

LIGHTWEIGHT FAN DRIVING TURBINE

CROSS-REFERENCE TO RELATED APPLICATION

[0001] This application claims priority to United States Provisional Application No. 61/604,572, filed February 29, 2012.

BACKGROUND OF THE INVENTION

[0002] This disclosure relates to a fan driving turbine for a gas turbine engine having a lightweight blade configuration.

[0003] A typical jet engine has multiple shafts or spools that transmit torque between turbine and compressor sections of the engine. In one example, a low speed spool generally includes a low shaft that interconnects a fan, a low pressure compressor, and a fan driving turbine. Fan driving turbine efficiency can be improved through increased speed, reduced blade/disk counts, and a reduced turbine stage count. Further, the vehicle fuel burn can be reduced because of weight reduction associated with the reduced blade/disk and stage count. The vehicle operator's savings from the reduction in fuel burn can be further enhanced by the need to buy few replacement parts as a result of the reduction in blade/disk and stage count.

SUMMARY OF THE INVENTION

[0004] In one exemplary embodiment, a gas turbine engine includes a shaft defining an axis of rotation and a fan driving turbine configured to drive the shaft. The fan driving turbine comprises a plurality of stages that are spaced apart from each other along the axis. Each stage includes a turbine disk comprised of a disk material and a plurality of turbine blades comprised of a blade material. The disk material and the blade material for at least one of the plurality of stages is selected based on a location of the one stage relative to the other stages of the plurality of stages.

[0005] In a further embodiment of the above, the disk material and blade material comprise a common material.

[0006] In a further embodiment of any of the above, the common material comprises a monolithic ceramic material.

[0007] In a further embodiment of any of the above, the disk material and blade material comprise different materials.

[0008] In a further embodiment of any of the above, at least one row of turbine blades are of a blade material comprising a ceramic matrix composite.

[0009] In a further embodiment of any of the above, at least one row of turbine blades are of a blade material comprising a forged titanium aluminide material.

[0010] In a further embodiment of any of the above, at least one row of turbine blades are of a blade material comprising a cast titanium aluminide material.

[0011] In a further embodiment of any of the above, the disk material comprises a forged powdered metal material and the blade material comprises one of a ceramic matrix composite material, a forged titanium aluminide material, or a cast titanium aluminide material.

[0012] In a further embodiment of any of the above, the disk material comprises a non-powdered metal material and the blade material comprises one of a forged titanium aluminide material or a cast titanium aluminide material.

[0013] In a further embodiment of any of the above, the blade material comprises one of a ceramic matrix composite material, a forged titanium aluminide material, or a cast titanium aluminide material.

[0014] In a further embodiment of any of the above, the disk material comprises a forged powdered metal material.

[0015] In a further embodiment of any of the above, the plurality of stages comprises at least a first turbine stage and a second turbine stage located aft of the first turbine stage, and wherein the blade material of the first turbine stage comprises a first material having a first temperature capability and the blade material of the second turbine stage comprises a second material having a second temperature capability that is lower than the first temperature capability.

[0016] In a further embodiment of any of the above, the first material comprises a ceramic matrix composite material and the second material comprises a titanium aluminide material.

[0017] In a further embodiment of any of the above, the common material is selected from the group consisting of: forged powdered metal material, forged titanium aluminide, cast titanium aluminide, and a monolithic ceramic material.

[0018] In a further embodiment of any of the above, the disk material of the first turbine stage comprises a third material having a third temperature capability and the disk material of the second turbine stage comprises a fourth material having a fourth temperature capability that is lower than the third temperature capability.

[0019] In a further embodiment of any of the above, the third material comprises a forged powdered metal material and the fourth material comprises a non-powdered metal material.

[0020] In a further embodiment of any of the above, the plurality of stages further includes at least one additional turbine stage, and wherein the blade material of the additional turbine stage is common with the blade material of the first or second turbine stage.

[0021] In a further embodiment of any of the above, the disk material of the additional turbine stage is common with the disk material of the first or second turbine stage.

[0022] In a further embodiment of any of the above, the shaft is configured to drive the fan via a speed change mechanism.

[0023] In another exemplary embodiment, a gas turbine engine includes a shaft defining an axis of rotation, a geared architecture coupling the shaft to a fan, and a fan driving turbine configured to drive the shaft. The fan driving turbine comprises a plurality of stages including at least a first stage and a second stage spaced aft of the first stage along the axis. The first stage includes a first disk comprised of a first disk material having a first disk temperature capability, and includes a first set of blades comprised of a first blade material having a first blade temperature capability. The second stage includes a second disk comprised of a second disk material having a second disk temperature capability, and includes a second set of blades comprised of a second blade material having a second blade temperature capability. The first disk temperature capability is higher than the second disk temperature capability and the first blade temperature capability is higher than the second blade temperature capability.

