# United States Patent

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[33] **Great Britain**
[31] **26169/68**


[54] **GAS TURBINE ENGINE AXIAL FLOW MULTISTAGE COMPRESSOR**
**5 Claims, 9 Drawing Figs.**


[52] **U.S. Cl.** ..... **416/198,** 416/193, 416/201, 416/210, 416/217, 416/244
[51] **Int. Cl.** ..... **F01d 5/30**
[50] **Field of Search** ..... 416/198, 201, 210, 213, 215, 217, 244



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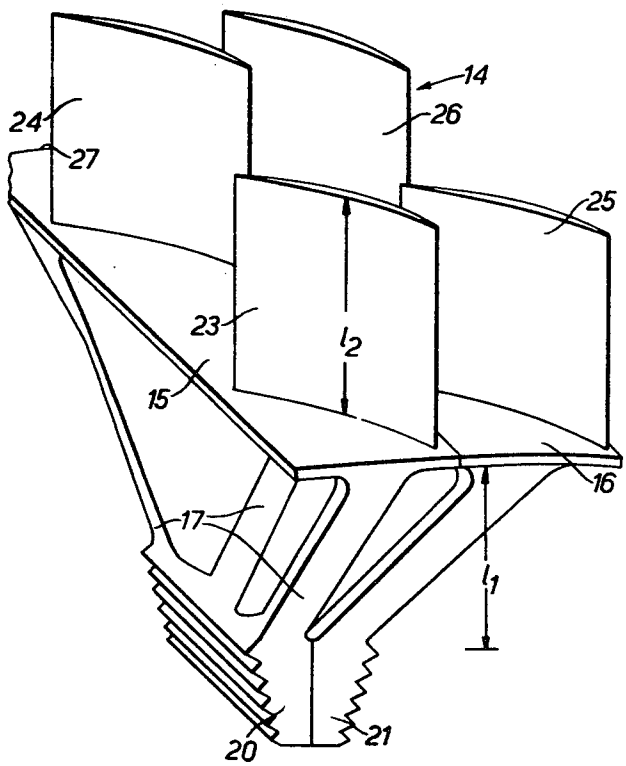
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**ABSTRACT:** The invention concerns a gas turbine engine axial flow multistage compressor having a rotor member, and a plurality of platform elements each of which has rotor blade members mounted thereon and each of which is itself disposed radially outwardly of and spaced from the rotor member, the plurality of platform elements defining an annular platform member and the axially consecutive annular platform member of different stages of rotor blade members contacting each other to form a drumlike structure.

