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(54) **Turbine nozzle with integral impingement blanket**

(57) A turbine nozzle segment includes: (a) an arcuate outer band segment (50); (b) a hollow, airfoil-shaped turbine vane (30) extending radially inward from the outer band segment (50); (c) a manifold cover (54) secured to the outer band such that the manifold cover (54) and the outer band segment (50) cooperatively define an im-

pingement cavity; and (d) an impingement blanket (62) disposed in the impingement cavity, the impingement blanket (62) having at least one impingement hole formed therethrough which is arranged to direct cooling air at the outer band segment (50). A method is provided for impingement cooling the outer band segment (50).

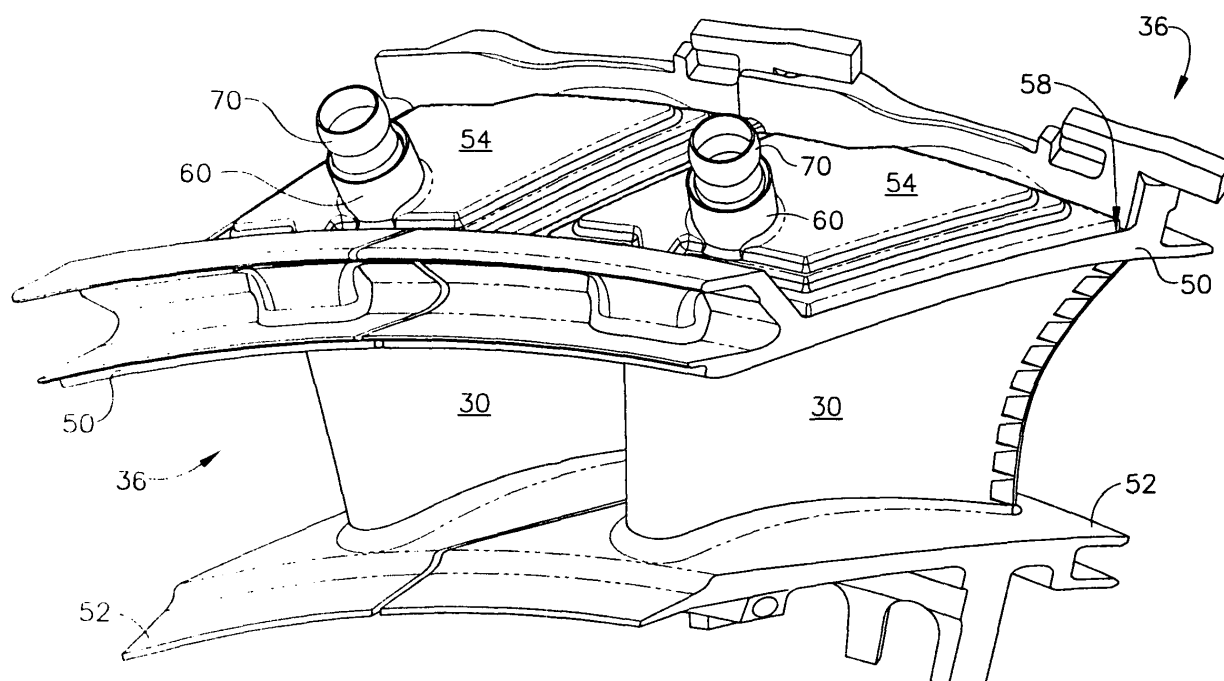


FIG. 2

Description

BACKGROUND OF THE INVENTION

[0001] This invention relates generally to gas turbine engine turbines and more particularly to methods for cooling turbine sections of such engines.

[0002] A gas turbine engine includes a turbomachinery core having a high pressure compressor, combustor, and high pressure or gas generator turbine in serial flow relationship. The core is operable in a known manner to generate a primary gas flow. In a turbojet or turbofan engine, the core exhaust gas is directed through a nozzle to generate thrust. A turboshaft engine uses a low pressure or "work" turbine downstream of the core to extract energy from the primary flow to drive a shaft or other mechanical load.

