CLAIMS

1. A gas turbine engine comprising:

a shaft defining an axis of rotation; and

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, each stage comprising:

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 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.

2. The gas turbine engine according to claim 1, wherein the disk material and blade material comprise a common material.

3. The gas turbine engine according to claim 2, wherein the common material comprises a monolithic ceramic material.

4. The gas turbine engine according to claim 1, wherein the disk material and blade material comprise different materials.

5. The gas turbine engine according to claim 4, wherein at least one row of turbine blades are of a blade material comprising a ceramic matrix composite.

6. The gas turbine engine according to claim 4, wherein at least one row of turbine blades are of a blade material comprising a forged titanium aluminide material.

7. The gas turbine engine according to claim 4, wherein at least one row of turbine blades are of a blade material comprising a cast titanium aluminide material.

8. The gas turbine engine according to claim 4, 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.

9. The gas turbine engine according to claim 4, 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.

10. 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.

11. The gas turbine engine according to claim 1, wherein the disk material comprises a forged powered metal material.

12. The gas turbine engine according to claim 1, wherein 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.

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

14. The gas turbine engine according to claim 2, wherein 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.

15. The gas turbine engine according to claim 12, wherein 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.

16. The gas turbine engine according to claim 15, wherein the third material comprises a forged powdered metal material and the fourth material comprises a non-powdered metal material.

17. The gas turbine engine according to claim 12, wherein 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.

18. The gas turbine engine according to claim 17, wherein the disk material of the additional turbine stage is common with the disk material of the first or second turbine stage.

19. The gas turbine engine according to claim 1, wherein the shaft is configured to drive the fan via a speed change mechanism.

20. A gas turbine engine comprising:

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 comprising a plurality of turbine stages including at least a first turbine stage and a second turbine stage spaced 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.

21. The gas turbine engine according to claim 20, wherein the first blade material comprises a ceramic matrix composite material and the second blade material comprises a forged or cast titanium aluminide.

22. The gas turbine engine according to claim 20, wherein the first disk material comprises a forged powdered metal material and the second disk material comprises a forged non-powdered metal material.

23. The gas turbine engine according to claim 20, wherein 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.

24. The gas turbine engine according to claim 20, wherein 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.

25. A 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.

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.