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(54) **Tilted turbine vane with impingement cooling**

(57) A turbine airfoil (10) having enhanced cooling capabilities. The turbine vane (10) may be configured such that when a generally elongated airfoil (12) of the turbine vane (10) is attached to a turbine engine, a longitudinal axis (14) of the generally elongated airfoil (12) may be positioned nonparallel relative to a radial axis (16) of the turbine engine in which the turbine vane (10) is mounted. In this position, cooling orifices (18) may be positioned in a region (48) that is typically a dead zone in a conventional turbine vane where no cooling occurs. In one embodiment, a plurality of cooling orifices (18) in an inner shroud (20) of the turbine vane (10) may be positioned between an outer edge (22) of the inner shroud (20) in closest proximity to a suction side (24) of the airfoil (12) near a leading edge (26) of the airfoil (12) and an intersection (28) between the suction side (24) and the inner shroud (20).

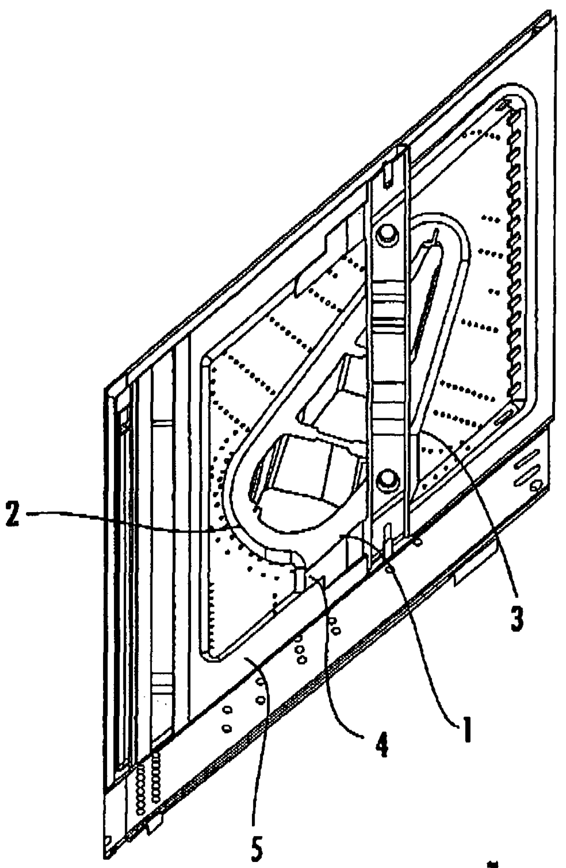

FIG. 1
(PRIOR ART)

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Description

FIELD OF THE INVENTION

[0001] This invention is directed generally to turbine airfoils, and more particularly to turbine vanes having internal cooling channels for passing fluids, such as air, to cool the airfoils.

BACKGROUND

[0002] Typically, gas turbine engines include a compressor for compressing air, a combustor for mixing the compressed air with fuel and igniting the mixture, and a turbine blade assembly for producing power. Combustors often operate at high temperatures that may exceed 2,500 degrees Fahrenheit. Typical turbine combustor configurations expose turbine vane and blade assemblies to these high temperatures. As a result, turbine vanes and blades must be made of materials capable of withstanding such high temperatures. In addition, turbine vanes and blades often contain cooling systems for prolonging the life of the vanes and blades and reducing the likelihood of failure as a result of excessive temperatures.

[0003] Typically, turbine vanes are formed from an elongated portion forming a vane having one end configured to be coupled to a vane carrier and an opposite end configured to be movably coupled to an inner endwall. The vane is ordinarily composed of a leading edge, a trailing edge, a suction side, and a pressure side. The inner aspects of most turbine vanes typically contain intricate cooling circuits forming a cooling system. The cooling circuits in the vanes receive air from the compressor of the turbine engine and pass the air through the ends of the vane adapted to be coupled to the vane carrier. The cooling circuits often include multiple flow circuits that control metal temperature to ensure component durability and functionality. At least some of the air passing through these cooling circuits is exhausted through orifices in the leading edge, trailing edge, suction side, and pressure side of the vane.