[0003] The gas generator turbine includes annular arrays ("rows") of stationary vanes or nozzles that direct the gases exiting the combustor into rotating blades or buckets. Collectively one row of nozzles and one row of blades make up a "stage". Typically two or more stages are used in serial flow relationship. These components operate in an extremely high temperature environment, and must be cooled by air flow to ensure adequate service life. Typically, the air used for cooling is extracted from one or more points in the compressor. These bleed flows represent a loss of net work output and/or thrust to the thermodynamic cycle. They increase specific fuel consumption (SFC) and are generally to be minimized as much as possible.

[0004] Prior art gas generator turbine nozzles have been cooled either using a "spoolie" fed manifold cover or a continuous impingement ring with a spoolie-fed airfoil insert. For the first system, air is fed into a manifold above the outer band, and then flows into the airfoil without directly cooling the outer band. The second configuration utilizes a separate impingement ring to cool the outer band, but this flow is susceptible to leakage through the gaps between adjacent nozzle segments. In either case, the turbine nozzle cooling is less efficient than desired.

BRIEF SUMMARY OF THE INVENTION

[0005] These and other shortcomings of the prior art are addressed by the present invention, which provides independent impingement cooling for individual turbine nozzle outer band segments.

[0006] According to one aspect of the invention, a turbine nozzle segment includes: (a) an arcuate outer band segment; (b) a hollow, airfoil-shaped turbine vane extending radially inward from the outer band segment; (c) a manifold cover secured to the outer band such that the manifold cover and the outer band segment cooperatively define an impingement cavity; and (d) an impingement blanket disposed in the impingement cavity, the impingement blanket having at least one impingement hole formed therethrough which is arranged to direct cooling

air at the outer band segment.

[0007] According to another aspect of the invention, a turbine nozzle assembly for a gas turbine engine includes: (a) a plurality of turbine nozzle segments arranged in an annular array, each turbine nozzle segment having: (i) an arcuate outer band segment; (ii) a hollow, airfoil-shaped turbine vane extending radially inwardly from the outer band segment; (iii) a manifold cover secured to the outer band such that the manifold cover and the outer band segment cooperatively define an impingement cavity; and (iv) an impingement blanket disposed in the impingement cavity, the impingement blanket having at least one impingement hole formed therethrough which is arranged to direct cooling air at the outer band segment; (b) an annular supporting structure surrounding the turbine nozzle segments; and (c) a plurality of generally cylindrical conduits, each conduit connecting one of the manifold covers in independent flow communication with the supporting structure.

[0008] According to another aspect of the invention, a method is provided for cooling a turbine nozzle which includes an array of nozzle segments each having an arcuate outer band with a hollow, airfoil-shaped turbine vane extending radially inward therefrom. The method includes: (a) providing each of the outer bands with a closed impingement cavity having an impingement blanket disposed therein; (b) directing cooling air separately into the impingement cavities; (c) directing cooling air through one or more impingement holes in the impingement blanket against the outer band; and (d) exhausting the cooling air from the impingement cavity.

BRIEF DESCRIPTION OF THE DRAWINGS

[0009] There follows a detailed description of embodiments of the invention by way of example only with reference to the accompanying drawings in which:

Figure 1 is a cross-sectional view of a high pressure turbine section of a gas turbine engine, constructed in accordance with an aspect of the present invention;

Figure 2 is a perspective view of a turbine nozzle shown in Figure 1, with a manifold cover assembled thereto;

Figure 3 is perspective view of an impingement blanket;

Figure 4 is a perspective view of a manifold cover; and

Figure 5 is a perspective view of the impingement blanket of Figure 3 assembled to the manifold cover of Figure 4.

DETAILED DESCRIPTION OF THE INVENTION

[0010] Referring to the drawings wherein identical reference numerals denote the same elements throughout the various views, Figure 1 depicts a portion of a gas generator turbine 10, which is part of a gas turbine engine of a known type. The function of the gas generator turbine 10 is to extract energy from high-temperature, pressurized combustion gases from an upstream combustor (not shown) and to convert the energy to mechanical work, in a known manner. The gas generator turbine 10 drives an upstream compressor (not shown) through a shaft so as to supply pressurized air to the combustor.

[0011] In the illustrated example, the engine is a turboshaft engine and a work turbine would be located downstream of the gas generator turbine 10 and coupled to an output shaft. However, the principles described herein are equally applicable to turboprop, turbojet, and turbofan engines, as well as turbine engines used for other vehicles or in stationary applications.