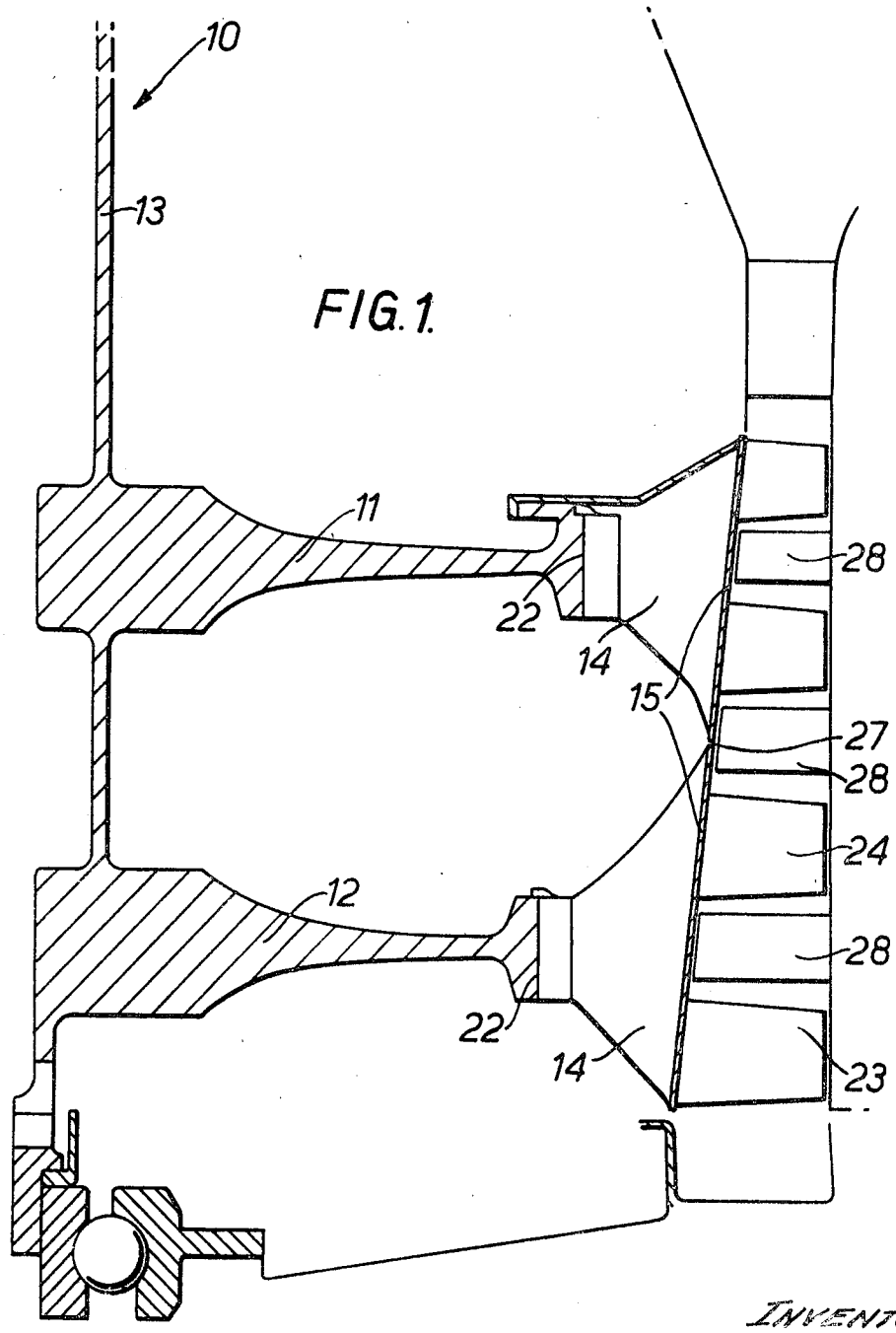
