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THERMAL PROTECTION FOR SEALS

Abstract

An aircraft engine has a flammable fluid containment assembly including a component containing a flammable fluid. A transfer tube is fluidly connected to the component. A sealing ring is provided at an interface between the aircraft engine component and the transfer tube. A first thermal barrier is provided between the sealing ring and a first one of the aircraft engine component and the flammable fluid inside the transfer tube. The first thermal barrier includes a first annular cavity filled with a first thermal insulation medium and axially spanning the sealing ring.

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Background/Summary

TECHNICAL FIELD

[0001] The application relates generally to flammable fluid containment systems for aircraft engines and, more particularly, to a thermal protection for the seals of such systems.

BACKGROUND OF THE ART

[0002] In order to meet airworthiness certification requirements, fire-risk zones of aircraft, such as the engines, are required by government regulations to be able to function for a specific period of time when exposed to fire, for example in the event of an engine fire. While existing fire protection structures perform satisfactorily, improvements remain desired.

SUMMARY

[0003] In one aspect, there is provided a flammable fluid containment assembly for an aircraft engine, comprising: an aircraft engine component having an internal cavity containing a flammable fluid; a transfer tube having a central axis, the transfer tube fluidly connected to the internal cavity of the aircraft engine component; a sealing ring at a radial interface between the aircraft engine component and the transfer tube; and a first thermal barrier between the sealing ring and a first one of the aircraft engine component and the flammable fluid inside the transfer tube, the first thermal barrier including a first annular cavity filled with a first thermal insulation medium and axially spanning the sealing ring.

[0004] In another aspect, there is provided a flammable fluid containment system for an aircraft engine, comprising: an aircraft engine component including a case having an inner diameter surface circumscribing a bore leading to an internal cavity containing a flammable fluid; a first sleeve mounted to the inner diameter surface circumscribing the bore of the case, the first sleeve having an outer diameter surface and an inner diameter surface; a first annular gap between the outer diameter surface of the first sleeve and the inner diameter surface of the case, the first annular gap containing a first thermal insulation medium; a transfer tube slidably engaged with the bore, the transfer tube having an outer diameter surface facing the inner diameter surface of the first sleeve and an inner diameter surface circumscribing a central passage fluidly connected to the internal cavity of the case; and a sealing ring between the inner diameter surface of the first sleeve and the outer diameter surface of the transfer tube.

[0005] In a further aspect, there is provided an aircraft engine comprising: a gearbox; a transfer tube fluidly connected to the gearbox; a sealing ring at an interface between the gearbox and the transfer tube; and a first thermal barrier configured to reduce a flow of thermal energy from the gearbox or the transfer tube to the sealing ring, the first thermal barrier including a first annular cavity filled with a first thermal insulation medium and extending axially from a first side of the sealing ring to a second side opposite to the first side.

Description

DESCRIPTION OF THE DRAWINGS

[0006] Reference is now made to the accompanying figures in which:

[0007] FIG. **1** is a schematic cross-section view of an aircraft engine;

[0008] FIG. **2** is schematic enlarged cross-section view of a flammable fluid containment system of the aircraft engine; and

[0009] FIG. **3** is a schematic enlarged cross-section view illustrating details of a thermal break for thermally shielding a sealing ring from the flammable fluid flowing through a transfer tube of the flammable fluid containment system.

DETAILED DESCRIPTION

[0010] FIG. **1** illustrates a gas turbine engine **10** of a type preferably provided for use in subsonic flight, generally comprising in serial flow communication a fan **12** through which ambient air is propelled, a compressor section **14** for pressurizing the air, a combustor **16** in which the compressed air is mixed with fuel and ignited for generating an annular stream of hot combustion gases, and a turbine section **18** for extracting energy from the combustion gases. A reduction gearbox **20** drivingly connect a power turbine of the turbine section **18** to the fan. Even though

FIG. **1** is specific to a turbofan gas turbine engine, it is understood that the various aspects of the present disclosure may be equally applicable to other types of aircraft engines including turboshaft engines, turboprop engines, turbojet engines, auxiliary power units and electric-hybrid aircraft engines to name a few.