[0012] The gas generator turbine 10 includes a first stage nozzle 12 which comprises a plurality of circumferentially spaced airfoil-shaped hollow first stage vanes 14 that are supported between an arcuate, segmented first stage outer band 16 and an arcuate, segmented first stage inner band 18. The first stage vanes 14, first stage outer band 16 and first stage inner band 18 are arranged into a plurality of circumferentially adjoining nozzle segments that collectively form a complete 360° assembly. The first stage outer and inner bands 16 and 18 define the outer and inner radial flowpath boundaries, respectively, for the hot gas stream flowing through the first stage nozzle 12. The first stage vanes 14 are configured so as to optimally direct the combustion gases to a first stage rotor 20.

[0013] The first stage rotor 20 includes a array of airfoil-shaped first stage turbine blades 22 extending outwardly from a first stage disk 24 that rotates about the centerline axis of the engine. A segmented, arcuate first stage shroud 26 is arranged so as to closely surround the first stage turbine blades 22 and thereby define the outer radial flowpath boundary for the hot gas stream flowing through the first stage rotor 20.

[0014] A second stage nozzle 28 is positioned downstream of the first stage rotor 20, and comprises a plurality of circumferentially spaced airfoil-shaped hollow second stage vanes 30 that are supported between an arcuate, segmented second stage outer band 32 and an arcuate, segmented second stage inner band 34. The second stage vanes 30, second stage outer band 32 and second stage inner band 34 are arranged into a plurality of circumferentially adjoining nozzle segments 36 (see Figure 2) that collectively form a complete 360° assembly. The second stage outer and inner bands 32 and 34 define the outer and inner radial flowpath boundaries, respectively, for the hot gas stream flowing through the second stage turbine nozzle 34. The second stage vanes 30 are configured so as to optimally direct the combustion gases

to a second stage rotor 38.

[0015] The second stage rotor 38 includes a radial array of airfoil-shaped second stage turbine blades 40 extending radially outwardly from a second stage disk 42 that rotates about the centerline axis of the engine. A segmented arcuate second stage shroud 44 is arranged so as to closely surround the second stage turbine blades 40 and thereby define the outer radial flowpath boundary for the hot gas stream flowing through the second stage rotor 38.

[0016] The segments of the first stage shroud 26 are supported by an array of arcuate first stage shroud hangers 46 that are in turn carried by an arcuate shroud support 48, for example using the illustrated hooks, rails, and C-clips in a known manner.

[0017] The second stage nozzle 28 is supported in part by mechanical connections to the first stage shroud hangers 46 and the shroud support 48. Each second stage vane 30 is hollow so as to be able to receive cooling air in a known fashion.

[0018] Figures 2-5 illustrate the construction of the second stage nozzle 28 in more detail. Figure 2 shows two individual nozzle segments 36 arranged side-by-side, as they would be in the assembled gas generator turbine 10. In the illustrated example, the nozzle segment 36 is a "singlet" casting which includes a segment 50 of the outer band 32, a segment 52 of the inner band 34, and a hollow second stage vane 30. The radially outer end of each outer band segment 50 is closed by a manifold cover 54. The manifold cover 54 (see Figure 4) is a unitary, slightly convex structure which has a lower peripheral edge 56 that matches the radially outer surface 58 of the outer band segment 50, and includes an outwardly-extending inlet tube 60.

[0019] A plate-like impingement blanket 62, best seen in Figure 3, has a plurality of impingement holes 64 formed through it. It may be cast or fabricated from sheet metal. It is placed inside a recess 66 on the radially inner side of the manifold cover 54, as seen in Figure 5, and is secured thereto, for example by brazing, welding, fasteners, or adhesives.

[0020] The manifold cover 54 is secured to the outer surface 58 of the outer band segment 50 so as to form an integral, sealed structure, with the sole inlet for air flow being the inlet tube 60. As seen in Figure 1, the manifold cover 54 and the outer band segment 50 cooperatively define an impingement cavity 68 which is divided into two sections by the impingement blanket 62.

[0021] When assembled, the inlet tube 60 is coupled to a generally cylindrical tube or conduit known as a "spoolie" 70. The spoolie 70 penetrates the shroud support 48 to provide a pathway for cooling air into the interior of the second stage vanes 30, as described in more detail below. One spoolie 70 is provided for each of the inlet tubes 60.

