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(54) BLADE LEADING EDGE TIP RIB

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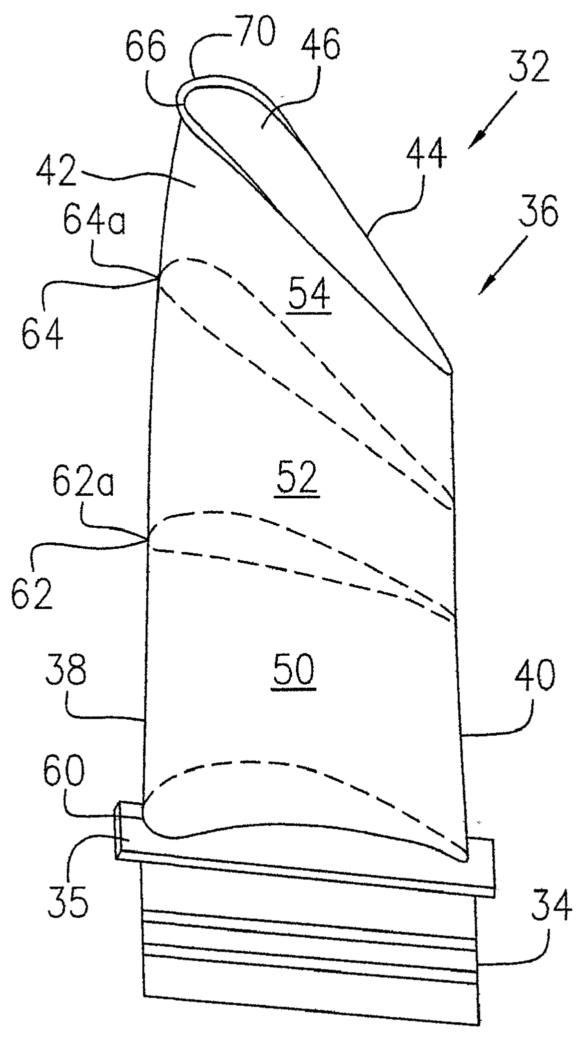
(57) ABSTRACT

A rotor blade for a gas turbine engine includes a leading edge tip rib projecting outwardly from an airfoil of the blade at a tip region thereof. The tip rib continuously surrounds a leading edge of the airfoil and extends rearwardly from the leading edge along respective pressure and suction side surfaces to thereby alter the blade tip leakage vortex structure and strength, resulting in a stage efficiency benefit.