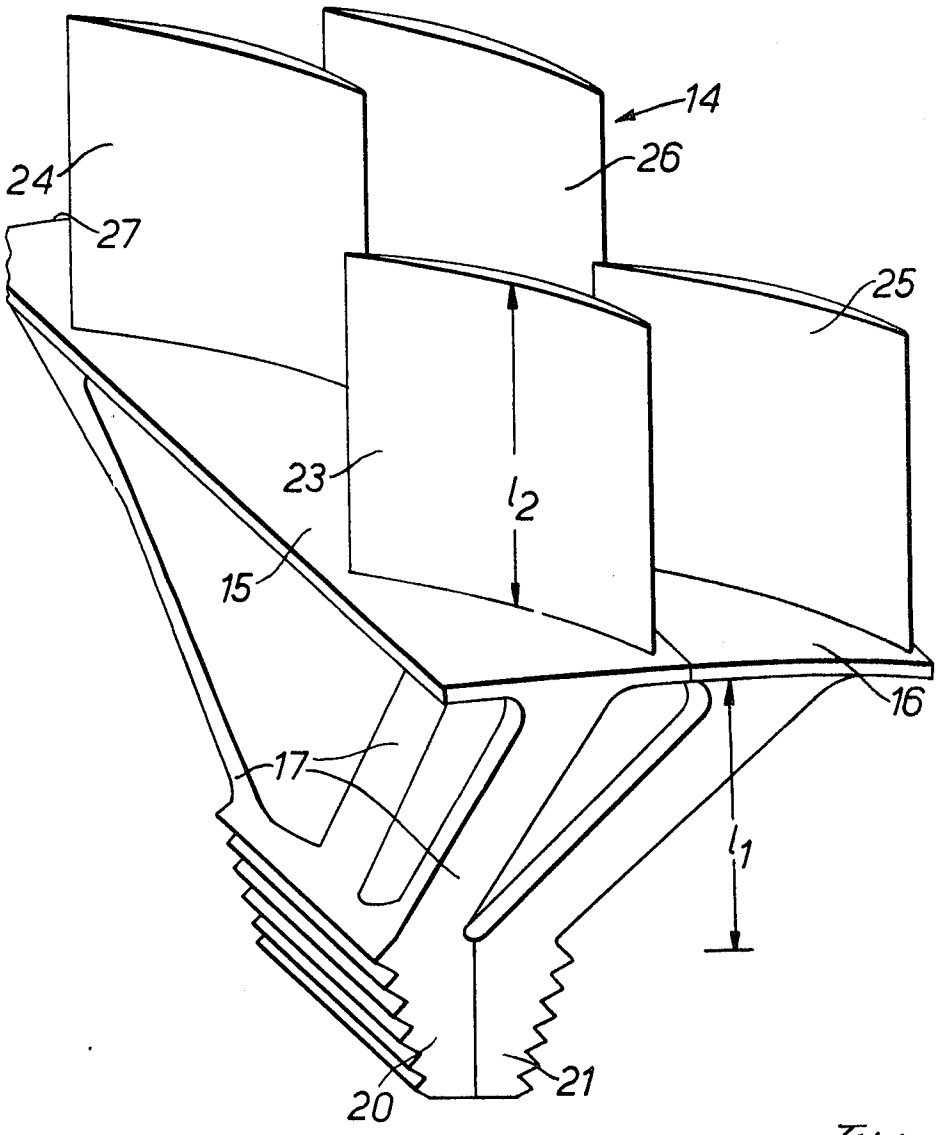