[0022] In operation, compressor discharge air (CDP), at the highest pressure in the compressor, or another suitable cooling air flow, is ducted to the shroud support

48 in a known manner. The CDP air enters the spoolies 66, depicted by the arrows labeled "C" in Figure 1. It then flows through the inlet tubes 60 into the individual impingement cavities 68 of each nozzle segment 36. The cooling air exits the impingement holes 64 as a series of jets, depicted by the arrows "J", which impinge against the outer band segment 50 and cool it. The spent impingement air is then exhausted to the interior of the turbine vane 30, where it may be used for additional cooling in a known manner. The area between the manifold cover 54 and the shroud support 48 is referred to as an outer band cavity 72, and is purged by a separate air flow source.

[0023] This configuration offers several advantages. By integrally joining the impingement blanket 62 to the manifold cover 54, and by joining the manifold cover 54 to the outer band segment 50, the outer band segment 50 can be impingement cooled using high pressure air without the associated inter-segment leakage penalties. This configuration then allows for the use of lower pressure air to purge the nozzle outer band cavities - as the air is at a lower pressure, the total amount of leakage flow will be reduced resulting in a lower performance penalty.

[0024] The foregoing has described cooling arrangements for a turbine nozzle. While specific embodiments of the present invention have been described, it will be apparent to those skilled in the art that various modifications thereto can be made without departing from the spirit and scope of the invention. Accordingly, the foregoing description of the preferred embodiment of the invention and the best mode for practicing the invention are provided for the purpose of illustration only and not for the purpose of limitation, the invention being defined by the claims.

Claims

1. A turbine nozzle segment comprising:
 - (a) an arcuate outer band segment (50);
 - (b) a hollow, airfoil-shaped turbine vane (30) extending radially inward from the outer band segment (50);
 - (c) a manifold cover (54) secured to the outer band such that the manifold cover (54) and the outer band segment (50) cooperatively define an impingement cavity; and
 - (d) an impingement blanket (62) disposed in the impingement cavity, the impingement blanket (62) having at least one impingement hole formed therethrough which is arranged to direct cooling air at the outer band segment (50).
2. The turbine nozzle segment of claim 1 wherein the impingement blanket (62) has a plurality of impingement holes formed therein.
3. The turbine nozzle segment of claim 1 or 2, wherein:
 - (a) the manifold cover (54) has a radially-inwardly facing recess formed therein, and
 - (b) the impingement blanket (62) comprises a plate which is secured to the manifold cover (54) so as to close off the recess.
4. The turbine nozzle segment of claim 3, wherein the impingement blanket (62) is brazed to the manifold cover (54).
5. The turbine nozzle segment of any of the preceding claims, wherein the manifold cover (54) is brazed to the outer band segment (50).
6. The turbine nozzle segment of any of the preceding claims, wherein the manifold cover (54) includes a radially-outwardly extending inlet tube.
7. The turbine nozzle segment of any of the preceding claims, further comprising an arcuate inner band segment disposed at a radially inner end of the turbine vane (30).
8. A turbine nozzle assembly for a gas turbine engine, comprising:
 - (a) a plurality of turbine nozzle segments arranged in an annular array, each turbine nozzle segment comprising:
 - (i) an arcuate outer band segment;
 - (ii) a hollow, airfoil-shaped turbine vane extending radially inward from the outer band segment;
 - (iii) a manifold cover secured to the outer band such that the manifold cover and the outer band segment cooperatively define an impingement cavity; and
 - (iv) an impingement blanket disposed in the impingement cavity, the impingement blanket having at least one impingement hole formed therethrough which is arranged to direct cooling air at the outer band segment;
 - (b) an annular supporting structure surrounding the turbine nozzle segments; and
 - (c) a plurality of generally cylindrical conduits, each conduit connecting one of the manifold covers in independent flow communication with the supporting structure.
9. The turbine nozzle assembly of claim 8, wherein each of the impingement blankets has a plurality of impingement holes formed therein.
10. The turbine nozzle assembly of claim 8 or 9, wherein:

- (a) each of the manifold covers has a radially-inwardly facing recess formed therein, and
- (b) each of the impingement blankets comprises a plate which is secured to the respective manifold cover so as to close off the recess.

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11. The turbine nozzle assembly of claim 10 wherein each of the impingement blankets is brazed to the respective manifold cover.