[0004] As shown in Figure 1, turbine vanes include inner and outer shrouds adapted to position the turbine vane in a turbine engine. Typically, a shroud includes a plurality of film cooling holes extending through the shroud forming pathways between internal cooling chambers in the turbine vane and regions outside of the turbine vane for further cooling applications. As shown in Figure 1, the plurality of film cooling holes may be positioned across the shroud. However, because of physical limitations of the turbine vane and the shroud, cooling orifices are not located in the region labeled as a dead zone 1. In the conventional configuration, the leading edge 2 near the suction side 3 of the turbine vane is positioned in close proximity to an outer edge 4 of the shroud 5. The wall forming the turbine vane prevents cooling film cooling holes from being positioned within the dead zone. Thus, the dead zone region of the conventional turbine vane does not receive adequate cooling. Furthermore, the dead zone region is often formed from relatively thick materials, thereby resulting in increased cooling requirements to cool the dead zone region. The lack of cooling in the dead zone and the thickness of the material in the turbine vane in the dead zone reduces the efficiency of the turbine engine and increases the chances of premature failure. Thus, a need exists for efficiently providing adequate cooling to the dead zone region of a turbine vane.

SUMMARY OF THE INVENTION

[0005] This invention relates to a turbine vane having enhanced cooling capabilities for improving the durability of the turbine vane, for increasing the efficiency of a turbine engine in which the turbine vane is mounted, and for providing other advantages as well. The turbine vane may be configured such that when a generally elongated airfoil of the turbine vane is attached to a turbine engine, a longitudinal axis of the generally elongated airfoil may be positioned non parallel relative to a radial axis of the turbine engine in which the turbine vane is mounted. In this position, cooling orifices may be positioned such that at least a portion of the plurality of cooling orifices in an inner shroud of the turbine vane may be positioned in an area that typically is a dead zone in conventional vanes. In particular, the cooling orifices may be positioned between an outer edge of the inner shroud in closest proximity to a suction side of the generally elongated hollow airtoil-near the leading edge of the generally elongated hollow airfoil and an intersection between the suction side of the generally elongated hollow airfoil and the inner shroud. Positioning the cooling orifices in this region provides a cooling mechanism to a region that in conventional turbine vanes is typically not cooled in conventional vanes. This invention results in reduced temperature gradients during operation relative to conventional designs.

[0006] In at least one embodiment, the turbine vane may be formed from a generally elongated hollow airfoil formed from an outer wall, and having a leading edge, a trailing edge, a pressure side, a suction side, an outer shroud coupled to an end of the generally elongated hollow airfoil and adapted to be coupled to a hook attachment, an inner shroud coupled to the generally elongated hollow airfoil at an end that is opposite to the outer shroud and adapted to be coupled to an inner endwall, and a cooling system formed from at least one cooling channel extending through the generally elongated airfoil and including a plurality of cooling orifices in the inner shroud. The generally elongated airfoil may be coupled to the inner attachment end such that a longitudinal axis of the generally elongated airfoil may be positioned nonparallel with a radial axis of a turbine engine in which the turbine airfoil is configured to be mounted.

[0007] The inner shroud of the turbine vane may include at least one inner shroud cooling channel extending generally along a non-gaspath surface of the inner shroud. The inner shroud cooling channel may be defined by an outer wall and an impingement plate. A plurality of cooling orifices may be positioned in the outer wall. The plurality of cooling airfoils may surround an intersection between the generally elongated hollow airfoil and the inner shroud. In addition, a plurality of film cooling orifices may be positioned in the impingement plate. Cooling fluids flowing through the cooling orifices and the film cooling orifices may be metered twice while flowing through the these orifices. Such metering enables an accurate prediction to be made of a cooling fluid flow rate through the turbine vane.