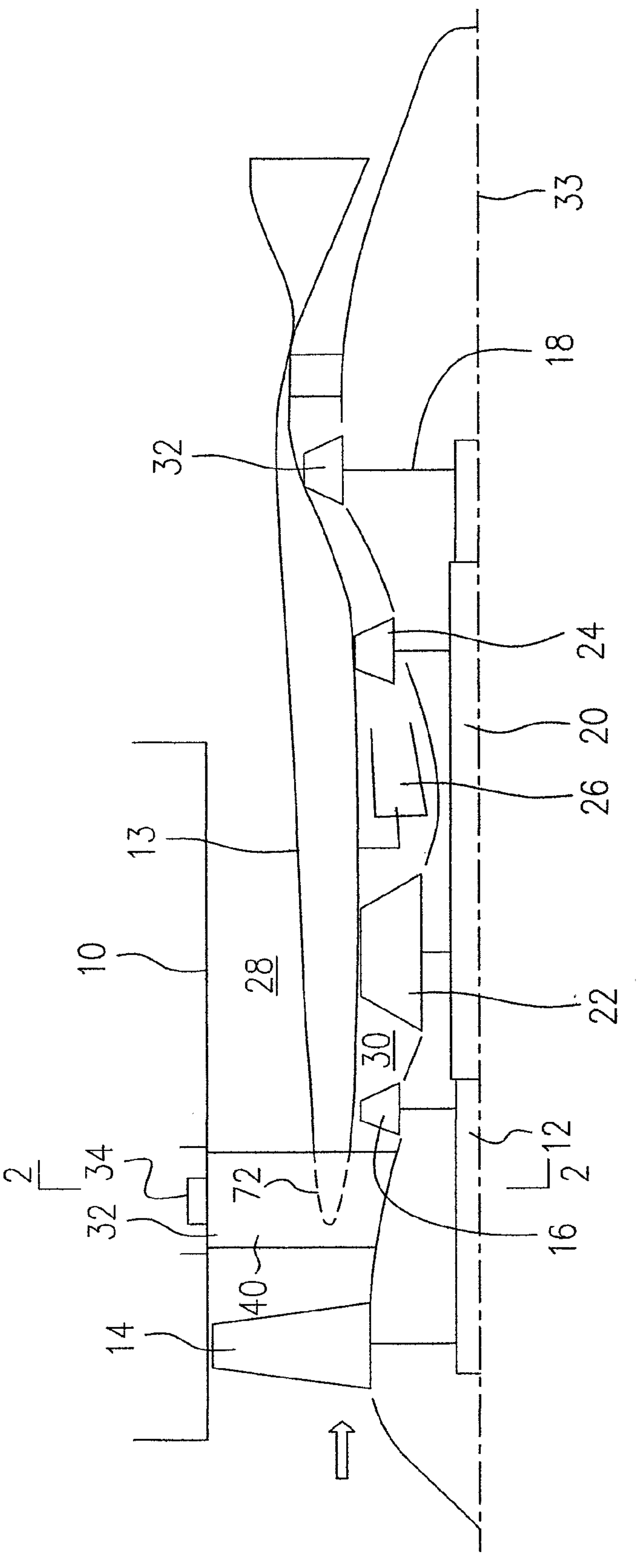


FIG. 1

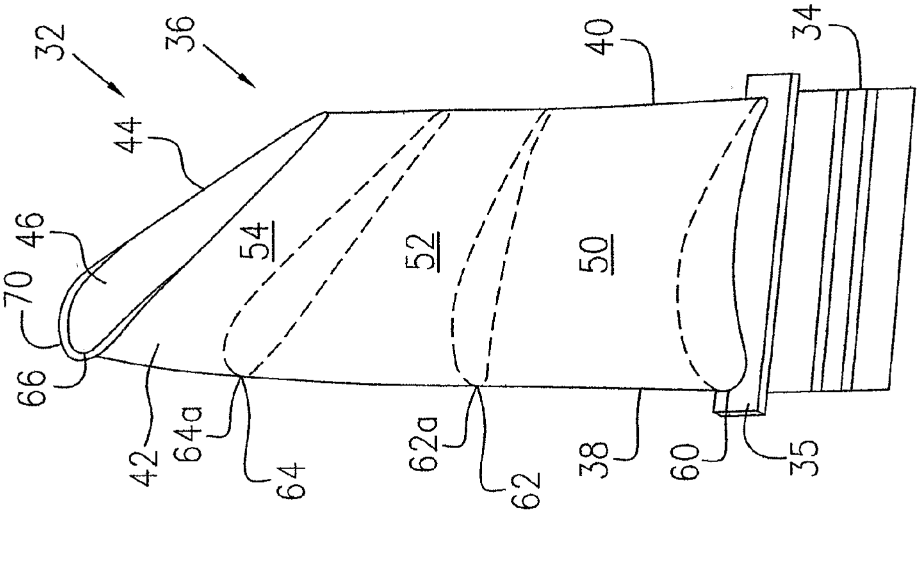

32
36
44
46
70
66
42
40
54
52
50
64a
64
62a
62
38
60
35
34

FIG. 2

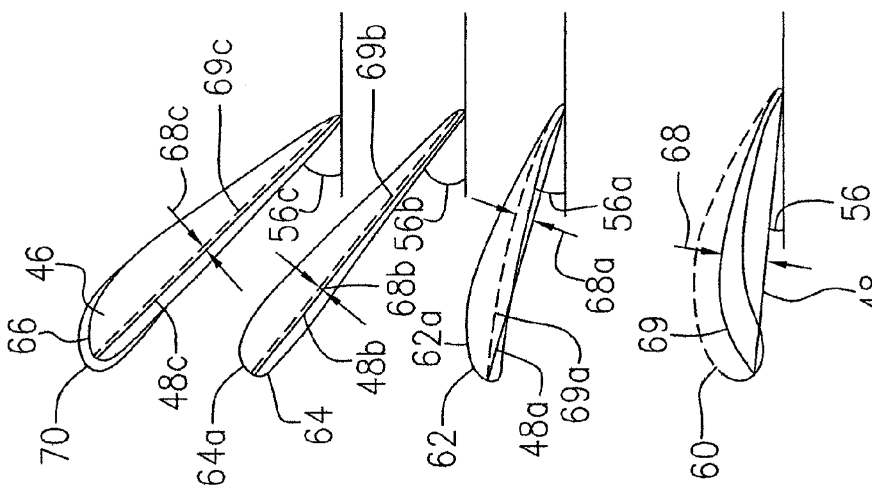

66
46
68c
69c
69b
70
48c
56c
56b
64a
64
68b
48b
62a
62
48a
69a
56a
68a
68
69
56