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12. The turbine nozzle assembly of any of claims 8 to 11, wherein each of the manifold covers is brazed to the respective outer band segment.

13. The turbine nozzle assembly of any of claims 8 to 12, wherein each of the manifold covers includes a radially-outwardly extending inlet tube to which the respective conduit is connected.

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14. The turbine nozzle assembly of any of claims 8 to 13, further comprising an arcuate inner band segment disposed at a radially inner end of each of the turbine vanes.

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15. A method of cooling a turbine nozzle which includes an array of nozzle segments each having an arcuate outer band segment with a hollow, airfoil-shaped turbine vane (30) extending radially inward therefrom, the method comprising:

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- (a) providing each of the outer band segments with a closed impingement cavity having an impingement blanket (62) disposed therein;
- (b) directing cooling air separately into the impingement cavities;
- (c) directing cooling air through one or more impingement holes in the impingement blanket (62) against the outer band segments; and
- (d) exhausting the cooling air from the impingement cavity.

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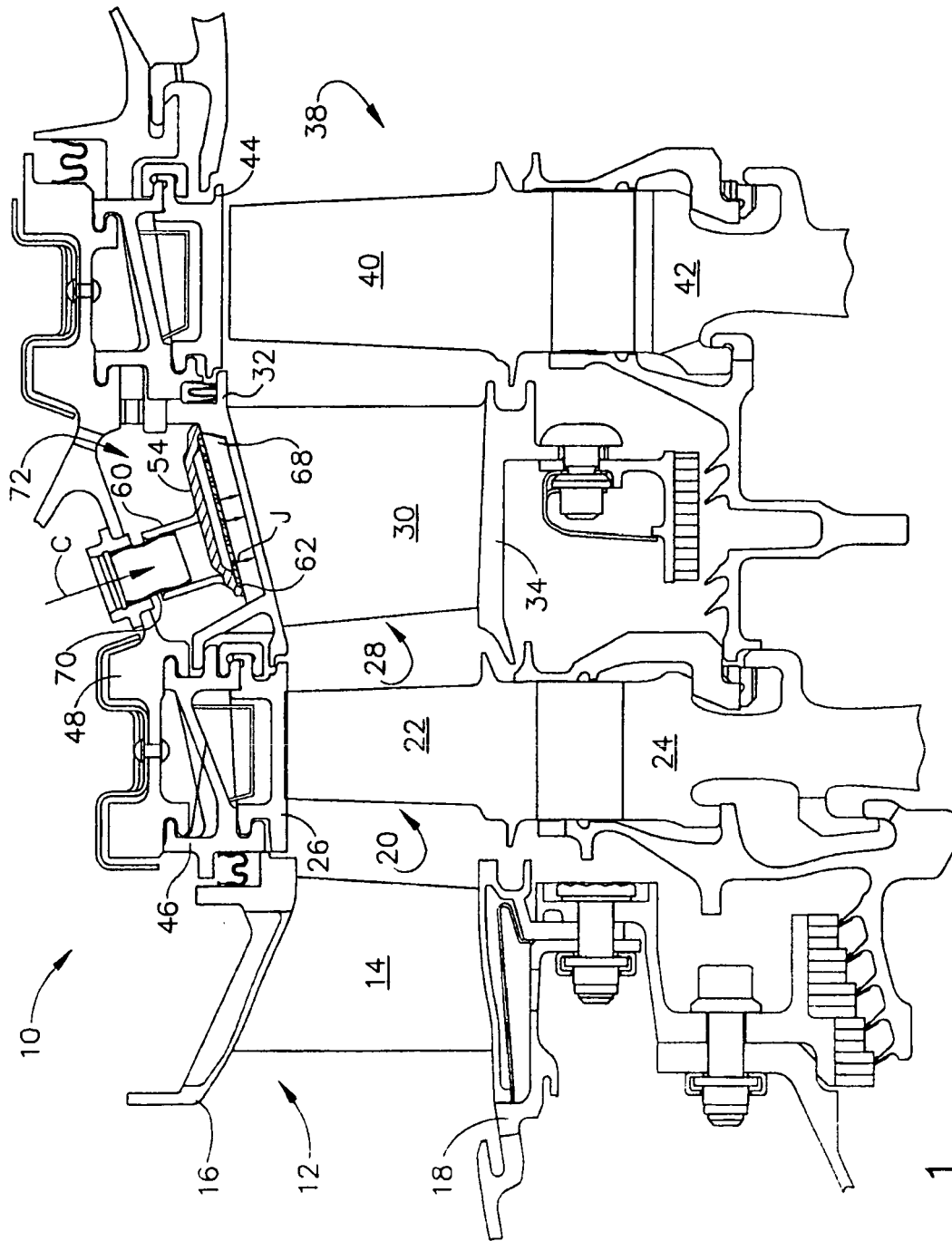