[0024] In a further embodiment of any of the above, the first blade material comprises a ceramic matrix composite material and the second blade material comprises a forged or cast titanium aluminide.

[0025] In a further embodiment of any of the above, the first disk material comprises a forged powdered metal material and the second disk material comprises a forged non-powdered metal material.

[0026] In a further embodiment of any of the above, the plurality of turbine stages further includes at least one additional turbine stage, and wherein a blade material of the additional stage is common with the first blade material, and wherein a disk material of the additional turbine stage is common with the first disk material.

[0027] In a further embodiment of any of the above, the plurality of turbine stages further includes at least one additional turbine stage, and wherein a blade material of the additional stage is common with the second blade material, and wherein a disk material of the additional turbine stage is common with the second disk material.

[0028] An exemplary method of assembling a gas turbine engine comprising the steps of: (a) providing a shaft defining an axis of rotation; (b) providing a fan driving turbine configured to drive the shaft, the fan driving turbine comprising a plurality of stages spaced apart from each other along the axis, wherein each stage includes: (i) a turbine disk comprised of a disk material and (ii) a plurality of turbine blades comprised of a blade material; (c) selecting the disk material and the blade material for one of the plurality of stages based on a location of the one stage relative to the other stages of the plurality of stages; and (d) assembling the stage from step (c) to the shaft at the location.

[0029] These and other features of this application will be best understood from the following specification and drawings, the following of which is a brief description.

BRIEF DESCRIPTION OF THE DRAWINGS

[0030] Figure 1 schematically illustrates a gas turbine engine.

[0031] Figure 2 is a schematic representation of an engine upper half including a fan driving turbine with a lightweight architecture.

[0032] Figure 3 is a schematic perspective representation of a portion of a turbine disk and blade assembly.

DETAILED DESCRIPTION

[0033] A gas turbine engine 20 is disclosed herein as a two-spool turbofan (shown in Figure 1) or three spool turbofan (not shown) that generally incorporates a fan section 22, a compressor section 24, a combustor section 26 and a turbine section 28. The compressor section 24 includes both a low pressure (or first) compressor section 44 and a high pressure (or second) compression section 52. The turbine section 28 may include two turbines in a two spool engine (e.g., a high pressure (or second) turbine section 54 and a low pressure (or

first) turbine section 46, as shown) or three turbines in a three spool engine. Both two and three spool architectures include a fan driving turbine as the last turbine in the turbine section, i.e., the low pressure turbine section 46 in Figure 1. Alternative engines might include an augmentor section (not shown) among other systems or features. The fan section 22 drives air along a bypass flowpath B while the compressor section 24 drives air along a core flowpath C for compression and communication into the combustor section 26 then expansion through the turbine section 28. Although depicted as a turbofan gas turbine engine in the disclosed non-limiting embodiment, it should be understood that the concepts described herein are not limited to use with turbofans as the teachings may be applied to other types of turbine engines including three or more-spool architectures. As used herein, a “high pressure” compressor or turbine experiences a higher pressure than a corresponding “low pressure” compressor or turbine.

[0034] The engine 20 generally includes a lower speed section 30 and a higher speed section 32 with separate spool or spools mounted for rotation about an engine central longitudinal axis A relative to an engine static structure 36 via several bearing systems 38. It should be understood that various bearing systems 38 at various locations may alternatively or additionally be provided.

[0035] The fan drive turbine 46 generally includes an inner shaft 40 that interconnects a fan 42. Additionally, in some engine embodiments, e.g., the embodiment shown in Figure 1, the fan drive turbine 46 will also drive a low pressure compressor 44. The inner shaft 40 of the fan drive turbine 46 in the embodiment of Figure 1 is connected to the fan 42 through a speed change mechanism, such as a geared architecture 48 for example, to drive the fan 42 at a lower speed than the fan drive turbine 46 and the low pressure compressor section 44. The high speed section 32 includes an outer shaft 50 that interconnects the high pressure compressor section 52 and high pressure turbine section 54. Turbines other than the fan driving turbines can be included to turn other parts of the compressor section. A combustor 56 provided in the combustion section 26 is arranged between the high pressure compressor section 52 and the high pressure turbine section 54. A mid-turbine frame 57 of the engine static structure 36 is arranged generally between the high pressure turbine section 54 and the fan driving turbine 46. The mid-turbine frame 57 supports one or more bearing systems 38 in the turbine section 28. The fan drive turbine shaft 40 is concentrically inside the other shaft or shafts 50; all of the shafts 40, 50 rotate via

bearing systems 38 about the engine central longitudinal axis A, which is collinear with their longitudinal axes.

