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(54) **ASSEMBLY FOR CONTROLLING THERMAL STRESSES IN CERAMIC MATRIX COMPOSITE ARTICLES**

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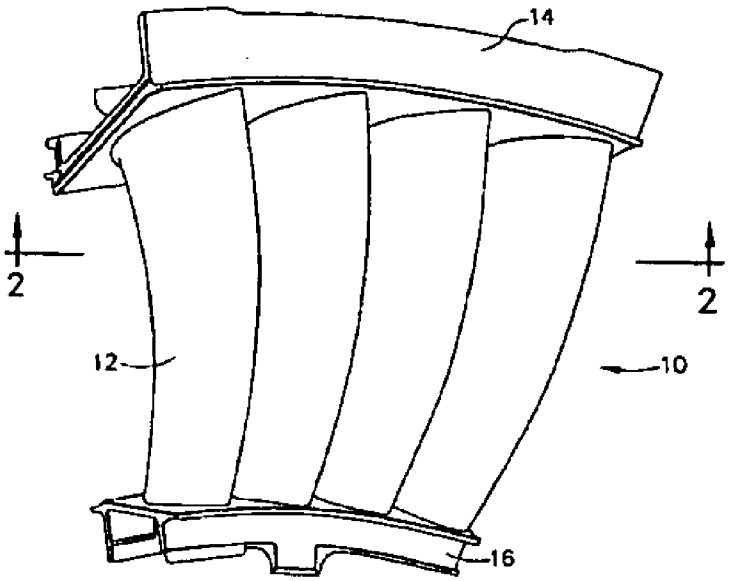
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(57) **ABSTRACT**

An assembly and method for controlling thermal stresses within ceramic-based articles when subjected to high temperatures while supported by a metallic article. The assembly includes a first body formed of a metallic material and having oppositely-disposed first and second surfaces, and a second body formed of a ceramic-based material and supported by the first body from the first surface thereof. The first and second bodies are located in a hot gas path such that the second body and the first surface of the first body are directly impinged by flowing hot gases. The assembly further includes a substantially uniform pattern of fins protruding from the second surface of the first body, and/or an interface structure between the first and second bodies that positively retains the second body to the first body and thermally insulates the first body from the first body.