FIG. 1

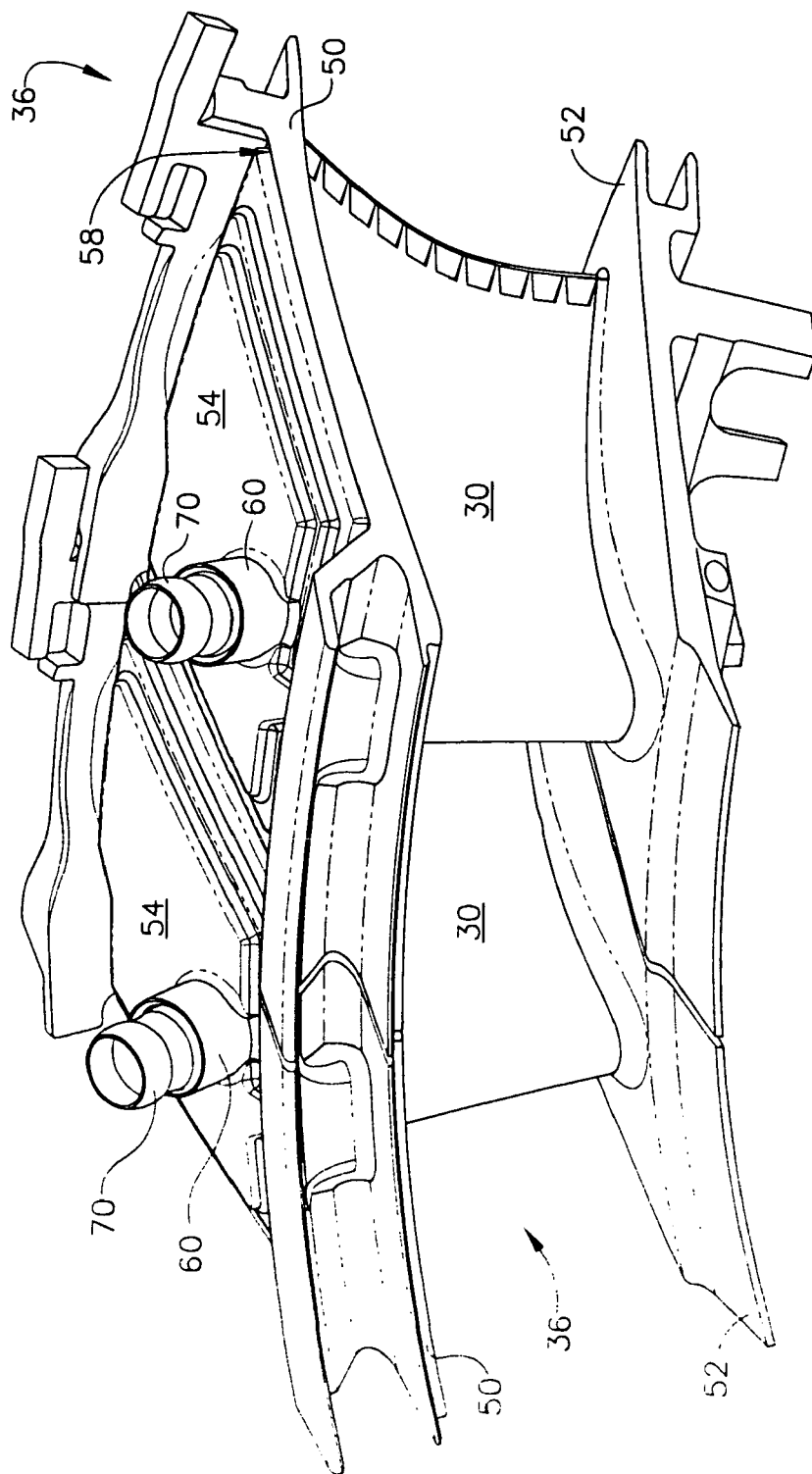


FIG. 2