[0011] The exemplary aircraft engine **10** has a number of components that may be designated as “fire-critical” components, i.e., components that need to be protected from fire, should an emergency fire ever occur within the engine. Such components include fluid system components carrying flammable fluids, such as oil and fuel. These may for example include, but are certainly not limited to, gearboxes, oil/fuel pumps, oil/fuel metering units, fuel/oil heat exchangers, fuel manifolds and the like. As such, certain “fire-critical” components that are disposed within or exposed to fire-risk zones of the engine may be protected by a protective fire shield (e.g., a fire protective blanket) such that, in the eventuality of a fire, these “fire-critical” components will be able to withstand fire for a specific period of time. More particularly, in order for the engine to meet the necessary airworthiness certification standards, these components are designed to withstand fire for a predetermined minimum period, for example a period of time sufficient long for the pilot of the aircraft to take the necessary precautionary actions, such as land the aircraft, shut down the engine, disable the fuel flow to the engine, etc. For example, in at least one certification test, the “fire-critical” components must be able to withstand a fire with a temperature of 2000 degrees F. for at least fifteen minutes.

[0012] Fluid system components typically comprise packings/seals (e.g., O-rings) at the interface between adjacent components. Such packings/seals are typically made of compressible elastomeric materials, which are less resistant to heat than the fire resistant or fireproof materials (e.g., metallic materials) forming the “fire-critical” fluid system components. Accordingly, even if properly fire shielded, the integrity of such packings/seals may be compromised if subjected to excessive temperatures. In case of a fire event, heat can be transferred by conduction through the body of the fluid system component to the flammable fluid inside the body and to the packings/seals. Such heat transfer from the body of the component under fire attack and/or from the flammable fluid to the packings/seals can cause the packings/seals to melt or degrade to a point where the packings/seals start to leak. Such flammable fluid leakage may represent a serious fire hazard within an aircraft fire-risk zone.

[0013] FIG. **2** illustrates an embodiment of a flammable fluid containment system **30** including thermal protection for the packings/seals of the system. The flammable fluid containment system **30** generally comprises an aircraft engine component **32** containing a flammable fluid. As mentioned hereinabove, the aircraft engine component **32** can take various forms. For instance, the aircraft engine component **32** can consist of a gearbox, such as the gearbox **20** in FIG. **1**. The component **32** comprises a metallic case **34** defining a port or bore **36** opening to an internal cavity **38** containing the flammable fluid (i.e., oil in the case of a gearbox). The bore **36** is configured for slidably receiving one end of a transfer tube **40** having a rigid tubular body extending axially along a central tube axis A. The opposed end of the transfer tube **40** is axially slidably engaged with another component (not shown) of the fluid system **30**, such as an oil tank, a scavenging pump or the like. In use, the transfer tube **40** is configured to accommodate relative movements between the fluid system components by virtue of its axially sliding engagement therewith. The tubular body of the transfer tube **40** may be made of any suitable fire-resistant or fireproof materials. For instance, the transfer tube **40** may be made out of stainless steel or aluminum.

[0014] Still referring to FIG. **2**, it can be appreciated that a sealing ring **42** (also herein referred to as a packing) is provided at an interface between the transfer tube **40** and the case **34**. In some embodiments, the sealing ring **42** is provided in the form of an elastomeric O-ring removably received in an annular groove **43** defined in an outer diameter surface of the transfer tube **40**. A first thermal barrier **44** is provided between the case **34** and the sealing ring **42**. The first thermal barrier **44** thermally shields the sealing ring **42** from the case **34**. The first thermal barrier **44** is configured to protect the sealing ring **42** from heat conduction through the body of the case **34**. In this way, in the case of a fire event, wherein a flame impinges upon the case **34** of the aircraft engine component **32**, heat transfer from the case **34** to the sealing ring **42** can be delayed and, thus, the sealing ring **42** can remain functional for a longer period of time.

[0015] In some embodiments, the first thermal barrier **44** comprises a first annular cavity **44***a* surrounding the sealing ring **42**, the first annular cavity **44***a* filled with a first thermal insulation medium **44***b*. As shown in FIG. **2**, the first annular cavity **44***a* axially spans the sealing ring **42**. That is the first annular cavity **44***a* extends axially from a first axial location **44***c* disposed on a first side of the sealing ring **42** to a second axial location **44***d* disposed on a second side of the sealing ring **42**. In some embodiments, the first thermal insulation medium **44***b* is air. The first annular cavity **44***a* thus forms an air gap or dead air cavity between the case **34** and the sealing ring **42**. Still according to some embodiments, a first sleeve **44***e* is used to create the first annular cavity **44***a*. As shown in FIG. **2**, the first annular cavity **44***a* is defined radially between an outer diameter surface of the first sleeve **44***e* and the inner diameter surface circumscribing the transfer tube receiving bore **36** of the case **34**. The sealing ring **42** is compressed in sealing engagement against the inner diameter surface of the first sleeve **44***e*, thereby sealing the interface between the case **34** and the transfer tube **40**. The first sleeve **44***e* can be fixed in position by any appropriate means. For instance, the first sleeve **44***e* can be brazed and/or welded to the case **34** inside the bore **36**. Alternatively, the sleeve **44***e* can be press fit into the bore **36**. Still referring to FIG. **2**, it can be appreciated that the sleeve **44***e* has a pair of circumferential rails **44***f*, **44***g* projecting radially outwardly from opposed axial ends of the sleeve **44***e* for engagement with the inner diameter surface of the bore **36**. In some embodiments, the leading rail **44***f* is brazed to the case **34** and the trailing rail **44***g* is brazed or welded to the case **34**. The first sleeve **44***e* is made of a fire-resistant or fireproof material suitable for brazing or welding to the case **34** (e.g., the same metallic material as the case **34**). The height of the rails **44***g*, **44***g* may be adjusted to create the desired air gap thickness between the case **34** and the sealing ring **42**.