[0008] The generally elongated airfoil may be coupled to the inner attachment end such that the longitudinal axis of the generally elongated airfoil is tilted between about one degree and about seven degrees from a radial axis of a turbine engine in which the turbine airfoil is configured to be mounted. In particular, the generally elongated airfoil may be tilted about four degrees from the radial axis of a turbine engine. Such a position enables cooling holes to be positioned in a conventional dead region.

[0009] An advantage of this invention is that the cooling fluids are used efficiently for impingement cooling, convective cooling within an impingement cooling chamber in an inner shroud of the turbine vane, and for cooling in locations where the coolant is injected onto a non-gaspath surface of the shroud.

[0010] Another advantage of this invention is that the tilted orientation of the airfoil relative to a radial axis of the turbine engine eliminates conventional dead zones. In addition, areas in conventional airfoils that often have thick walls are eliminated such that these areas have thinner walls in the invention. The thinner walls reduce the thermal gradient across the walls, thereby resulting in reduced thermal stresses on the turbine blade. This is important in the trailing edge of the airfoil, where large radial stiffness and ineffective cooling has historically caused vane cracking.

[0011] Yet another advantage of this invention is that creating the tilted orientation in the turbine airfoil does not incur additional manufacturing costs. Moreover, machining costs for the shrouds of the airfoil of this invention may be less than machining costs associated with conventional shrouds.

[0012] Another advantage of this invention is that the turbine airfoil together with the improved cooling fluid flow scheme creates a robust cooling configuration that has more flexibility in accommodating variations in combustion tuning, power plant operation protocols, and manufacturing process variations. In addition, the configuration of the cooling system in the airfoil directs cooling fluids to regions in conventional airfoils that required cooling but lacked cooling systems.

[0013] These and other embodiments are described in more detail below.

BRIEF DESCRIPTION OF THE DRAWINGS

[0014] The accompanying drawings, which are incorporated in and form a part of the specification, illustrate embodiments of the presently disclosed invention and, together with the description, disclose the principles of the invention.

Figure 1 is a top view of an inner shroud of a conventional turbine vane in which a dead zone of a cooling system exists between the turbine airfoil and an outer edge of the shroud.

Figure 2 is a perspective view of turbine vane having aspects of this invention.

Figure 3 is a bottom view of the inner shroud shown in Figure 2.

Figure 4 is an exploded perspective view of the turbine vane shown in Figure 2, rotated 180 degrees to view the non-gaspath surface of the inner shroud.

DETAILED DESCRIPTION OF THE INVENTION

[0015] As shown in Figures 2-4, this invention is directed to a turbine vane 10 having enhanced cooling capabilities for improving the durability of the turbine vane 10, for increasing the efficiency of a turbine engine in which the turbine vane 10 is mounted, and for providing other advantages as well. The turbine vane 10 may be configured such that when a generally elongated airfoil 12 of the turbine vane 10 is attached to a turbine engine, a longitudinal axis 14 of the generally elongated airfoil 12 may be positioned nonparallel relative to a radial axis 16 of the turbine engine in which the turbine vane 10 is mounted. In this position, cooling orifices 18 may be positioned such that at least a portion of the plurality of cooling orifices 18 in an inner shroud 20 of the turbine vane 10 may be positioned between an outer edge 22 of the inner shroud 20 in closest proximity to a suction side 24 of the generally elongated hollow airfoil 12 near the leading edge 26 of the generally elongated hollow airfoil 12 and an intersection 28 between the pressure side 32 of the generally elongated hollow airfoil 12 and the inner shroud 20. Positioning the cooling orifices 18 in this region provides a cooling mechanism to a region that in conventional turbine vanes is typically not cooled. This invention results in reduced temperature gradients during operation relative to conventional designs.

[0016] As shown in Figures 2 and 4, the turbine vane 10 may be formed from the generally elongated airfoil 12 having a gaspath surface 30 adapted for use, for example, in an axial flow turbine engine. Gaspath surface 30 may have a generally concave shaped portion forming a pressure side 32 and a generally convex shaped portion forming the suction side 24. The turbine vane 10 may also include an outer shroud 34 adapted to be coupled to a hook attachment and may include the inner shroud 20 adapted to be coupled to an inner endwall. The airfoil

22 may also include the leading edge 26 and a trailing edge 40.