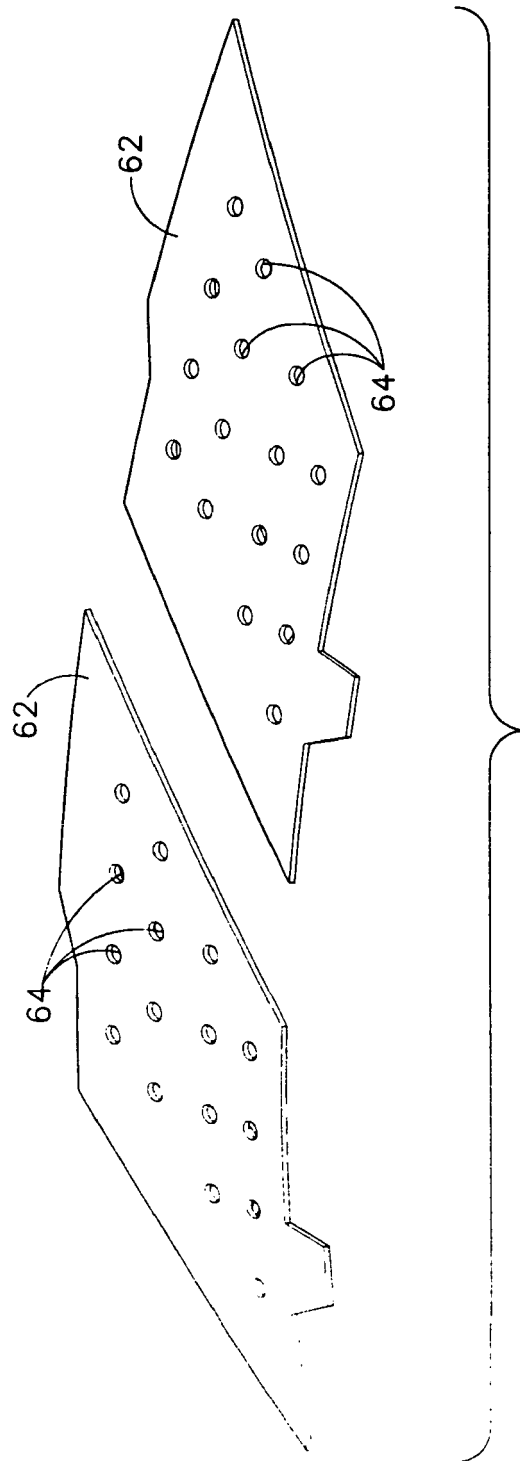


FIG. 3

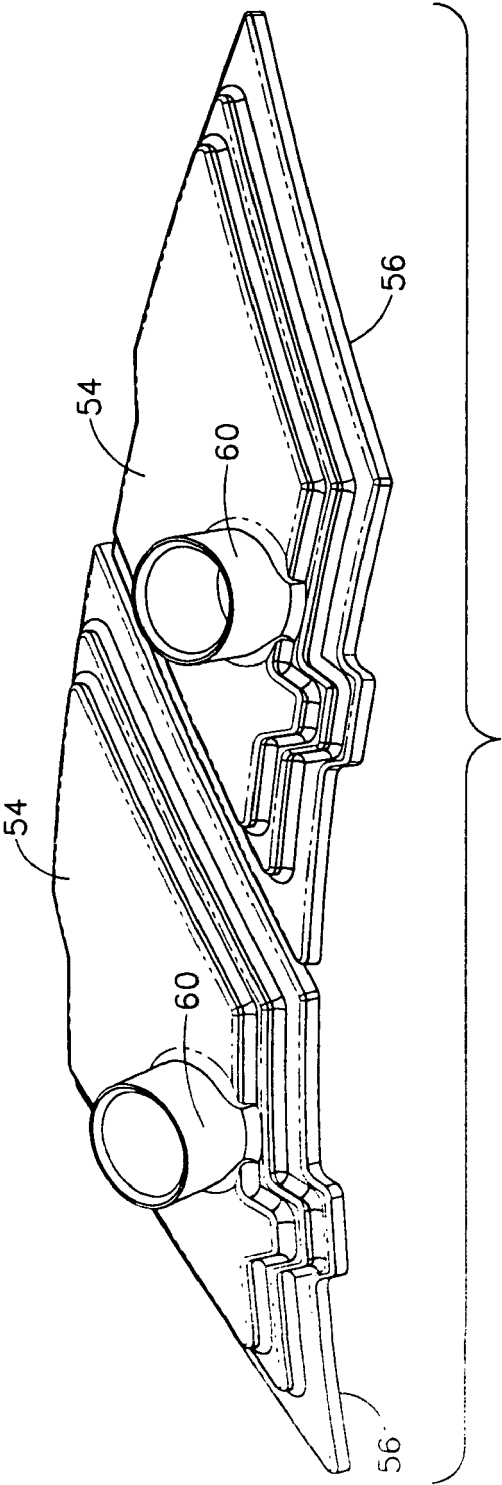