FIG. 1.

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FIG. 2.




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FIG. 3.

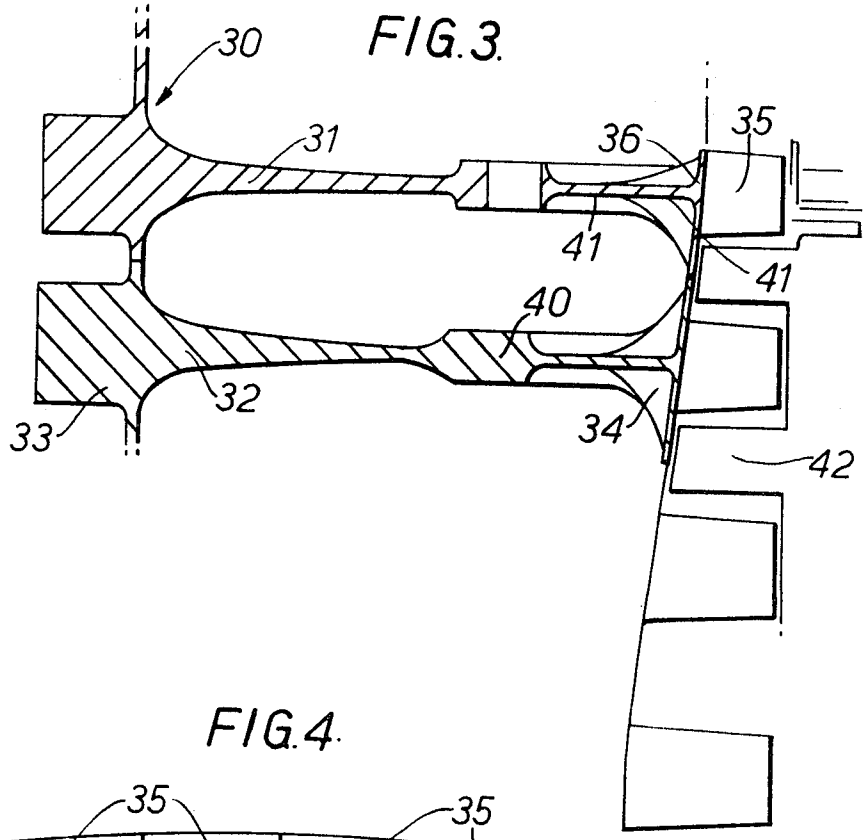
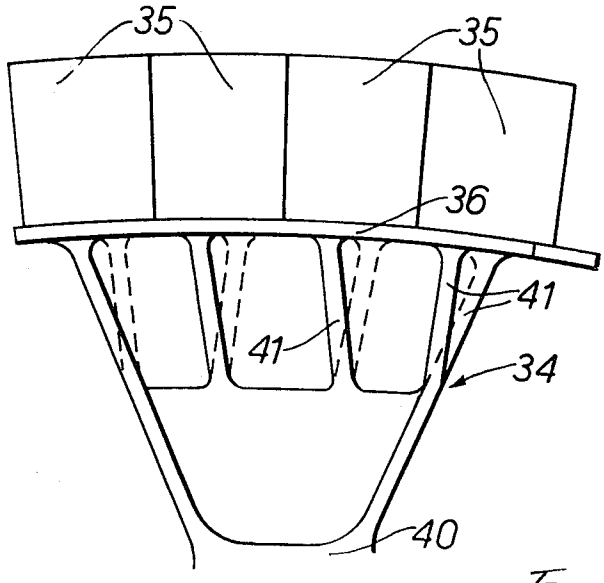