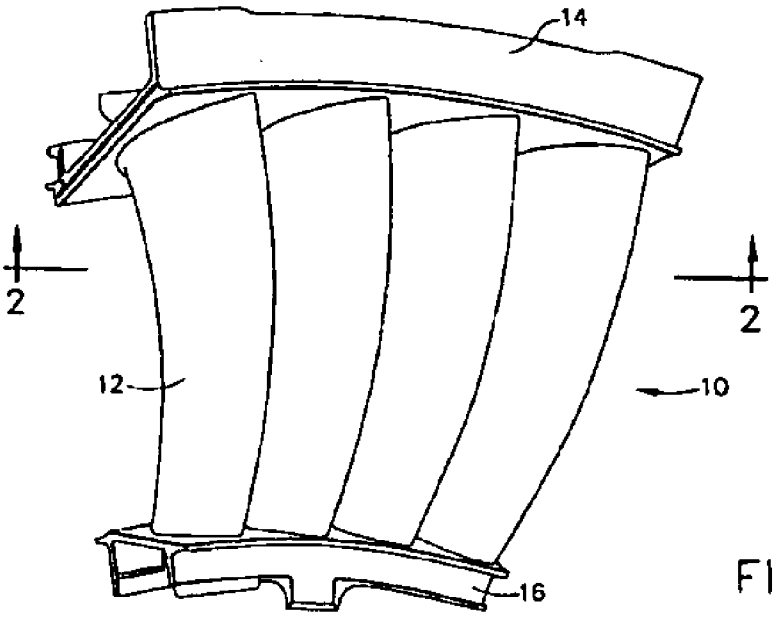


FIG. 1

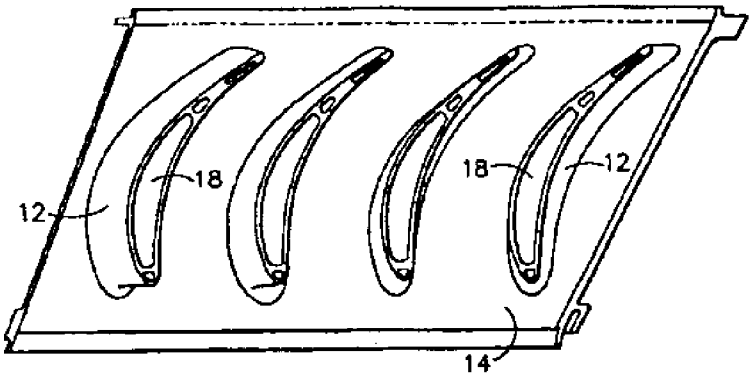


FIG. 2

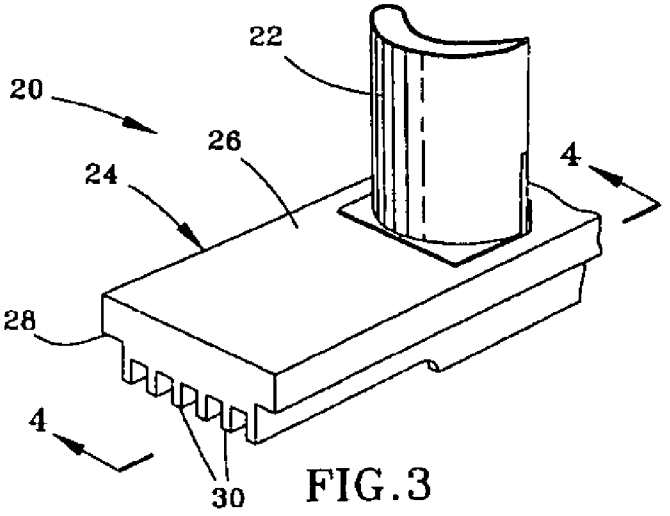


FIG.3

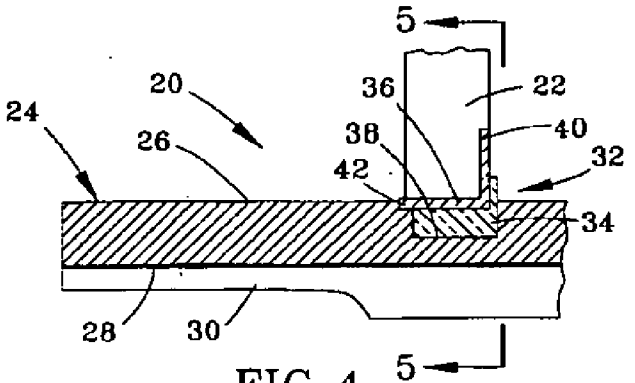


FIG.4

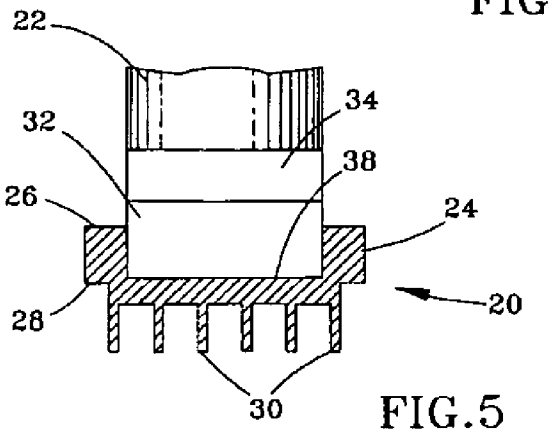


FIG.5

ASSEMBLY FOR CONTROLLING THERMAL STRESSES IN CERAMIC MATRIX COMPOSITE ARTICLES

BACKGROUND OF THE INVENTION

[0001] This invention relates to ceramic matrix composite (CMC) articles, such as CMC components of gas turbine engines. More particularly, this invention is directed to an assembly and method for controlling thermally-induced stresses that exist within CMC articles when subjected to high temperatures while supported by metallic structures.

[0002] Higher operating temperatures for gas turbine engines are continuously sought in order to increase their efficiency. However, as operating temperatures increase, the high temperature durability of the components of the engine must correspondingly increase. Significant advances in high temperature capabilities have been achieved through the formulation of iron, nickel and cobalt-base superalloys. Nonetheless, superalloy components must often be air-cooled and/or protected by some form of thermal and/or environmental coating system in order to exhibit suitable service lives in certain sections of a gas turbine engine, such as the turbine, combustor and augmentor.