FIG. 4

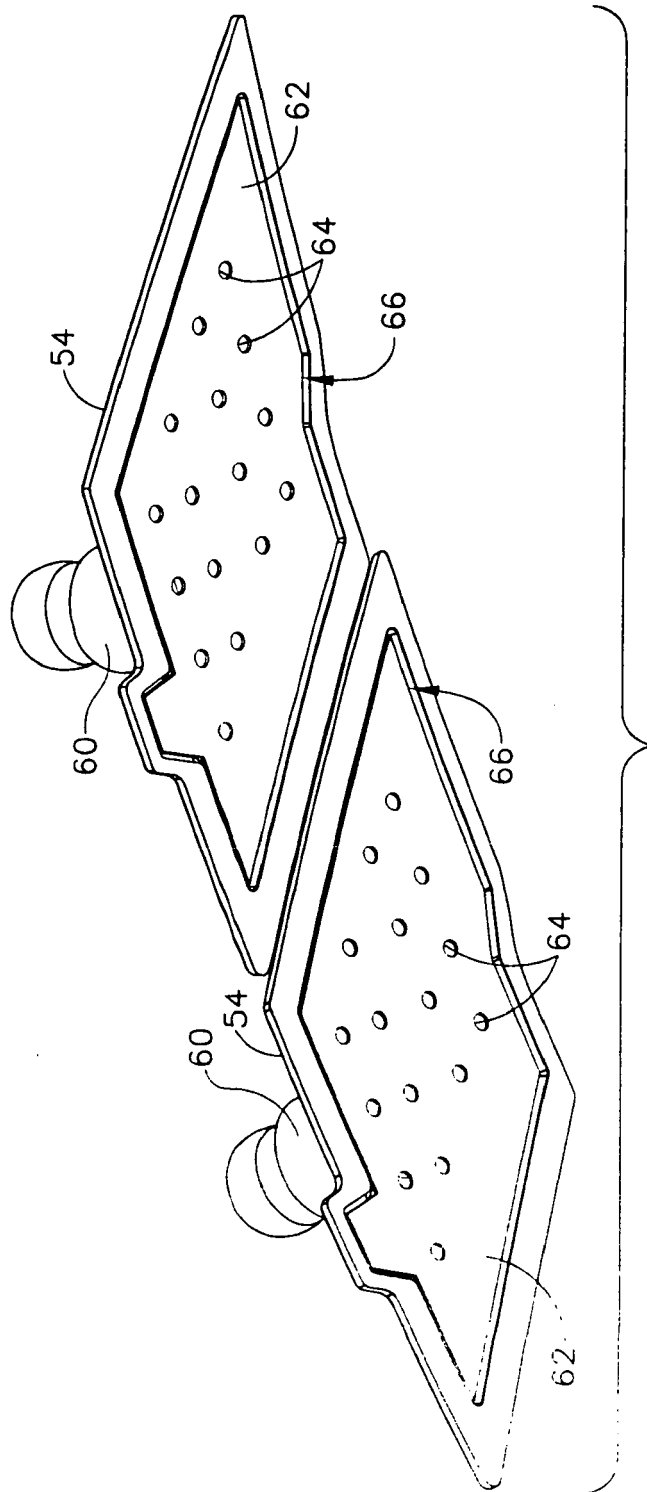


FIG. 5