FIG. 4.




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FIG.5.

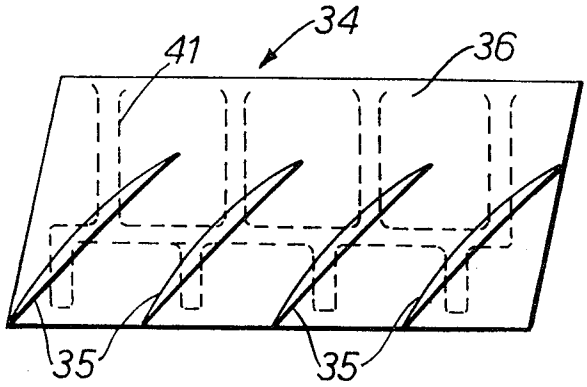



FIG.7.




FIG.8.

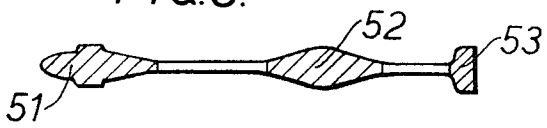
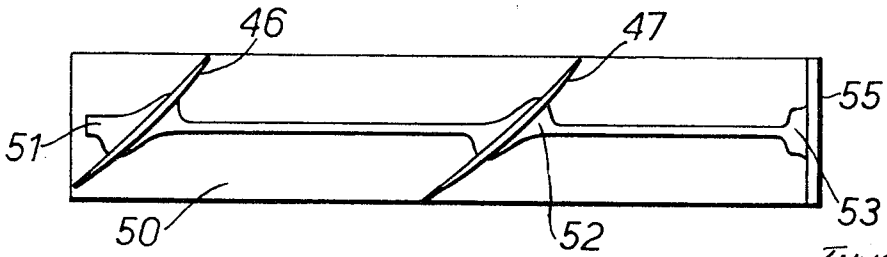
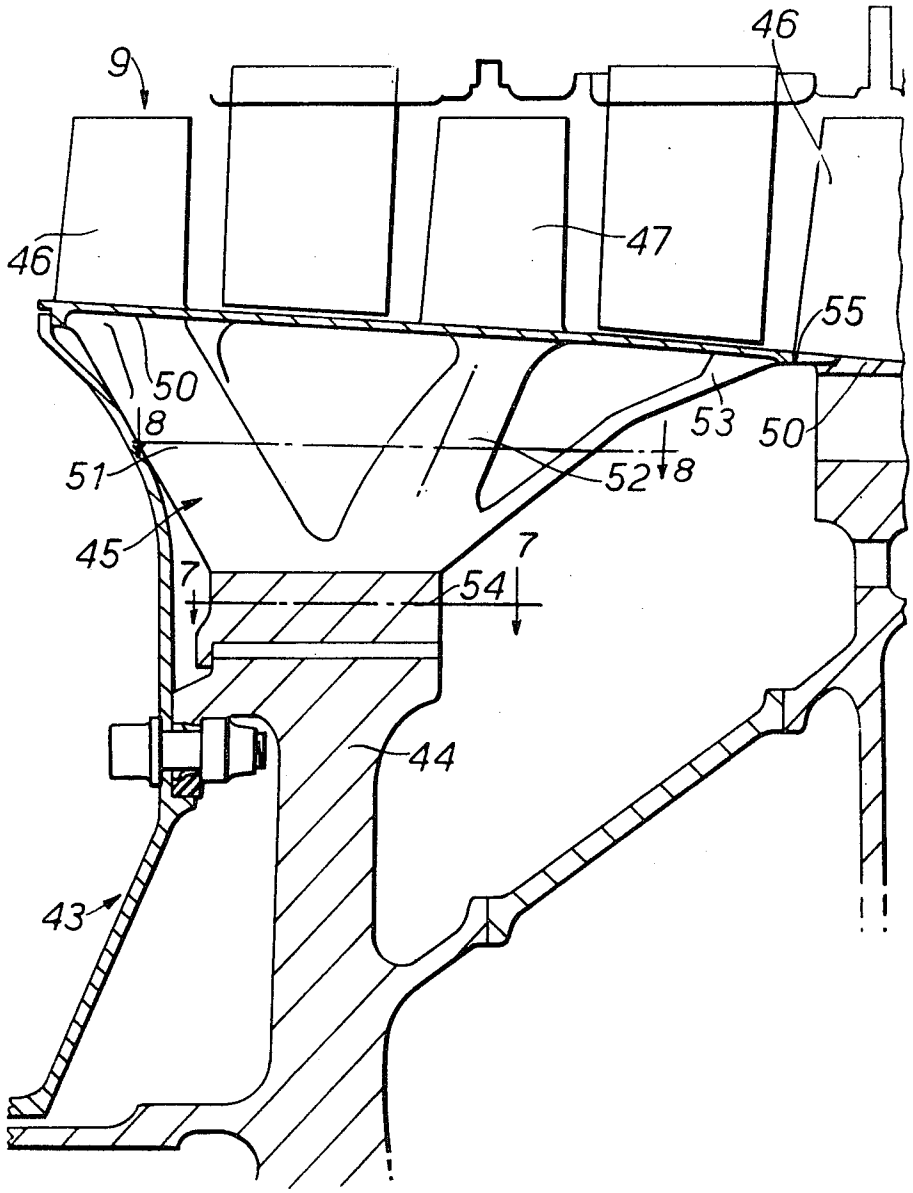



FIG.9.




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FIG.6.




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# GAS TURBINE ENGINE AXIAL FLOW MULTISTAGE COMPRESSOR

This invention concerns a gas turbine engine axial flow multistage compressor.

According to the present invention, there is provided a gas turbine engine axial flow multistage compressor having rotor blade members each of which is mounted on a platform which is itself disposed radially outwardly of and spaced from a rotor member, the axially consecutive platforms of different stages of rotor blade members contacting each other to form a drumlike structure.

Each platform may carry a plurality of rotor blade members of the same stage.

Additionally, or alternatively, each platform may carry a plurality of rotor blade members of different stages.

Each platform is preferably connected by connecting structure to a root portion which is mounted in a rotor disc, the connecting structure extending radially outwardly of the rotor disc.

The connecting structure may comprise a plurality of rib members, each platform may be integral with its blade or blade members, connecting structure and root portion.

Angularly consecutive platforms may be connected to root portions which contact each other and which are mounted in a common slot in a rotor disc.

The invention is illustrated, merely by way of example, in the accompanying drawings, in which:

FIG. 1 is a broken-away plan view of part of a gas turbine engine axial flow multistage compressor according to the present invention.