[0003] As an example, FIGS. **1** and **2** represent a nozzle segment **10** that is one of a number of nozzle segments that when connected together form an annular-shaped high pressure turbine (HPT) nozzle assembly of a gas turbine engine. The segment **10** is made up of multiple vanes **12**, each defining an airfoil and extending between outer and inner platforms (bands) **14** and **16**. The vanes **12** and platforms **14** and **16** can be formed separately and then assembled, such as by brazing the ends of each vane **12** within openings defined in the platforms **14** and **16**. Alternatively, the entire segment **10** can be formed as an integral casting. As a result of being located in the high pressure turbine section of the engine, the vanes **12** and the surfaces of the platforms **14** and **16** facing the vanes **12** are subjected to the hot combustion gases from the engine's combustor. Compressor bleed air is often supplied to the vanes **12** and platforms **14** and **16** to provide forced air cooling, such as with impingement and film cooling techniques. A thermal barrier coating (TBC) system is typically applied to the surfaces of the vanes **12** and platforms **14** and **16** exposed to the hot combustion gases to provide environmental protection and reduce heat transfer to the segment **10**.

[0004] As one would expect, the nozzle segment **10** expands and contracts when heated and cooled, respectively, during transient engine operating conditions. Though provided with thermal protection to minimize service temperatures, the placement of the vanes **12** directly in the hot gas path results in the vanes **12** sustaining temperatures that are significantly higher than that experienced by the platforms **14** and **16**. As a result, in the situation where the vanes **12** and platforms **14** and **16** are formed of similar materials, such as nickel-base superalloys widely used, the vanes **12** will expand and contract more than the platforms **14** and **16**, thereby inducing significant thermally-induced strains and stresses in the segment **10**. Furthermore, significant temperature gradients will exist along the length of each vane **12** as a result of direct heating by the combustion gases and the platforms **14** and **16** behaving as heat sinks, thereby further increasing the thermally-induced stresses within the vanes **12**.

[0005] In view of the higher operating temperatures sought for gas turbine engines to increase their efficiency, ceramic matrix composite (CMC) materials, such as silicon carbide (SiC)-containing CMC's, have been proposed as materials for certain components of gas turbine engines, including turbine vanes, blades, shrouds, combustor liners, and other high-temperature components of gas turbine engines. CMC materials have much higher temperature capabilities than the superalloys currently in use, and therefore better capable of withstanding the temperatures sustained in the hot gas path of a gas turbine engine. Depending on the exact materials used, CMC vanes **12** also typically have relatively lower coefficients of thermal expansion (CTE), resulting in lower strains and stresses thermally induced by the higher temperatures sustained by the vanes **12** as compared to the platforms **14** and **16**. However, CMC materials are also much less ductile and have significantly lower thermal conductivities than those of superalloys, rendering CMC vanes particularly prone to damage from thermally-induced stresses and strains. As a result, the general approach to implementing CMC vanes (e.g., **12**) supported with superalloy platforms (e.g., **14** and **16**) is to support the ends of the vanes with the platforms in a manner that allows differential thermal movement of the vanes relative to the platforms. Notable approaches are described in commonly-assigned U.S. Pat. No. 5,630,700 to Olsen et al. and U.S. Pat. No. 6,464,456 to Darolia et al.

[0006] Notwithstanding the above advancements proposed by Olsen et al. and Darolia et al., further improvements would be desirable in order to fully implement nozzle assemblies having CMC vanes supported by metallic platforms. In particular, it would be desirable to improve the interface and attachment of the CMC vanes to the metallic platforms in order to not only reduce temperatures and thermally-induced stresses and strains, but also minimize the significant temperature gradients that can exist within a CMC vane **12**, which have been observed at times to exceed about 600° F. per inch (about 124° C./cm).

BRIEF SUMMARY OF THE INVENTION

[0007] The present invention generally provides an assembly and method for controlling thermal stresses within a ceramic-based article when subjected to high temperatures while structurally supported by a metallic article.

[0008] The assembly includes a first body formed of a metallic material and having oppositely-disposed first and second surfaces, and a second body formed of a ceramic-based material and supported by the first body from the first surface thereof. The first and second bodies are located in a hot gas path such that the second body and the first surface of the first body are directly impinged by flowing hot gases. According to a first aspect of the invention, the assembly further includes a substantially uniform pattern of fins protruding from the second surface of the first body. The fins are of sufficient size to increase the rigidity of the first body and promote heat transfer from the first body. According to a preferred aspect of the invention, the fins further serve to achieve more uniform temperatures within the first and second bodies and increase the stiffness of the first body to the extent that the thermal mass of the first body can be minimized.