[0017] As shown in Figure 2, the turbine vane 10 may be tilted relative to the radial axis 16 of a turbine engine. In particular, the turbine vane 10 may be tilted such that the longitudinal axis 14 of the turbine vane 10 is positioned at an angle 42 of between about one degree and about seven degrees relative to the radial axis 16 of the turbine engine. In at least one embodiment, as shown in Figure 2, the turbine vane 10 may be positioned at an angle 42 of about four degrees. Positioning the turbine vane 10 at an angle 42 relative to the radial axis enables cooling orifices 18 to be positioned between the outer edge 22, as shown in Figure 3, of the inner shroud 20 in closest proximity to the suction side 24 of the generally elongated hollow airfoil 12 near the leading edge 26 of the generally elongated hollow airfoil 12 and an intersection 28 between the suction side 24 of the generally elongated hollow airfoil 12 and the inner shroud 20. Positioning the cooling orifices 18 in this region 48 enables a cooling system 44 of the turbine vane 10 to pass cooling fluids, such as, but not limited to, air through this region 48 through one or more cooling cavities 15. Region 48 is often not effectively cooled in conventional turbine vanes and often results in premature material failure and large temperature gradients. Thus, tilting of the turbine airfoil 10 relative to the radial axis 16 allows for locating cooling orifices 18 and forming of shroud cooling chambers 46 in a region 48 that typically is a dead zone in conventional turbine vanes.

[0018] As shown in Figure 3, the inner shroud 20 may include one or more shroud cooling chambers 46 positioned in the inner shroud 20 of the turbine vane 10. In at least one embodiment, the inner cooling chamber 46 may extend generally along an non-gaspath surface 50 of the inner shroud 20 and may be defined by an outer wall 52. The non-gaspath surface 50 may include a recess, as shown in Figure 3. The shroud cooling chamber 46 positioned in the inner shroud 20 enables the outer walls 52 to be thinner than conventional configurations and thinner than configurations not including the shroud cooling chamber 46. The shroud cooling chamber 46 may be formed with an impingement plate 54 configured to be attached to the outer wall 52, as shown in Figure 4. The impingement plate 54 may include a plurality of film cooling orifices 56 providing a fluid pathway between the cavity 46 and non-gaspath surfaces of components of the turbine vane 10. The cooling fluids impinge on non-gaspath surfaces of components of the turbine vane 10 to reduce the temperature of the materials forming the turbine vane 10. Film cooling orifices 18 may be sized to meter cooling fluids flowing from the turbine vane 10. Cooling orifices 18 in the inner shroud 20 may also be sized to meter the flow of cooling fluids from the turbine vane 10. Thus, the cooling orifices 18 may used together with the film cooling orifices 56 in the impingement plate 54 to meter the flow of cooling fluids twice within the turbine vane 10.

[0019] During operation, cooling fluids flow through internal aspects of the turbine vane 10 and pass through the cooling orifices 18 in the inner shroud 20. The cooling fluids are metered as the fluids pass through the cooling orifices 18. The cooling fluids then impinge on the impingement plate 54. The cooling fluids then flow through the film cooling orifices 56 in the impingement plate 54 and impinge upon other components of the turbine cooling system. Such metering enables the flow of cooling fluids to be accurately controlled. Therefore, the cooling fluid flow may be accurately predicted and accounted for during design. The shroud cooling chamber 46 results in the temperature gradient in the outer wall 52 during turbine engine operation being less than in a configuration in which an shroud cooling chamber 46 is not used. The resulting lower temperature gradient creates less thermal stress than the thermal stress that develops in turbine airfoils without the shroud cooling chambers 46 in the inner shroud 20.

[0020] The foregoing is provided for purposes of illustrating, explaining, and describing embodiments of this invention. Modifications and adaptations to these embodiments will be apparent to those skilled in the art and may be made without departing from the scope or spirit of this invention.