[0016] In some embodiments, 3D printing or additive manufacturing can be used to create the first annular cavity **44***a*. That is the first annular cavity **44***a* could be integrally formed with the case **34** of the aircraft engine component **32** (see FIG. **3**). This would eliminate the need to install a sleeve inside the bore **36**.

[0017] It is also understood that the first thermal insulation medium **44***b* is not limited to air. Indeed, the first thermal insulation medium **44***b* could include a wide variety of insulation materials such as fibreglass, cellulose, ceramic fibres, and the like. The thermal insulation medium could also include various combinations of such insulation materials to provide reduce thermal conductivity.

[0018] In some embodiments, a second thermal barrier **46** may be provided to thermally shield the sealing ring **42** from the flammable fluid flowing through the transfer tube **40**. The second thermal barrier **46** may have a composition/configuration similar to or different from the first thermal barrier **44**. In some embodiments, the second thermal barrier **46** includes a second annular cavity **46***a* on the flammable fluid side of the sealing ring **42**. As shown in FIG. **2**, the second annular cavity **46***a* may be created radially inwardly of the sealing ring **42** and, more specifically, inside the transfer tube **40**. Like the first annular cavity **44***a*, the second annular cavity **46***a* axially spans the sealing ring **42**. The second annular cavity **46***a* is filled with a second thermal insulation medium **46***b*. The second thermal insulation medium **46***b* can be the same as the first thermal insulation medium **44***b* or it could be a different insulation material. In the illustrated embodiment, the second thermal insulation medium **46***b* is air. The second annular cavity **46***a* thus forms an air gap (i.e., a dead air space) radially between the sealing ring **42** and the flammable fluid flowing through the transfer tube **40**. In some embodiments, a second sleeve **46***e* is axially inserted inside the first end of the transfer tube **40** to create the second annular cavity **46***a*. As shown in FIG. **2**, the second annular cavity **46***a* is defined radially between an inner diameter surface of the transfer tube **40** and an outer diameter surface of the second sleeve **46***e*. The second sleeve **46***e* may be fabricated from sheet metal material and welded and/or brazed at opposed axial ends thereof to the transfer tube **40**. For instance, the leading end **46***f* of the second sleeve **46***e* may be brazed to the inner diameter surface of the transfer tube **40** and the trailing end **46***g* may be brazed or welded to the axially end facing surface of the transfer tube **40**. As illustrated in FIG. **2**, the second sleeve **46***e* may be profiled/shaped in order to minimize flow obstruction through the transfer tube **40** and provide a smooth flow boundary surface inside the transfer tube **40**.

[0019] Referring to FIG. **3**, it can be appreciated that the second annular cavity could be provided in the form of a circumferentially extending recess **46***a′* defined in the inner diameter surface of the transfer tube **40** at an axial location aligned with the sealing ring receiving groove **43**. The second annular cavity **46***a′* could be filled with a solid thermal insulation material **46***b′*. As shown in FIG. **3**, by so embedding the insulation material into the wall thickness of the transfer tube **40**, a uniform inner diameter surface can be preserved along all the length of the transfer tube **40**.

[0020] From the foregoing, it can be appreciated that in some embodiments, a thermal break may be provided on both the inner and outer sides of the sealing ring **42**. According to still further embodiments, one of the first thermal barrier **44** or the second thermal barrier **46** could be omitted. That is the system **30** could include a single thermal barrier on either the case side or the flammable fluid side of the sealing ring **42**. In embodiments including a single thermal break, the expression “first thermal barrier” is herein intended to designate a thermal break on the case side or on the fluid side of the sealing ring **42** (i.e., the first thermal barrier is not herein limited to the thermal barrier **44** on the case side).

