine nozzle assembly **41**. Liners **40** include an inner liner **42** and an outer liner **44**. Each liner **42** and **44** is annular and includes a plurality of separate panels **50**. In the exemplary embodiment, each panel **50** includes a series of steps **52**, each of which form a distinct portion of combustor liner **40**.

[0016] Outer liner **44** and inner liner **42** each include a respective aft-most panel **64** and **66**. Panels **64** and **66** are each located at the aft end **68** of combustion zone **36** and are adjacent turbine nozzle assembly **41**. Specifically, each panel **64** and **66** couples an aft end **70** and **72** of each respective liner **44** and **42** to turbine nozzle assembly **41**.

[0017] Each liner **42** and **44** also includes an annular support mount, or aft mount, **80** and **82**, respectively. Specifically, each support mount **80** and **82** couples an aft end **70** and **72** of each respective liner **44** and **42** to turbine nozzle assembly **41** and to a combustor casing **84** that extends substantially circumferentially around combustor **30**. More specifically, each support mount **80** and **82** extends radially outward from each respective liner **42** and **44** such that a radially outer cooling passageway **86** and a radially outer cooling passageway **88** are defined between combustor casing **84** and combustor liner **40**. Accordingly, cooling passageway **86** is defined between liner **42** and combustor casing **84** and cooling passageway **88** is defined between liner **44** and combustor casing **84**.

[0018] Each combustor panel **50** includes a combustor liner surface **90** and an exterior surface **92** that is radially outward from liner surface **90**. When panels **50** are coupled together, combustor liner surface **90** extends generally from dome assembly **32** to turbine nozzle assembly **41**. In the exemplary embodiment, each panel **50** is generally rectangular and includes a pair of circumferentially-spaced side edges **100** that are connected together by a leading edge side **102** and an opposed trailing edge side **104**.

[0019] Each liner **42** and **44** also includes at least one panel **110** that is downstream from fuel injector **34**, and includes a plurality of circumferentially-spaced primary airflow openings **111** that extend through panel **110** between combustor liner surface **90** and an exterior surface **92**. Openings **111** are substantially circular and have a diameter D_1. In the exemplary embodiment, openings **111** extend substantially circumferentially around combustion chamber **36**. Accordingly, openings **111** connect each cooling passageway **86** and **88** in flow communication with combustion chamber **36**. In the exemplary embodiment, panel **110** is at least two panels **50** upstream from turbine nozzle assembly **41**.

[0020] Each liner **42** and **44** also includes at least one panel **112** that is downstream from panel **110** and includes a plurality of circumferentially-spaced secondary or dilution airflow openings **116**. In the exemplary embodiment, openings **116** are spaced substantially circumferentially around combustion chamber **36**. Openings **116** extend through panel **112** between combustor liner surface **90** and an exterior surface **92** and are non-circular. More specifically, in the exemplary embodiment, openings **116** are substantially race-tracked shaped or generally elliptical and are defined by a pair of opposed, generally parallel sidewalls **120** that are connected by a pair of opposed arcuate sidewalls **122**.

[0021] In the exemplary embodiment, sidewalls **122** are formed with a pre-determined radius of curvature R_1 that is smaller than an associated radius /**1**;2D_1 of each primary cooling opening **111**. More specifically, in the exemplary embodiment, each sidewall **122** is substantially semi-circular. Accordingly, because sidewalls **120** are substantially parallel, within each opening, sidewalls **120** are separated by the diameter D_3 (twice the radius of curvature R_1) of each arcuate sidewall **122**.

[0022] In the exemplary embodiment, openings **116** are oriented such that sidewalls **120** are aligned generally axially. Accordingly, a distance of separation, known as web spacing, D_2 between circumferentially adjacent openings **116** is measured between adjacent opening sidewalls **120**. In the exemplary embodiment, distance D_2 is at least twice that of the diameter D_3 of each opening **116**. The distance of separation D_2 facilitates maintaining structural integrity of each panel **112**.