Claims

1. A turbine airfoil (10), **characterized in that**: a generally elongated hollow airfoil (12) formed from an outer wall, and having a leading edge (26), a trailing edge (40), a pressure side (32), a suction side (24), an outer shroud (34) coupled to an end of the generally elongated hallow airfoil (12) and adapted to be coupled to a hook attachment, an inner shroud (20) coupled to the generally elongated hollow airfoil (12) at an end that is opposite to the outer shroud (34) and adapted to be coupled to an inner endwall, and a cooling system (44) formed from at least one cooling channel (15) extending through the generally elongated airfoil (12) and including a plurality of cooling orifices (18) in the inner shroud (20); and wherein the generally elongated airfoil (12) is coupled to the inner shroud (20) such that a longitudinal axis (14) of the generally elongated airfoil (12) is positioned nonparallel with a radial axis (16) of a turbine engine in which the turbine airfoil (10) is configured to be mounted.

2. The turbine airfoil (10) of claim 1, further **characterized in that** at least one impingement plate (54) forming at least one inner shroud cooling channel (46) in the inner shroud (20) that extends generally along a non-gaspath surface (50) of the inner shroud (20), wherein the at least one inner shroud cooling channel (46) is defined by an outer wall (52) and an impingement plate (54).

3. The turbine airfoil (10) of claim 2, further **characterized in that** a plurality of film cooling orifices (56) extend through the impingement plate (54).

4. The turbine airfoil (10) of claim 1, **characterized in that** at least a portion of the plurality of cooling orifices (18) in the inner shroud (20) are positioned between an outer edge of the inner shroud (20) in closest proximity to the suction side (24) of the generally elongated hollow airfoil (12) near the leading edge (26) of the generally elongated hollow airfoil (12) and an intersection (28) between the suction side (24) of the generally elongated hollow airfoil (12) and the inner shroud (20).

5. The turbine airfoil (10) of claim 1, **characterized in that** the plurality of cooling orifices (18) surround an intersection (28) between the generally elongated hollow airfoil (12) and the inner shroud (20).

6. The turbine airfoil (10) of claim 1, **characterized in that** the generally elongated airfoil (12) is coupled to the inner attachment end such that the longitudinal axis (14) of the generally elongated airfoil (12) is tilted between about one degree and about seven degrees from the radial axis (16) of a turbine engine in which the turbine airfoil (10) is configured to be mounted.

7. The turbine airfoil (10) of claim 6, **characterized in that** the generally elongated airfoil (12) is coupled to the inner attachment end such that the longitudinal axis (14) of the generally elongated airfoil (12) is tilted about four degrees from the radial axis (16)of a turbine engine in which the turbine airfoil (10) is configured to be mounted.