[0021] It is noted that the thermal barriers can be retrofitted to existing parts or be integrated as part of new components to protect the seals from indirect heat via conduction.

[0022] From the foregoing description, it can be appreciated that the described thermal barriers protect the seals from heat conduction, thereby allowing to delay temperature rise. By so breaking the thermal conduction, leakage of the flammable fluid due to seal degradation may be at least delayed if not avoided. By disposing thermal barriers radially on both sides of the seals, it may be possible to thermally shield the seals from both the case of the component under fire attack and the flammable fluid contained inside the component.

[0023] It is noted that various connections are set forth between elements in the preceding description and in the drawings. It is noted that these connections are general and, unless specified otherwise, may be direct or indirect and that this specification is not intended to be limiting in this respect. A coupling between two or more entities may refer to a direct connection or an indirect connection. An indirect connection may incorporate one or more intervening entities. The term “connected” or “coupled to” may therefore include both direct coupling (in which two elements that are coupled to each other contact each other) and indirect coupling (in which at least one additional element is located between the two elements). Herein, an element characterized as fireproof means that this element is able to withstand a flame of 2000° F. for a minimum of 15 minutes or as otherwise specified by the airworthiness directives of regulatory authorities. An element characterized as fire resistant herein means that this element is able to withstand a flame of 2000° F. for a minimum of 5 minutes or as otherwise specified by the airworthiness directives of regulatory authorities.

[0024] Furthermore, no element, component, or method step in the present disclosure is intended to be dedicated to the public regardless of whether the element, component, or method step is explicitly recited in the claims. As used herein, the terms “comprises”, “comprising”, or any other variation thereof, are intended to cover a non-exclusive inclusion, such that a process, method, article, or apparatus that comprises a list of elements does not include only those elements but may include other elements not expressly listed or inherent to such process, method, article, or apparatus.

[0025] While various aspects of the present disclosure have been disclosed, it will be apparent to those of ordinary skill in the art that many more embodiments and implementations are possible

within the scope of the present disclosure. For example, the present disclosure as described herein includes several aspects and embodiments that include particular features. Although these particular features may be described individually, it is within the scope of the present disclosure that some or all of these features may be combined with any one of the aspects and remain within the scope of the present disclosure. References to “various embodiments,” “one embodiment,” “an embodiment,” “an example embodiment,” etc., indicate that the embodiment described may include a particular feature, structure, or characteristic, but every embodiment may not necessarily include the particular feature, structure, or characteristic. Moreover, such phrases are not necessarily referring to the same embodiment. The use of the indefinite article “a” as used herein with reference to a particular element is intended to encompass “one or more” such elements, and similarly the use of the definite article “the” in reference to a particular element is not intended to exclude the possibility that multiple of such elements may be present.
[0026] The skilled person will understand that embodiments described in this document provide non-limiting examples of possible implementations of the present technology. Upon review of the present disclosure, a person of ordinary skill in the art will recognize that changes may be made to the embodiments described herein without departing from the scope of the present technology.

Claims

1. A flammable fluid containment assembly for an aircraft engine, comprising: an aircraft engine component having an internal cavity containing a flammable fluid; a transfer tube having a central axis, the transfer tube fluidly connected to the internal cavity of the aircraft engine component; a sealing ring at a radial interface between the aircraft engine component and the transfer tube; a first thermal barrier between the sealing ring and the aircraft engine component, the first thermal barrier including a first sleeve mounted to an inner diameter surface of the aircraft engine component around the sealing ring. the first sleeve having an outer diameter surface and an inner diameter surface, the outer diameter surface of the first sleeve and the inner diameter surface of the aircraft engine component defining a first annular cavity therebetween, the first annular cavity filled with a first thermal insulation medium and axially spanning the sealing ring; and a second thermal barrier between the sealing ring and the flammable fluid inside the transfer tube, the second thermal barrier including a second sleeve mounted inside the transfer tube, the second sleeve having an inner diameter surface and an outer diameter surface, the outer diameter surface of the second sleeve and an inner diameter surface of the transfer tube defining a second annular cavity therebetween, the second annular cavity filled with a second thermal insulation medium and axially spanning the sealing ring, the inner diameter surface of the second sleeve forming a flow boundary surface for the flammable fluid flowing through the transfer tube.
2. The flammable fluid containment assembly according to claim 1, wherein the sealing ring is disposed radially between the first and second thermal barriers.
3. The flammable fluid containment assembly according to claim 2, wherein the first annular cavity and the second annular cavity are at least partly filled with air to form first and second annular air gaps on opposed inner and outer radial sides of the sealing ring.
4. The flammable fluid containment assembly according to claim 1, wherein the first thermal insulation medium is air, and wherein the first annular cavity forms an air gap between the aircraft engine component and the sealing ring.
5. The flammable fluid containment assembly according to claim 1, wherein the aircraft engine component includes a case having a transfer tube receiving bore in fluid communication with the internal cavity, the transfer tube receiving bore circumscribed by the inner diameter surface of the aircraft engine component, the transfer tube having a first end axially slidably received in the transfer tube receiving bore, and wherein the sealing ring is in sealing engagement with an outer diameter surface of the transfer tube and the inner diameter surface of the first sleeve.