48
60

FIG. 3

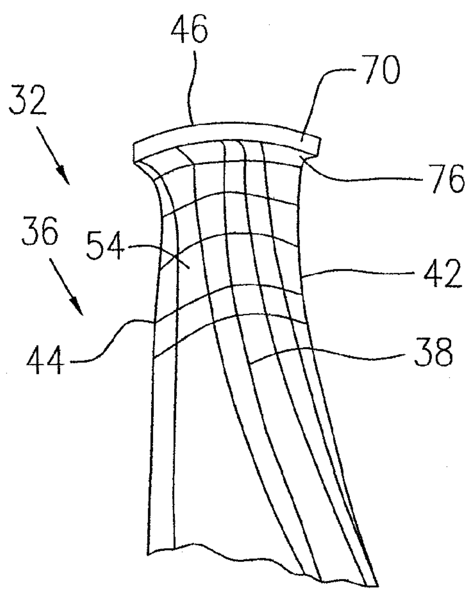


FIG. 4

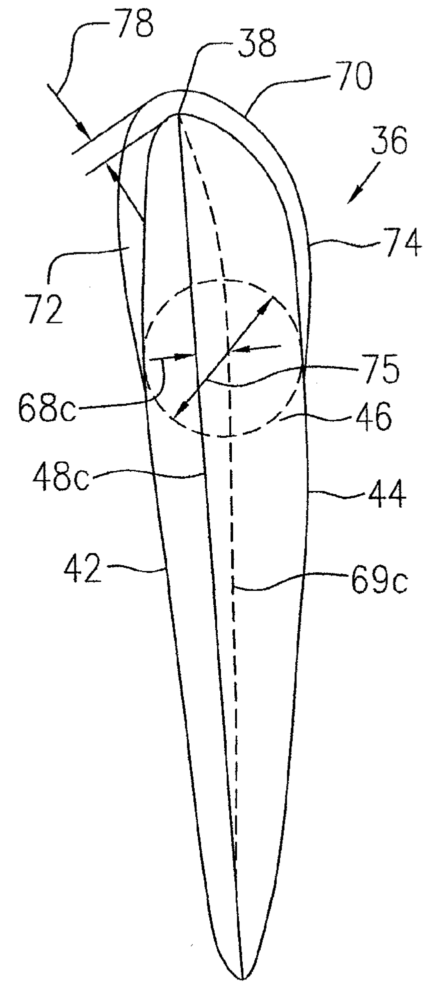


FIG. 5

BLADE LEADING EDGE TIP RIB

TECHNICAL FIELD

[0001] The application relates generally to gas turbine engines and more particularly, to a rotor blade for such engines.