[0009] According to a second aspect of the invention, the assembly further has an interface structure between the first

and second bodies. The interface structure comprises a resilient sealing member disposed between the first and second bodies and received in a recess in the first surface of the first body, and a ceramic saddle formed separately from the first and second bodies, received in the recess with the resilient sealing member, and disposed between the resilient sealing member and the second body. According to this aspect of the invention, the ceramic saddle and resilient sealing member cooperate to assist in positively retaining the second body to the first body and thermally insulating the first body from the first body in a manner that reduces thermal stresses within the second body, such as those caused by thermal gradients within the second body.

[0010] The method of this invention generally entails forming a first body of a metallic material to have a first surface and a substantially uniform pattern of fins protruding from an oppositely-disposed second surface. The fins are of sufficient size to increase the rigidity of the first body and promote heat transfer from the first body. A second body of a ceramic-based material is then supported from the first surface of the first body. The resulting assembly is then placed in a hot gas path such that the second body and the first surface of the first body are directly impinged by flowing hot gases.

[0011] It can be appreciated from the above that the invention is particularly applicable to nozzle assemblies of gas turbine engines, specifically where a ceramic-based vane is supported by and between a pair of platforms.

[0012] Other objects and advantages of this invention will be better appreciated from the following detailed description.

BRIEF DESCRIPTION OF THE DRAWINGS

[0013] FIGS. **1** and **2** are perspective and cross-sectional views, respectively, of a nozzle segment for a gas turbine engine and of a type that can benefit from the present invention.

[0014] FIG. **3** is a perspective view showing a portion of a nozzle assembly configured in accordance with a preferred embodiment of the present invention.

[0015] FIG. **4** is a cross-sectional view of the nozzle assembly of FIG. **3** taken along section line **4-4** in FIG. **3**.

[0016] FIG. **5** is a sectional view of the nozzle assembly of FIG. **3** taken along section line **5-5** in FIG. **4**.

DETAILED DESCRIPTION OF THE INVENTION

[0017] FIGS. **3** through **5** schematically represent a portion of a nozzle segment **20** that, when assembled with similarly configured segments, forms an annular-shaped HPT nozzle assembly of a gas turbine engine. The nozzle segment **20** is depicted as including a single vane **22** supported by a single platform **24**, though it will be understood that multiple vanes can be supported by the platform **24** in combination with a second platform, resulting in a construction generally similar to that shown in FIGS. **1** and **2**. The vane **22** is depicted as being hollow, though a variety of configurations are possible, including vanes configured to have struts, spars, inserts for mechanical support, baffles for enhanced internal cooling, etc. While the invention will be described in reference to a HPT nozzle assembly, it will be appreciated that the benefits of the invention can be applied to a variety of other components, including but not limited to low pressure turbine (LPT) nozzle assemblies and other hot section components of gas turbine engines.

[0018] The platform **24** can be formed of such conventional materials as a single-crystal nickel, cobalt, or iron-base superalloy of a type suitable for use in gas turbine engines. Conventional practice has been to also form the vane **22** of the same or similar superalloy, such that the vane **22** and platform **24** would have similar CTE's and thermal conductivities to minimize thermally-induced strains and stresses during engine operation. However, according to the invention the vane **22** is formed of a ceramic-based material, more preferably a CMC material such as a SiC/SiC (reinforcement/matrix) CMC. However, it should be understood that this invention is applicable to the use of a variety of ceramic-based and metallic materials, as well as intermetallic materials such as nickel aluminides (NiAl), and particularly combinations of these materials that result in combinations with significantly different CTE's and/or thermal conductivities. For example, CMC materials of particular interest for the vane **22** may have CTE's and thermal conductivities in ranges of about 1.9×10^{-6} to about 2.3×10^{-6} in/in·° F. (about 8.7×10^{-5} to about 1.9×10^{-4} mm/mm·° C.) and about 7.8 to about 19.6 BTU/hr·ft·° F. (about 13.5 to about 33.9 W/mK), respectively, as compared to nickel-base superalloys whose CTE's and thermal conductivities of generally about 7.3×10^{-6} to about 8.5×10^{-6} in/in·° F. (about 3.3×10^{-4} to about 3.9×10^{-4} mm/mm·° C.) and about 6.8 to about 14.6 BTU/hr·ft·° F. (about 11.8 to about 25.3 W/mK), respectively. Such differences in CTE's can cause considerable differential thermal movement between the vane **22** and platform **24**, particularly during transient engine conditions when thermal conductivity and thermal mass also come into play.