FIG. 2 is a perspective view of a blade assembly forming part of the compressor in FIG. 1,

FIG. 3 is a broken-away plan view of part of another embodiment of a gas turbine engine axial flow multistage compressor according to the present invention,

FIGS. 4 and 5 are respectively an elevation and a plan view of a blade assembly forming part of the compressor shown in FIG. 3,

FIG. 6 is a broken-away sectional view through part of yet another embodiment of a gas turbine engine axial flow multistage compressor according to the present invention,

FIGS. 7 and 8 are sections taken respectively on the lines 7–7 and 8–8 of FIG. 6, and

FIG. 9 is a plan view looking in the direction of the arrow 9 of FIG. 6.

Referring first to the embodiment shown in FIGS. 1 and 2, a gas turbine engine, axial flow, multistage, high-pressure compressor 10 comprises rotor discs 11, 12 which are carried by a shaft 13, each of the discs 11, 12 being provided with a plurality of angularly spaced-apart blade assemblies 14.

As best seen in FIG. 2, each blade assembly 14 comprises two adjacent angularly consecutive platforms 15, 16 which are mounted in contact with each other and which extend axially of the compressor. The platforms 15, 16 are respectively integral, via a plurality of rib members 17, with root portions 20, 21 which collectively form a "fir-tree"-shaped root and are mounted in a common slot 22 in the respective rotor disc 11, 12.

The platform 15 is provided with two rotor blade members 23, 24 which are mounted thereon and integral therewith while the platform 16 is similarly provided with two rotor blade members 25, 26. The rotor blade members 23, 25 constitute members of a common rotor stage, while the rotor blade members 24, 26 constitute members of a different, i.e. the adjacent, rotor stage. Thus each of the platforms 15, 16 supports rotor blade members of two adjacent rotor stages.

The rib members 17, by means of which each of the platforms 15, 16 is disposed radially outwardly of and is spaced from the rotor disc 11, 12, extend radially outwardly of the rotor discs 11, 12 throughout a radial length $l_1$ which is at least 50 percent of (and which may, indeed, be at least equal to) the radial length $l_2$ of the rotor blade members 23—26.



As will be seen from FIG. 1, each of the platforms 15, which carries two consecutive stages of rotor blade members 23, 24 contacts the axially consecutive platform 15, which carries the next two consecutive stages of rotor blade members 23, 24, the axially consecutive platforms 15 contacting each other at their adjacent edges 27 to form a drumlike structure. Similarly, of course, the axially consecutive platforms 16 contact each other at their adjacent edges.

The compressor 10 is also provided with stator blades 28.

The structure described above has a number of advantages.

In a conventional compressor, an annular flange member extends between and is sealed to adjacent rotor discs and forms with the radial extremities of the adjacent rotor discs an annular groove. Into this annular groove there extends an annular inner shroud structure of the respective stator row, this inner shroud structure having sealing members which cooperate with corresponding sealing members on the annular flange member. Since, however, in the case of the structure shown in FIGS. 1 and 2, the axially consecutive platforms 15, 16 contact each other to form a drumlike structure, the sealing losses which normally occur through the said annular groove, and the heating of the rotor discs due to air friction, are avoided, while the stator blades 28 do not need to be provided with the said inner shroud structure, so that their efficiency is improved.

The provision of the radially elongated rib members 17 also ensures that the air which is heated by being compressed in the compressor 10 is kept well away from the rotor discs 11, 12. This is of particular value in a high-pressure compressor having a high overall compression ratio since the temperature of the air at the downstream end of such a compressor would otherwise necessitate that the rotor discs should be formed of high-temperature-resistant materials which are heavy and expensive. Although in the construction shown in FIGS. 1 and 2 it is necessary to use somewhat heavier blade members and rib members than usual, it is also possible to employ lighter materials in the rotor discs than would otherwise be the case and thus to effect an overall saving in weight. The provision of the radially elongated rib members would normally increase the volume of the above-mentioned annular groove and therefore increase the sealing losses to unacceptably high values. Since, however, in the construction of FIGS. 1 and 2, the said annular groove is avoided, the said construction is also such as to permit the use of the said radially elongated rib members.