BACKGROUND OF THE ART

[0002] A rotor blade for a gas turbine engine typically includes an attachment and an airfoil. The airfoil extends between the attachment and a tip and has a concave pressure side surface, a convex suction side surface, a leading edge and a trailing edge. The airfoil is sized such that when it is configured within the engine, a clearance gap is defined between the blade tip and the surrounding static structure. Blade tip clearance for an unshrouded turbine blade, and the resultant air flow that flows from the pressure side surface to the suction side surface through this space, is a significant contributor to energy loss in an axial flow turbine stage. This "leakage" airflow mixes with the suction side airflow to form a vortex. The vortex disburses, causing relatively significant flow disturbances along most of the suction side surface. The cumulative result of these flow disturbances is a reduction in engine efficiency. Therefore, it is a challenging task to disrupt, block and/or reduce the leakage flow in order to mitigate the resultant damaging effects.

[0003] Accordingly, there is a need for an improved rotor blade configuration to reduce tip gap flow leakage.

SUMMARY

[0004] In one aspect, there is provided a rotor blade for a gas turbine engine, comprising: an attachment; an airfoil defining a pressure side surface and a suction side surface, extending between the attachment and a tip, a base region disposed adjacent to the attachment, a tip region, and a transition region located between the base region and the tip region, the tip defining a chord line, and the pressure side surface adjacent to the tip being substantially planar; and wherein the tip region includes a tip rib projecting outwardly from the airfoil, the tip rib continuously surrounding a leading edge of the airfoil, extending from the leading edge rearwardly along the respective pressure side surface and suction side surface, to define a first axial extension on the pressure side surface and a second axial extension on the suction side surface, each of the first and second axial extensions being in a range between 5% and 35% of the length of the chord line immediately downstream of the leading edge.

[0005] In another aspect, there is provided a gas turbine engine, comprising: a compressor section; a combustor section; and a turbine section; wherein the turbine section includes a plurality of rotors having a plurality of radially disposed rotor blades, each of the rotor blades including an attachment and an airfoil, the airfoil defining a pressure side surface and a suction side surface, extending between the attachment and a tip, a base region disposed adjacent to the attachment, a tip region, and a transition region located between the base region and the tip region, the tip region defining a chord line, and the pressure side surface in the tip region being substantially planar; and wherein the tip region includes a tip rib projecting outwardly from the airfoil, continuously surrounding a leading edge of the airfoil, extending from the leading edge rearwardly along the respective pressure side surface and suction side surface to define a first axial extension on the pressure side surface and a second axial extension on the suction side surface, each of the first and second axial extensions being in a range between 5% and 35% of the length of the chord line immediately downstream of the leading edge.

DESCRIPTION OF THE DRAWINGS

[0006] Reference is now made to the accompanying figures in which:

[0007] FIG. **1** is a partial schematic side cross-sectional view of a gas turbine engine, illustrating an example of the application of the described subject matter;

[0008] FIG. **2** is a schematic illustration of a rotor blade for the gas turbine engine in FIG. **1**;

[0009] FIG. **3** is a schematic illustration of a cross-sectional slice of the airfoil of the rotor blade of FIG. **2**;

[0010] FIG. **4** is a partial schematic front elevational view of the rotor blade of FIG. **2**; and

[0011] FIG. **5** is a top plan view of a tip of the rotor blade of FIG. **2**.

[0012] It will be noted that throughout the appended drawings, like features are identified by like reference numerals.

DETAILED DESCRIPTION

[0013] FIG. **1** illustrates a turbofan gas turbine engine according to one embodiment. The engine includes a housing or nacelle **10**, a core casing **13**, a low pressure spool assembly (not numbered) which includes a fan rotor **14**, a low pressure compressor assembly **16** and a low pressure turbine assembly **18** connected by a shaft **12**, and a high pressure spool assembly (not numbered) which includes a high pressure compressor assembly **22** and a high pressure turbine assembly **24** connected by a turbine shaft **20**. The housing or nacelle **10** surrounds the core casing **13** and in combination with the housing **10** and the core casing **13**, defines an annular bypass duct **28** for directing a bypass flow. The core casing **13** surrounds the low and high pressure spool assemblies to define a core fluid path **30** therethrough. In the core fluid path **30** there is provided a combustor **26** to form a combustion gas generator assembly which generates combustion gases in order to power the high pressure turbine assembly **24** and the low pressure turbine assembly **18**. The engine therefore defines in series a compressor section (not numbered), a combustor section (not numbered) and a turbine section (not numbered). Low and high pressure compressor assemblies **16**, **22** in the compressor section and the high and low pressure turbine assemblies **18**, **24** in the turbine section, each include a plurality of radially disposed rotor blades **32** attached to a rotor disc. The radially disposed rotor blades **32** are rotatable along a longitudinally extending axis **33** of the engine.