[0019] From FIGS. **3** through **5**, the vane **22** is seen as being supported from a surface **26** of the platform **24**. In the particular embodiment shown in FIGS. **3** through **5**, the platform **24** would be oriented radially inward from the vane **22** within the engine, and therefore may be referred to as the inner platform (or band) of the nozzle segment **20**. When installed on a gas turbine engine, the vane **22** and surface **26** of the platform **24** are directly impinged by hot combustion gases discharged by the combustor (not shown) and flowing along the hot gas path of the engine. As such, the vane **22** and platform **24** are both subjected to intense heating during engine operation. Consequently, a bleed air system may be employed that draws a portion of the compressed air from the engine's compressor (not shown) to cool the vane **22** and platform **24**, such as through backside cooling of the platform **24** by directing bleed air at the inner surface **28** of the platform **24** opposite the vane **22**, and/or by flowing bleed air through the vane **22**, a portion of which may be optionally discharged through film cooling holes (not shown) on the surface of the vane **22**. Such cooling techniques are well known in the art, and therefore do not require further explanation.

[0020] As evident from FIGS. **3** through **5**, the platform **24** is configured to include fins **30** protruding from its inner surface **28**. The fins **30** are of a sufficient size to serve as stiffeners that increase the rigidity of the platform **24**, thereby allowing the cross-sectional thickness of the plat-

form **24** to be minimized to reduce the thermal mass of the platform **24**. As a result, the thermal inertia of the platform **24** is reduced, promoting more rapid heat transfer from the platform **24** with backside cooling. The fins **30** also preferably serve a secondary role of promoting radiation heat transfer from the platform **24** as a result of increased surface area from which heat can be radiated. Suitable dimensions for the fins **30** along the length of the platform **24** generally include a thickness (parallel to the surface **28**) of about 2 to about 3 mm, and a height (normal to the surface **28**) of about 2.5 to about 10 mm. In the embodiment depicted in FIGS. **3** through **5**, the heights of the fins **30** increase immediately below the vane **22** to structurally accommodate a recess **38** (described in greater detail below) defined in the outer surface **26** of the platform **24**. To promote a more uniform temperature within the platform **24**, the fins **30** are preferably configured to define a substantially uniform pattern, such as the parallel pattern shown in FIG. **3** through **5**, with a suitable uniform spacing between fins **30** of about 6 to about 13 mm. Because of the increased stiffness contributed by the fins **30** to the platform **24**, the cross-sectional thickness of the platform **24** (excluding the fins **30**) can be reduced by, for example, about 15 to about 25 percent while maintaining the same level of stiffness, and simultaneously resulting in a thermal mass reduction of about 10 to 20% or more for the platform **24**. The fins **30** preferably extend the full circumferential length of the platform **24**, and are integrally formed with the remainder of the platform **24** such as during a casting process of any type known and used to produce platforms for gas turbine engine nozzle assemblies. Alternatively, the fins **30** could be formed separately and attached by welding, brazing, etc.

[**0021**] The vane **22** is shown in FIGS. **3** through **5** as mounted to the platform **24** with an interface structure **32** that provides a resilient, low thermal conductivity path between the vane **22** and platform **24**. The interface structure **32** is represented as including a seal **34** and saddle **36**, both of which are shown as being nested in the aforementioned recess **38** defined in the outer surface **26** of the platform **24**. The recess **38** provides positive axial and tangential retention of the vane **22**, the effect of which may be promoted by forming the recess **38** to extend through the inner surface **28** of the platform **24** and into the taller fins **30** shown immediately below the recess **38** in FIG. **4**. Together the seal **34** and saddle **36** are shown in FIG. **4** as completely filling the recess **38** and continuous between the opposing surfaces of the vane **22** and platform **24**.