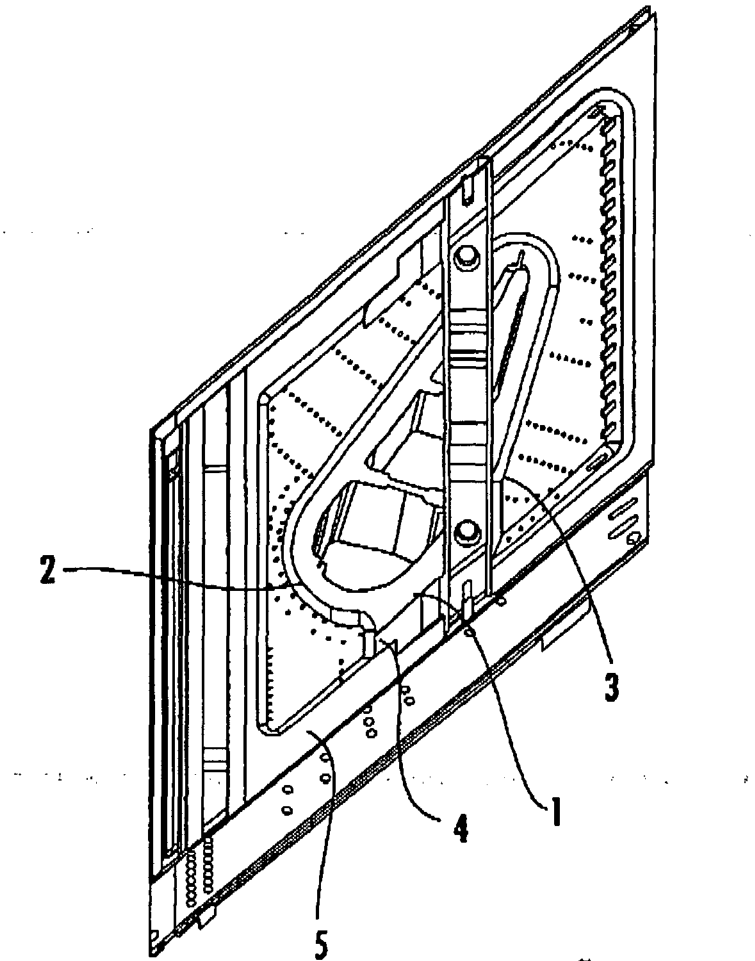

FIG. 1
(PRIOR ART)

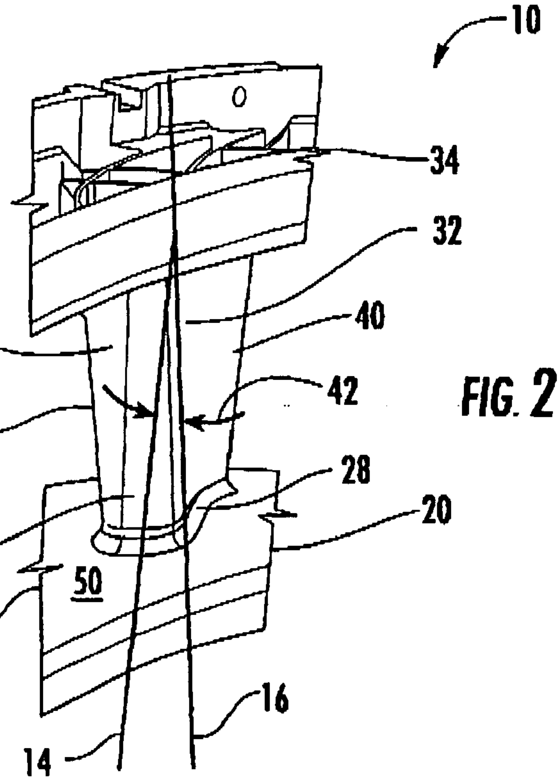
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FIG. 2

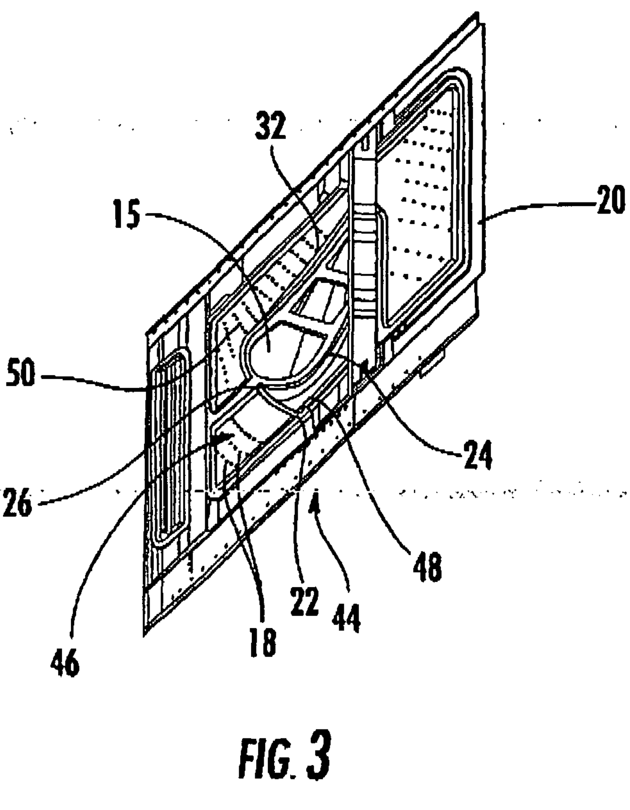
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FIG. 3