[0014] FIG. **2** is a schematic illustration of one embodiment of the rotor blade **32** for use, for example in the high pressure turbine assembly **24** of the gas turbine engine. The rotor blade **32** includes an attachment **34**, a platform **35**, and an airfoil **36**. Some embodiments of the rotor blade **32** may not include the platform **35**. To simplify the description herein, the attachment **34** may be considered to include the platform **35** for purpose of defining the beginning of the airfoil **36**. The rotor blade attachment **34** is adapted to be received within a slot disposed within the rotor disc. Rotor blade attachments are well known in the art and will not be described herein. The described subject matter is not limited to any particular attachment configuration.

[0015] The airfoil **36** has a leading edge **38**, a trailing edge **40**, a pressure side or pressure side surface **42**, and a suction side or suction side surface **44**. The pressure side surface **42** and the suction side surface **44** extend radially between the attachment **34** (including the platform **35**) and a tip **46**. The airfoil **36** includes a base region **50**, disposed adjacent to the attachment **34**, a tip region **54** disposed adjacent to the tip **46** and a transition region **52** disposed between the base region **50** and the tip region **54**. For convenience of description, the base region **50** may be defined between a first end **60** and a second end **62** thereof. The first end **60** of the base region **52** is located at a cross-sectional "slice" of the airfoil **36** where the base region **50** abuts the attachment **34**. The second end **62** of the base region **52** is located at a cross-sectional "slice" of the airfoil **36** where the base region **50** abuts the transition region **52**. The transition region **52** is defined between a first end **62**a and a second end **64** thereof. The first end **62***a* of the transition region **52** is located at the same cross-sectional "slice" of the airfoil **36** as the second end **62** of the base region **50**. The second end **64** of the transition region **52** is located at a cross-sectional "slice" of the airfoil **36** where the transition region **52** abuts the tip region **54**. The tip region **54** is defined between a first end **64**a and a second end **66** thereof. The first end **64***a* of the tip region **54** is located at the same cross-sectional "slice" of the airfoil **36** as the second end **64** of the transition region **52**. The second end **66** of the tip region **54** is located at the tip **46** of the airfoil **36**.

[0016] FIG. **3** schematically illustrates the cross-sectional "slices" of the airfoil **36** of FIG. **2**, forming the respective ends defining the base, transition and tip regions **50**, **52** and **54**. Stagger angles **56**, **56***a*, **56***b* and **56***c* in the respective cross-sectional "slices" are defined as the angle between a respective chord line **48**, **48***a*, **48***b* or **48***c* of the airfoil **36** and an axis (e.g. the longitudinally extending axis **33** of the gas turbine engine, etc.). In accordance with one embodiment the stagger angle may increase in a direction defined by a line (not shown) beginning at the attachment **34** and extending along the span of the airfoil **36** toward the tip **46**. In one embodiment the chord lines of the airfoil **36** such as indicated by **48**, **48***a*, **48***b* or **48***c* as well as their related stagger angles **56**, **56***a*, **56***b* and **56***c*, may increase as the airfoil extends from the attachment **34** to the tip **46**. A maximum camber line thickness of the airfoil **36** is measured by the maximum distance between a "mean camber line" **69**, **69***a*, **69***b* or **69***c* and the corresponding chord line **48**, **48***a*, **48***b* or **48***c*, and is indicated by respective arrows **68**, **68***a*, **68***b*, **68***c*. FIG. **3** illustrates that the maximum camber line airfoil thickness may vary along the airfoil span from the cross-sectional "slice" representing the first end **60** of the base region **50**, to the cross-sectional slice representing the second end **66** of the tip region **54**, such as by decreasing. The pressure side surface **42** adjacent to the tip **46** may be substantially planar or may be substantially parallel to the chord line **48***c* defined in the tip **46**.