[0036] The core airflow C is compressed by the low and high pressure compressor sections 44, 52 of the compressor section 24, mixed and burned with fuel in the combustor 56, then expanded over the high pressure turbine section 54 and the fan driving turbine 46. The mid-turbine frame 57 includes airfoils 59 which are in the core airflow path C to optimally present the gaspath flow from the high pressure turbine section 54 to the first blades of the fan drive turbine section 46. The turbines 46, 54 rotationally drive the respective low speed spool 30 and higher speed spools 32 in response to the expansion.

[0037] The engine 20 in one example is a high-bypass geared aircraft engine. In a further example, the engine 20 bypass ratio is greater than about six (6), with an example embodiment being greater than ten (10), the geared architecture 48 is an epicyclic gear train, such as a planetary gear system or other gear system, with a gear reduction ratio of greater than about 2.3 and the fan driving turbine 46 has a pressure ratio that is greater than about 5. In one disclosed embodiment, the engine 20 bypass ratio is greater than about ten (10:1), the fan diameter is significantly larger than that of the low pressure compressor 44, and the fan driving turbine 46 has a pressure ratio that is greater than about 5:1. Fan driving turbine 46 pressure ratio is pressure measured prior to inlet of fan driving turbine 46 as related to the pressure at the outlet of the fan driving turbine 46 prior to an exhaust nozzle. The geared architecture 48 may be an epicycle gear train, such as a planetary gear system or star or other gear system, with a gear reduction ratio of greater than about 2.5:1. It should be understood, however, that the above parameters are only exemplary of one embodiment of a geared architecture engine and that the present invention is applicable to other gas turbine engines including direct drive turbofans.

[0038] A significant amount of thrust is provided by the bypass flow B due to the high bypass ratio. The fan section 22 of the engine 20 is designed centering on a particular flight condition -- typically cruise at about 0.8 Mach and about 35,000 feet. The flight condition of 0.8 Mach and 35,000 ft, with the engine at its best fuel consumption - also known as "bucket cruise Thrust Specific Fuel Consumption ('TSFC')"- is the industry standard parameter of lbf of fuel being burned per hour divided by lbf of thrust the engine produces at that minimum point. "Low fan pressure ratio" is the pressure ratio across the fan blade alone, without a fan Exit Guide Vane ("FEGV") system. The low fan pressure ratio as

disclosed herein according to one non-limiting embodiment is less than about 1.45. "Low corrected fan tip speed" is the actual fan tip speed in ft/sec divided by an industry standard temperature correction of $[(T_{\text{ambient deg R}}) / 518.7]^{0.5}$. The "Low corrected fan tip speed" as disclosed herein according to one non-limiting embodiment is less than about 1150 ft / second.

[0039] A gas turbine engine 60 shown in Figure 2 includes a two-spool turbofan as described above, which generally incorporates a fan section 22, a compressor section 24, a combustor section 26 and a turbine section 28. In this configuration, the fan driving turbine 46 is comprised of a lightweight architecture that allows a significant increase of turbine speed. The fan driving turbine 46 is comprised of a plurality of stages. In the example shown, the fan driving turbine includes a first stage 62, a second stage 64, and a third stage 66. It should be understood that while three stages are shown, a two stage configuration could also be utilized depending upon the application.

[0040] Each stage 62, 64, 66 includes a turbine disk comprised of a disk material. Each disk is associated with a plurality of turbine blades comprised of a blade material. In the example shown, the first stage 62 includes a first disk 70 which is associated with a first plurality of blades 72 as shown in Figure 3. The second 64 and third 66 stages have a similar configuration to that of Figure 3. The second stage includes a second disk 74 associated with a second plurality of blades 76 and the third stage 66 includes a third disk 78 associated with a third plurality of blades 80.

[0041] As shown in Figure 3, the disk 70 includes a plurality of slots 82 that receive roots 84 of the associated blades 72. An outer shroud 86 is associated with distal ends of the blades 72. It should be understood that the disk/blade configuration shown in Figure 3 is merely one example, and that other configurations could also be used.