The provision of these radially elongated rib members would, moreover, normally make it difficult to find sufficient circumferential space on the rim of the rotor disc to mount each rotor blade member individually therein. However, since each root portion 20, 21 carries a plurality of rotor blade members, this space problem is solved. At the same time, since each root portion may be cast integrally with its rib members, platform and blade members, the assembly of the compressor is simplified by reason of a reduction in the number of parts requiring to be assembled together.

In the embodiment shown in FIGS. 3 to 5, a gas turbine engine axial flow, multistage compressor 30 has rotor discs 31, 32, which are carried by a shaft 33 and each of which is provided with a plurality of angularly spaced-apart blade assemblies 34. Each of the blade assemblies 34 has four rotor blade members 35 (see FIGS. 4 and 5) which are integral with and mounted on a common platform 36. The rotor blade members 35 are all members of a common rotor stage.

The platform 36 of each of the blade assemblies 34 is connected to a root portion 40 by a plurality of rib members 41. The rib members 41, by means of which each of the platforms 36 is disposed radially outwardly of and is spaced from the rotor discs 31, 32, extend radially outwardly of the latter throughout a radial length which exceeds that of the rotor blade members 35.

The compressor 30 is also provided with stator blades 42.

As will be seen from FIG. 3, the axially consecutive platforms 36 of different stages of rotor blade members 35 contact each other to form a drumlike structure.

In the embodiment illustrated in FIGS. **6** to **9**, a gas turbine engine axial flow, multistage compressor **43** (FIG. **6**) has a plurality of rotor discs **44** each of which is provided with a plurality of angularly spaced-apart blade assemblies **45**. Each of the blade assemblies **45** has two rotor blade members **46**, **47**, which form members of different rotor stages and which are integral with and mounted on a common platform **50**. The platform **50** is itself integral with rib members **51**, **52**, **53** which serve to connect the platform **50** to a root portion **54** by means of which the blade assembly is mounted in the disc **44**.

The rib members **51**, **52**, **53**, by means of which each of the platforms **50** is disposed radially outwardly of and is spaced from the rotor disc **44** extend outwardly of the latter throughout a radial length which exceeds that of the rotor blade members **46**, **47**.

As will be seen from FIG. **6**, each of the platforms **50**, which carries two consecutive stages of rotor blade members **46**, **47**, contacts the axially consecutive platform **50**, which carries the next two consecutive stages of rotor blade members **46**, **47**, the axially consecutive platforms **50** contacting each other at their adjacent edges **55** to form a drumlike structure.

We claim:

**1**. A gas turbine engine axial flow multistage compressor comprising: a plurality of axially spaced rotor discs; each of said rotor discs supporting a plurality of angularly spaced blade assemblies, the angularly spaced blade assemblies of one of said rotor discs being axially spaced with respect to the angularly spaced blade assemblies of an adjacent one of said discs; each blade assembly of said plurality of angularly spaced blade assemblies being an integral structure including a root portion mounted in a slot in a respective rotor disc, a connecting structure extending from the root portion radially outwardly of the respective rotor disc, a platform element disposed on the outer end of said connecting structure, and a plurality of rotor blade members mounted on said platform element; and said platform elements of said angularly spaced blade assemblies for the respective rotor discs defining an annular platform member and consecutive annular platform members of axially spaced-apart rotor discs contacting one another to form a drumlike structure.

**2**. A compressor as claimed in claim **1** in which each platform element carries a plurality of rotor blade members of a same stage.

**3**. A compressor as claimed in claim **1** in which each platform element carries a plurality of rotor blade members of different stages.

**4**. A compressor as claimed in claim **1** in which said connecting structure of each blade assembly comprises at least one rib member angularly spaced from a rib member of an adjacent blade assembly.

**5**. A compressor as claimed in claim **1** in which adjacent angularly spaced blade assemblies have root portions which contact each other and which are mounted in a common slot in the respective rotor disc.