6. The flammable fluid containment assembly according to claim 5, wherein the first annular cavity is a dead air cavity radially between the inner diameter surface of the transfer tube receiving bore and an outer diameter surface of the first sleeve.
7. The flammable fluid containment assembly according to claim 1, wherein an inner diameter of the second sleeve varies along the central axis of the transfer tube.
8. The flammable fluid containment assembly according to claim 1, wherein the second sleeve has a first end brazed to the inner diameter surface of the transfer tube and a second end brazed or welded to an axially facing end surface of the transfer tube.
9. (canceled)
10. A flammable fluid containment system for an aircraft engine, comprising: an aircraft engine component including a case having an inner diameter surface circumscribing a bore leading to an internal cavity containing a flammable fluid; a first sleeve mounted to the inner diameter surface circumscribing the bore of the case, the first sleeve having an outer diameter surface and an inner diameter surface; a first annular gap between the outer diameter surface of the first sleeve and the inner diameter surface of the case, the first annular gap containing a first thermal insulation medium; a transfer tube slidably engaged with the bore, the transfer tube having an outer diameter surface facing the inner diameter surface of the first sleeve and an inner diameter surface circumscribing a central passage fluidly connected to the internal cavity of the case; and a sealing ring between the inner diameter surface of the first sleeve and the outer diameter surface of the transfer tube.
11. The flammable fluid containment system according to claim 10, further comprising: a second sleeve mounted to the inner diameter surface of the transfer tube; and a second annular gap between the inner diameter surface of the transfer tube and an outer diameter surface of the second sleeve, the second annular gap containing a second thermal insulation medium.
12. The flammable fluid containment system according to claim 11, wherein the first and second thermal insulation media is air.
13. The flammable fluid containment system according to claim 11, wherein the second sleeve includes a sheet metal material brazed to the transfer tube, and wherein the second thermal insulation medium is air.
14. The flammable fluid containment system according to claim 10, wherein the first sleeve has a pair of circumferential rails projecting radially outwardly from opposed axial ends thereof, and wherein the circumferential rails are welded or brazed to the inner diameter surface circumscribing the bore of the case, and wherein the first annular gap is defined axially between the circumferential rails.
15. The flammable fluid containment system according to claim 14, wherein the first thermal insulation medium is air.
16. The flammable fluid containment system according to claim 11, wherein the first and second thermal insulation media axially span the sealing ring.
17. The flammable fluid containment system according to claim 10, wherein the aircraft engine component is a gearbox, and wherein the flammable fluid is oil.
18. The flammable fluid containment system according to claim 10, wherein a recess is defined in the inner diameter surface of the transfer tube at an axial location aligned with the sealing ring, and wherein the recess is filled with a second thermal insulation medium.
19. An aircraft engine comprising: a gearbox; a transfer tube fluidly connected to the gearbox; a sealing ring at an interface between the gearbox and the transfer tube; and a first thermal barrier configured to reduce a flow of thermal energy from the gearbox or the transfer tube to the sealing ring, the first thermal barrier including a first annular cavity filled with a first thermal insulation medium and extending axially from a first side of the sealing ring to a second side opposite to the first side.
20. The aircraft engine according to claim 19, further comprising: a second thermal barrier

including a second annular cavity filled with a second thermal insulation medium and extending axially from the first side to the second side opposite of the sealing ring, and wherein the sealing ring is disposed radially between the first and second thermal barriers.

21. The flammable fluid containment assembly according to claim 5, wherein the first sleeve has a pair of circumferential rails projecting radially outwardly from opposed axial ends thereof, and wherein the circumferential rails are welded or brazed to the inner diameter surface circumscribing the transfer tube receiving bore of the case, and wherein the first annular gap is defined axially between the circumferential rails.