[0042] As discussed above, each of the disks 70, 74, 78 is comprised of a disk material and each of the blades 72, 76, 80 is comprised of a blade material. The disk material and the blade material for an associated stage 62, 64, 66 are each selected based on a location of that stage relative to the other stages. In other words, the materials used in each stage are selected based on a location of that stage within the core flow C of the engine. This is because upstream stages are subjected to higher temperatures within the core flow C due to their proximity to the combustor section 26.

[0043] In order to advantageously increase the speed capability of the fan driving turbine 46, to make the turbine smaller in diameter, to reduce the number of stages and the number of airfoils, and to make the entire turbine shorter and lighter, the upstream stages, i.e. the stages closest to the combustor section 26, may be comprised of a lightweight high temperature material for both the associated disk and blades. For example, the first stage 62, which is the most upstream stage of the three stages 62, 64, 66, has the disk 70 comprised of a powdered metal material to withstand the temperature in the engine and blades 72 comprised of a ceramic matrix composite material. In one example, the disk 70 comprises a powdered metal forging and the blades 72 are cast of a ceramic material and machined or formed from a ceramic matrix composite. Any type of powdered metal material and any type of ceramic matrix composite material can be used.

[0044] In another example, both the disk 70 and blades 72 are formed of a monolithic ceramic material and the entire rotor section is one piece. In this example, the disk 70 and blades 72 comprise a single-piece, monolithic cast ceramic structure.

[0045] In one example, the second stage 64, which is positioned downstream of the first stage 62 and upstream of the third stage 66, includes a second disk 74 and second blades 76 made from similar materials to that of the first stage 62. For example, the second disk 74 is a powdered metal forging and the second blades 76 are comprised of a cast ceramic matrix composite material. Optionally, the disk 74 and blades 76 could be a monolithic structure comprised of a cast ceramic material.

[0046] In one example, the third stage 66, which is downstream of the second stage 64, has a third disk 78 and third blades 80 made from a material different than that of the first stage 62. For example, the third blades 80 are comprised of a forged titanium aluminide material and the third disk 78 is comprised of a non-powdered metal material. For example, the third disk 78 could be made from a conventional disk material.

[0047] In another example, a cast titanium aluminide blade is used for reduced cost, if the rotor speed is lower or the gas path temperatures are lower so as to allow the use of such a lower strength titanium aluminum alloy.

[0048] In one example, the third stage 66 is made from a material that has a lower temperature capability as compared to materials used in the first 62 and second 64 stages. This is because the third stage 66 is positioned at a cooler location within the core flow C relative to the first 62 and second 64 stages, which are positioned closer to the combustor

section 26. Thus, the blade material of the more upstream stages comprises a material having a higher temperature capability than the blade material of downstream stages. Using a titanium aluminide material for the third blades and a conventional material for the third disk is less expensive than using the high temperature capability materials of the first and second stages.

[0049] Further, in another example, the second stage 64 could include blade and disk materials that are common with the third stage 66. This could be used in a configuration where the second stage 64 is located in a sufficiently lower temperature region as compared to the first stage 62. Also, if all three stages are in a high temperature region, all three stages may have disk and blade materials common with those of the first stage described above. Additional stages beyond the three stages could use either the higher temperature ceramic matrix composite materials or the cast or forged titanium aluminide materials for the blades.

[0050] As known, the discs 70, 74, 78 and associated blades 72, 76, 80 are configured to drive the low shaft 40, which in turn may also drive the lower pressure compressor 44. The low shaft 40 also drives the fan 42 via the geared architecture 48.

[0051] The configuration shown in Figure 2 provides for improved efficiency in the fan driving turbine 46 as the lightweight rotors made from lightweight materials are capable of turning at very high speeds by reducing the pull on the blade neck and on the disk rotor. Further, using higher speeds in combination with smaller turbine annular passage and high speed reduction devices, these lightweight materials allow the number of stages and the number of blades to be reduced. This facilitates a further speed increase due to the reduction in weight by eliminating stages. For example, a fan driving turbine 46 with only three stages 62, 64, 66, such as that shown in Figure 3, can be more efficiently operated than a lower pressure turbine 46 with four or more stages, such as the example shown in Figure 1 because there is less handling of the fluid through multiple blade rows and there is less leakage through the blade and vane seals.

[0052] Optionally, the fan driving turbine may only require two stages. In this example, the upstream stage would be comprised of materials similar to that of the first stage 62 discussed above, and the downstream stage may be comprised of materials similar to that of the third stage 66. However, due to proximity to the combustor, both stages in a two-stage configuration could be comprised of materials similar to those of the first stage 62 described above.