[**0022**] The seal **34** primarily provides the desired resilient interconnection between the vane **22** and platform **24**, while also serving to inhibit gas leakage and heat transfer between the vane **22** and the platform **24**. The seal **34** is preferably in the form of what may be termed a cloth seal, meaning a fabric-type sheet material woven from fibers. To withstand the high temperatures of the combustion gases within a high pressure turbine, the fibers are preferably formed of an oxide dispersion strengthened (ODS) material, though the use of other high-temperature materials is foreseeable. To have sufficient resilience and provide a desired level of thermal insulation, the seal **34** is preferably at least 2 to 3 mm thick (normal to the surface **26**) and a porosity of about 0.5 to about 1.0%. Furthermore, the seal **34** is preferably continuous beneath the vane **22**, in contrast to the use of annular-shaped rope seals that surround the base of CMC vanes as proposed in the past. The seal **34** must also be sufficiently strong and stiff to resist compaction when under a compressive load between the vane **22** and platform **24**. In view of the above considerations, an example of a material suitable for use as the seal **34** is an ODS FeCrAl alloy commercially available from Plansee GmbH under the name PM2000.

[**0023**] The saddle **36** is preferably formed of a ceramic-based material, more preferably a precast monolithic ceramic material such as SiC. A precast monolithic is believed to be preferred over a CMC material because of the desire for relatively precise control of the geometry of the saddle **36**, such as small radii fillets joining the portions of the saddle **36** parallel to and normal to the surface **26** of the platform **24**. Without interfering with the resilient connection provided by the seal **34**, the saddle **36** provides for positive retention of the vane **22** to the platform **24** by abutting a stepped shoulder **42** defined by the recess **38**. The abutting-supporting arrangement between the edge of the saddle **36** and the shoulder **42**, in combination with appropriate support at the end of the vane **22** opposite the platform **24**, also inhibits compaction of the seal **34** by the saddle **36**. In the preferred embodiment, the saddle **36** does not intentionally compress the seal **34**. The ceramic material of the saddle **36** also provides additional thermal insulation within the interface structure **32** to inhibit heat transfer between the vane **22** and the platform **24**. To have sufficient thickness for structural strength, the portions of the saddle **36** parallel to and normal to the surface **26** of the platform **24** are each preferably at least 2.5 to 5 mm thick. The depth for the stepped shoulder **42** below the surface **26** of the platform **24** is preferably equal to the thickness of the portion of the saddle **36** within the recess **38** so that that portion of the saddle **36** is generally flush with the surface **26**.

[**0024**] The saddle **36**, vane **22**, and recess **38** in the surface **26** of the platform **24** are shown as having complementary configurations that form shiplap joints therebetween, as evident from FIG. **4**. In particular, both the seal **34** and saddle **36** is depicted as having L-shaped cross-sections that nest with each other and with a recess **40** defined in a wall of the vane **22**, defining overlaps in both the plane parallel to the surface **26** and the plane normal to the surface **26**. The presence of in-series shiplap joints serves to reduce gas leakage between the vane **22** and platform **24**.

[**0025**] As noted above, the fins **30** serve to reduce the temperature of the platform **24** by promoting radiation heat transfer from the platform **24** and reducing the thermal mass of the platform **24**. With the combination of the seal **34** and saddle **36** represented in FIGS. **3** through **5**, the interface structure **32** thermally insulates the vane **22** from the platform **24**, thereby reducing thermal gradients within the vane **22** that could cause structural damage. The interface structure **32** further enables the vane **22** to be secured to the platform **24** in a manner that allows the vane **22** to expand and contract relative to the platform **24** during temperature excursions with reduced thermal-induced strains and stresses within the vane **22** that could cause the vane **22** to fracture during engine operation. More particularly, the vane **22** is able to expand and contract both radially and laterally, the latter of which includes the circumferential and axial directions of the engine. The end of the vane **22** opposite the platform **24** can be secured with a second platform (corresponding to the outer platform **14** of FIG. **1**) in the same manner or optionally in a manner consistent with the prior art, including the use of more rigid attachment techniques.

With either approach, the interface structure **32** can potentially provide a sufficiently resilient connection between the vane **22** and its platform **24** to avoid the prior practice of constructing nozzle assemblies from multiple nozzle segments such as that shown in FIGS. **1** and **2**, and instead forming the inner platform **24** (as well as the outer platform) as a single continuous ring.

[**0026**] While the invention has been described in terms of a particular embodiment, it is apparent that other forms could be adopted by one skilled in the art. For example, the vane **22**, platform **24**, fins **30**, and interface structure **32** could be configured differently from that shown in the Figures while still achieving one or more of the intended objects of the invention. Accordingly, the scope of the invention is to be limited only by the following claims.