[0017] The embodiment of the airfoil **32** as above-described is similar to the rotor blade described in US Patent Publication 2011/0135482 A1, published on Jun. 9, 2011 which is incorporated by reference herein. The rotor blade as above-described is a rotor blade with very low or no camber at the tip or a rotor blade having a substantially un-cambered symmetrical airfoil tip. The difference between the embodiment illustrated in FIGS. **2** and **3** and the rotor blade described in US 2011/0135482 A1 lies in that the rotor blade **32** as shown in FIGS. **2** and **3** includes a novel feature of a blade leading edge tip rib **70**.

[0018] Shrouded blade tips with fins are used to counter the blade tip clearance leakage however, this is not feasible for high pressure turbines. For an un-shrouded blade tip in high pressure turbines, various winglet configurations on the pressure and suction sides have been studied. The pull-stress created by the actual weight of the winglet configurations has been shown to be a drawback to its use in actual aero engines. It is even more challenging and has not been attempted, to provide a suitable winglet design for a blade with a very low or no camber at the tip. However, it has been found by computational fluid dynamics (CFD) analysis on a specific blade design, such as the airfoil **36** described above, that a mini-winglet at the leading edge, referred to herein as a leading edge tip rib **70**, smoothly blended on the pressure and suction side surfaces **42**, **44**, has shown to provide up to 80% of the improvement in stage efficiency with respect to a full tip rib (not shown) which encompasses the leading edge, pressure and suction side surfaces and trailing edge. Thus, most of the benefit is obtained for the leading edge tip rib **70** with minimum increase in pull-stress at the blade tip **46** and low impact on modal frequencies.

[0019] In accordance with one embodiment illustrated in FIGS. **3**, **4** and **5**, the rotor blade **32** may have a specific highly loaded and front-loaded airfoil tip loading distribution. The front-loaded airfoil tip loading footprint is that the minimum pressure may occur between 5% and 20% axial chord on the suction side surface **44** when the leading edge tip rib **70** is not incorporated with the airfoil **36**. The tip region **54** may include the leading edge tip rib **70** projecting outwardly from the airfoil **36**, continuously surrounding the leading edge **38** of the airfoil **36** and extending from the leading edge **38** rearwardly along the respective pressure side surface **42** and suction side surface **44** to define a first axial extension **72** on the pressure side surface **42** and a second axial extension **74** on the suction side surface **44**. The leading edge tip rib **70** in the tip region **54**, for example may be flush with the tip **46** of the airfoil **36** and smoothly blended along the pressure and suction side surfaces **42** and **44**, for example by a suitable casting requirement fillet **76**. In some embodiments the leading edge tip rib **70** may have a lateral extension indicated for example by arrows **78**, which may or may not be equal one to another on the leading edge **38**, pressure and suction side surfaces **42**, **44**. The axial extension **72** and **74** of the leading edge tip rib **70** on the respective pressure and suction side surfaces **42**, **44** may or may not be equal to each other. However, equal lateral and equal axial extents may be required from a structural point of view. The lateral extension **78** and the axial extensions **72**, **74** of the leading edge tip rib **70** may be selected depending on the blade pressure loading distribution and thus tend to be an optimization for a particular blade design.