[0023] During operation, an annular diffuser **124** channels air discharged from compressor **14** into the combustor dome assemblies **32** and, more specifically, into mixer assemblies **37**, wherein the compressed air is mixed with fuel provided by fuel injector **34**. The fuel/air mixture is then ignited within combustion zone **36** to form combustion gases, which are discharged from the combustion zone **36** through turbine nozzle assembly **41**.

[0024] A portion of the compressed air discharged from compressor **14** is channeled into each cooling passageway **86** and **88** for cooling combustor assembly **30**. More specifically, the compressed air channeled through passageways **86** and **88** is also channeled into combustion zone **36** through primary cooling openings **111** defined within panels **110**. The compressed air channeled through openings **111** facilitates convectively cooling liners **42** and **44** in regions adjacent openings **111**. Moreover, air channeled through openings **111** also mixes with the fuel-air mixture within combustion chamber **36** to facilitate complete oxidation of all of the fuel supplied to chamber **36**.

[0025] As the fuel-air mixture is channeled downstream, the mixture is mixed with air channeled through dilution openings **116**. Openings **116** facilitate diluting the burned combustion products within chamber **36** to facilitate reducing the temperature of the combustion gases channeled downstream to the turbines. Moreover, the elongation of openings **116** facilitates increasing the penetration of the airflow jets discharged into chamber **36** from openings **116** in comparison to other known dilution openings, such as circular openings. The increased penetration of the dilution airflow enables openings **116** to facilitate shaping the radial temperature profile to a predetermined profile shape in an area where the mainstream velocities are relatively high.

[0026] The elongated shape of openings **116** facilitates enough penetration of the dilution air such that the air is not readily turned or forced over by the mainstream flow. Moreover, because openings **116** are oriented with their narrowest dimension in the circumferential direction, airflow discharged through openings **116** is more streamlined than airflow discharged through circular openings, which enables the airflow to penetrate the mainstream flow to a greater extent than is possible through a round opening.

[**0027**] The above-described gas turbine engine combustor includes a liner that includes at least one panel including a plurality of non-circular, dilution openings extending therethrough. The dilution openings are oriented such that their narrowest dimension extends circumferentially across the panel. The shape and orientation of the dilution openings enables airflow discharged from the openings to penetrate the mainstream flow to a greater extent than is possible through known round openings. As a result, the openings facilitate enhanced control of the radial temperature profile generated within the combustion chamber and increasing the useful life of the combustor in a cost-effective and reliable manner.

[**0028**] Exemplary embodiments of a combustor for a gas turbine engine are described above in detail. The systems and assembly components of the combustor are not limited to the specific embodiments described herein, but rather, components of each system may be utilized independently and separately from other components described herein. Each system and assembly component can also be used in combination with other combustor systems and assemblies or with other gas turbine engine components.

[**0029**] While the invention has been described in terms of various specific embodiments, those skilled in the art will recognize that the invention can be practiced with modification within the spirit and scope of the claims.

What is claimed is:

1. A method for operating a gas turbine engine, said method comprising:

channeling airflow into a cooling passageway defined between the combustor casing and an inner liner of the combustor, wherein the inner liner is fabricated from a plurality of panels coupled together;

channeling airflow into a cooling passageway defined between the combustor casing and an outer liner of the combustor; wherein the outer liner is fabricated from a plurality of panels coupled together; and

channeling dilution airflow into a combustion chamber defined between the inner and outer liners, through a plurality of openings formed within at least one panel within at least one of the inner liner panels and the outer liner panels, wherein the plurality of openings are non-circular.

2. A method in accordance with claim 1 wherein channeling dilution airflow into a combustion chamber further comprises channeling dilution airflow into the combustion chamber to facilitate controlling an exit temperature profile of the combustor.