What is claimed is:

1. An assembly comprising:

a first body formed of a metallic material and having a first surface and an oppositely-disposed second surface;

a second body formed of a ceramic-based material and supported by the first body from the first surface thereof, the first and second bodies being located in a hot gas path such that the second body and the first surface of the first body are directly impinged by flowing hot gases; and

a substantially uniform pattern of fins protruding from the second surface of the first body, the fins being of sufficient size to increase the rigidity of the first body and promote heat transfer from the first body.

2. An assembly according to claim 1, wherein the fins are integrally formed with the first body.

3. An assembly according to claim 1, wherein the first body has a recess in the first surface thereof, the assembly further comprising an interface structure between the first and second bodies, the interface structure comprising a resilient sealing member received in the recess and disposed between the first and second bodies, the interface structure further comprising a ceramic saddle formed separately from the first and second bodies, received in the recess with the resilient sealing member, and disposed between the resilient sealing member and the second body.

4. An assembly according to claim 3, wherein the ceramic saddle and the second body define a shiplap joint therebetween.

5. An assembly according to claim 3, wherein the ceramic saddle and the resilient sealing member define a shiplap joint therebetween.

6. An assembly according to claim 3, wherein the resilient sealing member comprises a fabric sheet material that is continuous between the vane and the platform.

7. An assembly according to claim 3, wherein the ceramic saddle is formed of a monolithic ceramic material.

8. An assembly according to claim 1, wherein the assembly is a nozzle segment of a gas turbine engine, the first body is a platform of the nozzle segment, and the second body is a vane of the nozzle segment.

9. An assembly according to claim 8, further comprising a third body that supports the second body opposite the first body such that the second body is between the first and third bodies.

10. An assembly comprising:

a first body formed of a metallic material and having a first surface, an oppositely-disposed second surface, and a recess in the first surface;

a second body formed of a ceramic-based material and supported by the first body from the first surface thereof, the first and second bodies being located in a hot gas path such that the second body and the first surface of the first body are directly impinged by flowing hot gases; and

an interface structure between the first and second bodies, the interface structure comprising a resilient sealing member received in the recess and disposed between the first and second bodies, the interface structure further comprising a ceramic saddle formed separately from the first and second bodies, received in the recess with the resilient sealing member, and disposed between the resilient sealing member and the second body.

11. An assembly according to claim 10, wherein the ceramic saddle and the second body define a shiplap joint therebetween.

12. An assembly according to claim 10, wherein the ceramic saddle and the resilient sealing member define a shiplap joint therebetween.

13. An assembly according to claim 10, wherein the resilient sealing member comprises a fabric sheet material that is continuous between the vane and the platform.

14. An assembly according to claim 10, wherein the ceramic saddle is formed of a monolithic ceramic material.

15. An assembly according to claim 10, further comprising a substantially uniform pattern of fins protruding from the second surface of the first body, the fins being of sufficient size to increase the rigidity of the first body and promote heat transfer from the first body.

16. An assembly according to claim 10, wherein the assembly is a nozzle segment of a gas turbine engine, the first body is a platform of the nozzle segment, and the second body is a vane of the nozzle segment.

17. An assembly according to claim 10, further comprising a third body that supports the second body opposite the first body such that the second body is between the first and third bodies.

18. A nozzle assembly of a gas turbine engine, the nozzle assembly comprising:

first and second platforms cast from a metallic material, each of the first and second platforms having first surfaces facing a hot gas path of the gas turbine engine so as to be directly impinged by flowing combustion gases, the first platform having a recess in the first surface thereof and integrally-cast fins protruding from a second surface thereof opposite the first surface, the fins being of sufficient size to increase the rigidity of the first platform and promote heat transfer from the first platform;

a vane formed of a ceramic matrix composite material and supported by the first and second platforms from the first surfaces thereof, the vane being located in the hot gas path such that the vane is directly impinged by the flowing combustion gases; and

an interface structure between the first platform and the vane, the interface structure comprising a resilient sealing member received in the recess in the first surface of the first platform and disposed between the first platform and the vane, the interface structure further comprising a saddle formed of a monolithic ceramic material, received in the recess with the resilient sealing member, and disposed between the resilient sealing member and the vane, the ceramic saddle and the vane defining a first shiplap joint therebetween, the ceramic saddle and the resilient sealing member defining a second shiplap joint therebetween.

19. A nozzle assembly according to claim 18, further comprising means for impinging the second surface of the first platform with cooling air.

20. A nozzle assembly according to claim 18, wherein the resilient sealing member comprises a fabric sheet material that is continuous between the vane and the platform.

* * * * *