[0020] In one embodiment of the rotor blade **32** the dimension for the lateral extension **78** may be selected between a range of 5% and 30% as thick as the maximum airfoil thickness **75** along the mean camber line **69***c* at the tip **46** (the preferred embodiment was 25%). The dimensions for the axial extension **72** or **74** of the leading edge tip rib **70** may be selected in a range of 5% and 35% of the length of the chord line immediately downstream of the leading edge **38**. Diminishing returns in stage efficiency gains have been shown outside these ranges and thus, optimum results are obtained with a minimum leading edge tip rib in size and weight that can lend itself applicable to high pressure turbines within structural limitations. The previous assertion assumed smooth blending of the axial extension **72** or **74** of the leading edge tip rib **70**. Without smooth axial blending, the first axial extension **72** on the pressure side surface **42** would be up to a range between 5% and 10% and the axial extension **74** on the suction side surface **44** would be up to a range between 5% to 20% of the length of the chord line, both normal to the airfoil walls, but this would be difficult to cast as a blade shape and maintain structural integrity when being ground for final blade tip radius at assembly. With the optimized leading edge tip rib **70**, the pass resistance for the leakage of fluid as it migrates from the pressure side surface to the suction side surface and from the leading edge to the suction side surface, is increased, thus altering the blade tip leakage vortex structure and strength. The latter leads to a stage efficiency benefit.

[0021] The above description is meant to be exemplary only, and one skilled in the art will recognize that changes may be made to the embodiments described without departing from the scope of the described subject matter. Still other modifications will be apparent to those skilled in the art, in light of a review of this disclosure and such modifications are intended to fall within the appended claims.

1. A rotor blade for a gas turbine engine, comprising:

an attachment; and

an airfoil defining a pressure side surface and a suction side surface, extending between the attachment and a tip, a base region disposed adjacent to the attachment, a tip region, and a transition region located between the base region and the tip region, the tip defining a chord line, and the pressure side surface adjacent to the tip being substantially planar; and

wherein the tip region includes a tip rib projecting outwardly from the airfoil, the tip rib continuously surrounding a leading edge of the airfoil, extending from the leading edge rearwardly along the respective pressure side surface and suction side surface, to define a first axial extension on the pressure side surface and a second axial extension on the suction side surface, each of the first and second axial extensions being in a range between 5% and 35% of the length of the chord line immediately downstream of the leading edge.

2. The rotor blade as defined in claim **1** wherein the tip rib has a lateral extension in a range between 5% and 30% as thick as the maximum airfoil thickness along the mean camber line.

3. The rotor blade as defined in claim **1** wherein the tip rib has a lateral extension which is 25% as thick as the maximum airfoil thickness along the mean camber line.

4. The rotor blade as defined in claim **1** wherein each of the first and second axial extensions is 35% of the length of the chord line immediately downstream of the leading edge.

5. The rotor blade as defined in claim **1** wherein tip rib is flush with the tip of the airfoil.

6. The rotor blade as defined in claim **1** wherein the chord line is substantially parallel to the pressure side surface.

7. The rotor blade as defined in claim **1** wherein a chord line increases as the airfoil extends from the attachment to the tip.

8. The rotor blade as defined in claim **1** wherein the first and second axial extensions of the tip rib are equal.

9. The rotor blade as defined in claim **1** wherein the first and second axial extensions of the tip rib are different.

10. The rotor blade as defined in claim **3** wherein the lateral extension on the respective leading edge, pressure side surface and suction side surface are equal.

11. The rotor blade as defined in claim **3** wherein the lateral extension varies on the leading edge, pressure side surface and suction side surface.

12. A gas turbine engine, comprising:

a compressor section;

a combustor section; and

a turbine section;

wherein the turbine section includes a plurality of rotors having a plurality of radially disposed rotor blades, each of the rotor blades including an attachment and an airfoil, the airfoil defining a pressure side surface and a suction side surface, extending between the attachment and a tip, a base region disposed adjacent to the attachment, a tip region, and a transition region located between the base region and the tip region, the tip region defining a chord line and the pressure side surface in the tip region being substantially planar; and

wherein the tip region includes a tip rib projecting outwardly from the airfoil, continuously surrounding a leading edge of the airfoil, extending from the leading edge rearwardly along the respective pressure side surface and suction side surface to define a first axial extension on the pressure side surface and a second axial extension on the suction side surface, each of the first and second axial extensions being in a range between 5% and 35% of the length of the chord line immediately downstream of the leading edge.

13. The gas turbine engine as defined in claim **12** wherein the tip rib has a lateral extension in a range between 5% and 30% as thick as the maximum airfoil thickness along the mean camber line.

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