3. A method in accordance with claim 1 wherein channeling dilution airflow into a combustion chamber further comprises channeling dilution airflow into the combustion chamber through the plurality of openings, wherein the openings are shaped to enable cooling air to penetrate into the combustion chamber to facilitate achieving a desired radial temperature profile within the combustion chamber.

4. A method in accordance with claim 1 wherein channeling dilution airflow into a combustion chamber further comprises channeling dilution airflow into the combustion chamber through the plurality of openings, wherein the openings are generally elliptically shaped.

5. A method in accordance with claim 1 wherein channeling dilution airflow into a combustion chamber further comprises channeling dilution airflow into the combustion chamber through the plurality of openings, wherein the openings are defined by a pair of substantially parallel walls that are connected together by a pair of opposed arcuate sidewalls formed with a predetermined radius of curvature.

6. A combustor for a gas turbine engine, said combustor comprising:

an inner liner comprising a plurality of panels coupled together;

an outer liner comprising a plurality of panels coupled together; and

a combustion chamber defined between said inner and outer liners, at least one of said plurality of inner liner panels and said plurality of outer liner panels comprises a plurality of openings extending therethrough for channeling dilution airflow into said combustion chamber, said plurality of openings are non-circular.

7. A combustor in accordance with claim 6 wherein said plurality of openings facilitate controlling an exit temperature profile of said combustor.

8. A combustor in accordance with claim 6 wherein said plurality of openings are each substantially elliptically-shaped.

9. A combustor in accordance with claim 6 wherein said plurality of openings are shaped to enable cooling air to penetrate into said combustion chamber to facilitate achieving a desired radial temperature profile within said combustion chamber.

10. A combustor in accordance with claim 6 wherein said plurality of openings are defined by a pair of opposed substantially parallel sidewalls connected together by a pair of opposed arcuate walls formed with a pre-determined radius.

11. A combustor in accordance with claim 10 wherein adjacent of said plurality of openings are separated by a distance that is approximately equal to twice the diameter of said arcuate walls.

12. A combustor in accordance with claim 6 wherein said at least one panel comprises a pair of opposed circumferential edges coupled together by a leading edge and a side edge, said plurality of openings comprises at least three openings spaced approximately equi-distantly between said pair of opposed circumferential edges.

13. A gas turbine engine comprising a combustor comprising an inner liner, an outer liner, and a combustion chamber defined between said inner and outer liners, each of said inner and outer liners comprises a plurality of panels coupled together, at least one of said panels within at least one of said inner liner and said outer liner comprises a plurality of openings extending therethrough for channeling dilution air into said combustion chamber, said plurality of openings are non-circular.

14. A gas turbine engine in accordance with claim 13 wherein said combustor plurality of openings extending through said at least one panel facilitate controlling an exit temperature profile of said combustor.

15. A gas turbine engine in accordance with claim 14 wherein said combustor plurality of openings extending through said at least one panel are each generally elliptically-shaped.

16. A gas turbine engine in accordance with claim 14 wherein said combustor plurality of openings extending through said at least one panel are shaped to enable cooling air to penetrate into said combustion chamber to facilitate achieving a desired radial temperature profile within said combustion chamber.

17. A gas turbine engine in accordance with claim 14 wherein said combustor plurality of openings extending through said at least one panel are defined by a pair of opposed substantially parallel sidewalls that are connected together by a pair of opposed arcuate walls formed with a pre-determined radius.

18. A gas turbine engine in accordance with claim 17 wherein adjacent of said plurality of openings extending through said at least one panel are separated within said panel by a distance that is approximately equal to twice the diameter of said arcuate walls.

19. A gas turbine engine in accordance with claim 14 wherein each of said plurality of panels comprises a pair of opposed circumferential edges coupled together by a leading edge and a side edge, said plurality of openings extending through said at least one panel comprises at least three openings spaced approximately equi-distantly between said pair of opposed circumferential edges.

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