[0053] An exemplary method of assembling a gas turbine engine comprises selecting the disk material and the blade material for one of the plurality of stages based on a location of that stage relative to the other stages of the plurality of stages. That stage is then assembled to the low shaft 40 at the location. For example, for a first stage 62, which is located at the most upstream location relative to the other stages 64, 66, higher temperature capability materials are selected for the disk and blades. The disk and blade material for the next stage is also selected based on the location of that stage relative to the other stages. For example, for a second stage 64 or a third stage 66, which are located downstream of the first stage 62, lower temperature capability materials may be selected for the disks and blades.

[0054] The fan driving turbine can be comprised of any number of turbine stages. Depending upon the location of these stages, they may or may not require the higher temperature capability materials. Generally, the stages closer to the combustion section will require expensive, higher temperature capable materials while the downstream stages will not require these more expensive materials. However, any combination of high temperature stages and lower temperature stages can be utilized depending upon the engine application.

[0055] It should be understood that the temperature capability of the different materials between stages is a relative measure. In other words, all of the stages could be understood to be formed from materials having high temperature capability. However, in the disclosed examples, the stage at the most upstream location is comprised of a material that has a higher temperature capability than a downstream stage.

[0056] The disclosed fan driving turbine provides improved efficiency by providing a configuration that can handle increased speed. Further, as discussed above, the configuration reduces blade and stage count. In addition to the turbomachinery-based benefit of speed, the reduction in blade and stage count reduces cumulative boundary layer flow and tip seal leakage. The speed change mechanism of the geared architecture reduces the otherwise very high speed of the fan driving turbine. The speed change mechanism can comprise an epicyclic gearbox and may comprise a star or planet type, for example. The compactness of the back end of this configuration also benefits nacelle lines and nacelle-to-wing blockage and nacelle aerodynamics.

[0057] Although an embodiment of this invention has been disclosed, a worker of ordinary skill in this art would recognize that certain modifications would come within the

scope of this invention. For that reason, the following claims should be studied to determine the true scope and content of this invention.

CLAIMS

1. A gas turbine engine comprising:
 - at least a first shaft and a second shaft rotating at a higher speed than the first shaft, wherein the first and second shafts rotate about an engine axis of rotation;
 - a geared architecture coupling the first shaft to a fan;
 - at least a first driving turbine configured to drive the first shaft and a second driving turbine to drive the second shaft, wherein the first driving turbine is positioned downstream of the second driving turbine;
 - a combustor positioned immediately upstream of the second driving turbine; and
 - wherein the first driving turbine comprises a plurality of stages including at least a first turbine stage and a second turbine stage spaced immediately aft of the first turbine stage along the axis, and wherein each of the first and second turbine stages are comprised of:
 - a turbine disk comprised of a disk material; and
 - a plurality of turbine blades comprised of a blade material,
 - wherein the disk material and the blade material for the first and second stages are selected independently of each other based on operational temperatures at each stage's respective location along the shaft of the first and second turbine stages.
2. The gas turbine engine according to claim 1, wherein the disk material and blade material comprise different materials.
3. The gas turbine engine according to claim 2, wherein at least one row of turbine blades are of a blade material comprising a ceramic matrix composite.
4. The gas turbine engine according to claim 2, wherein at least one row of turbine blades are of a blade material comprising a forged titanium aluminide material.
5. The gas turbine engine according to claim 2, wherein at least one row of turbine blades are of a blade material comprising a cast titanium aluminide blades.
6. The gas turbine engine according to claim 2, wherein the disk material comprises a forged powdered metal material and the blade material comprises one of a ceramic matrix composite material, a forged titanium aluminide material, or a cast titanium aluminide material.