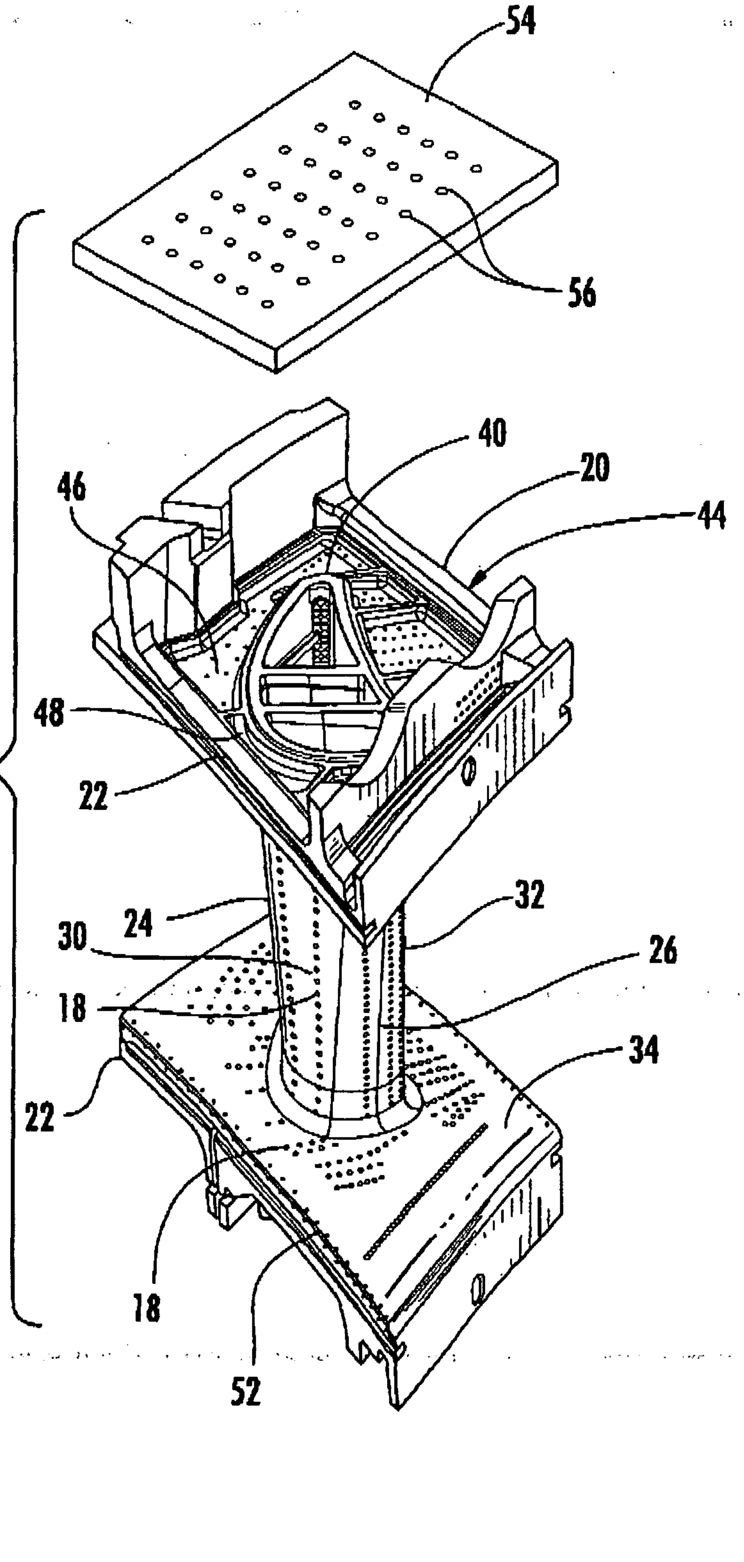

FIG. 4