7. The gas turbine engine according to claim 2, wherein the disk material comprises a non-powdered metal material and the blade material comprises one of a forged titanium aluminide material or a cast titanium aluminide material.
8. The gas turbine engine according to claim 1, wherein the blade material comprises one of a ceramic matrix composite material, a forged titanium aluminide material, or a cast titanium aluminide material.
9. The gas turbine engine according to claim 1, wherein the geared architecture has a gear reduction ratio of about 2.3 or greater.
10. The gas turbine engine according to claim 1, including a compressor section upstream of the combustor, the compressor section driving air along a core flowpath, and wherein the fan drives air along a bypass flowpath, and wherein the gas turbine engine has a bypass ratio of about six or greater.
11. The gas turbine engine according to claim 1, wherein the turbine disk includes a plurality of slots to receive the turbine blades, and wherein distal ends of each of the turbine blades is associated with a shroud.
12. A gas turbine engine comprising:
 - at least a first shaft and a second shaft rotating at a higher speed than the first shaft, wherein the first and second shafts rotate about an engine axis of rotation;
 - a geared architecture coupling the first shaft to a fan;
 - at least a first driving turbine configured to drive the first shaft and a second driving turbine to drive the second shaft, wherein the first driving turbine is positioned downstream of the second driving turbine;
 - a combustor positioned upstream of the second driving turbine; and
 - wherein the first driving turbine comprises a plurality of turbine stages including at least a first turbine stage and a second turbine stage spaced immediately aft of the first turbine stage along the axis,

wherein the first turbine stage includes a first disk comprised of a first disk material having a first disk temperature capability and a first set of blades comprised of a first blade material having a first blade temperature capability,

wherein the second turbine stage includes a second disk comprised of a second disk material having a second disk temperature capability and a second set of blades comprised of a second blade material having a second blade temperature capability, and

wherein the first disk temperature capability is higher than the second disk temperature capability and the first blade temperature capability is higher than the second blade temperature capability.

13. The gas turbine engine according to claim 12, wherein the geared architecture has a gear reduction ratio of about 2.3 or greater.

14. The gas turbine engine according to claim 12, wherein the second driving turbine comprises a high pressure turbine including at least two stages, and wherein the first driving turbine comprises a low pressure turbine with the plurality of turbine stages comprising at least three stages.

15. The gas turbine engine according to claim 12, including a compressor section upstream of the combustor, the compressor section driving air along a core flowpath, and wherein the fan drives air along a bypass flowpath, and wherein the gas turbine engine has a bypass ratio of about six or greater.

16. The gas turbine engine according to claim 12, wherein the first disk includes a first plurality of slots to receive the first set of blades and the second disk includes a second plurality of slots to receive the second set of blades, and wherein distal ends of each of the first and second sets of blades is associated with a shroud.

17. The gas turbine engine according to claim 12, wherein the first disk material comprises a powdered metal material and the second disk material comprises a non-powdered metal material.

18. The gas turbine engine according to claim 12, wherein the first blade material comprises a ceramic material and the second blade material comprises a titanium aluminide material.

19. A gas turbine engine comprising:

at least a first shaft and a second shaft rotating at a higher speed than the first shaft, wherein the first and second shafts rotate about an engine axis of rotation;

a geared architecture coupling the first shaft to a fan;

at least a first driving turbine configured to drive the first shaft and a second driving turbine to drive the second shaft, wherein the first driving turbine is positioned downstream of the second driving turbine;

a combustor positioned upstream of the second driving turbine; and

wherein the first driving turbine comprises a plurality of turbine stages including at least a first turbine stage, a second turbine stage spaced immediately aft of the first turbine stage along the axis, and a third turbine stage spaced immediately aft of the second turbine stage along the axis,

wherein the first turbine stage includes a first disk comprised of a first disk material having a first disk temperature capability and a first set of blades comprised of a first blade material having a first blade temperature capability,

wherein the second turbine stage includes a second disk comprised of a second disk material having a second disk temperature capability and a second set of blades comprised of a second blade material having a second blade temperature capability,

wherein the third turbine stage includes a third disk comprised of a third disk material having a third disk temperature capability and a third set of blades comprised of a third blade material having a third blade temperature capability,

and

wherein the first and second disk temperature capability is higher than the third disk temperature capability and the first and second blade temperature capability is higher than the third blade temperature capability.

20. The gas turbine engine according to claim 19, wherein the first and second disk materials comprise the same material and the third disk material comprises a different material from the first and second disk materials.

21. The gas turbine engine according to claim 19, wherein the geared architecture has a gear reduction ratio of about 2.3 or greater.

22. The gas turbine engine according to claim 19, including a compressor section upstream of the combustor, the compressor section driving air along a core flowpath, and wherein the fan drives air along a bypass flowpath, and wherein the gas turbine engine has a bypass ratio of about six or greater.

23. The gas turbine engine according to claim 19, wherein the first disk and second disk material comprises a powdered metal material and the third disk material comprises a non-powdered metal material.

24. The gas turbine engine according to claim 23, wherein the first blade material and second blade material comprises a ceramic material and the third blade material comprises a titanium aluminide material.