

IN SITU GAS TURBINE PREVENTION OF CRACK GROWTH PROGRESSION

FIELD OF THE INVENTION

[0001] The present subject matter relates generally to gas turbine engines and, more particularly, to a system and method for performing an in situ repair of an internal component of a gas turbine engine.

BACKGROUND OF THE INVENTION

[0002] A gas turbine engine typically includes a turbomachinery core having a high pressure compressor, combustor, and high pressure turbine in serial flow relationship. The core is operable in a known manner to generate a primary gas flow. The high pressure compressor includes annular arrays (“rows”) of stationary vanes that direct air entering the engine into downstream, rotating blades of the compressor. Collectively one row of compressor vanes and one row of compressor blades make up a “stage” of the compressor. Similarly, the high pressure turbine includes annular rows of stationary nozzle vanes that direct the gases exiting the combustor into downstream, rotating blades of the turbine. Collectively one row of nozzle vanes and one row of turbine blades make up a “stage” of the turbine. Typically, both the compressor and turbine include a plurality of successive stages.

[0003] Gas turbine engines, particularly aircraft engines, require a high degree of periodic maintenance. For example, periodic maintenance is often scheduled to allow internal components of the engine to be inspected for defects and subsequently repaired. Unfortunately, many conventional repair methods used for aircraft engines require that the engine be removed from the body of the aircraft and subsequently partially or fully disassembled. As such, these repair methods result in a significant increase in both the time and the costs associated with repairing internal engine components.

[0004] Accordingly, a system and method for performing an in situ repair of an internal component of a gas turbine engine would be welcomed within the technology.

BRIEF DESCRIPTION OF THE INVENTION

[0005] Aspects and advantages of the invention will be set forth in part in the following description, or may be obvious from the description, or may be learned through practice of the invention.

[0006] A method is generally provided for remotely stopping a crack in a component of a gas turbine engine. In one embodiment, the method includes inserting an integrated repair interface attached to a cable delivery system within a gas turbine engine; positioning the tip adjacent to a defect defined on a surface on the component; and locally heating the base of the defect.

[0007] A method is also generally provided for clamping a crack defined between a first surface and a second surface of a component of a gas turbine engine. In one embodiment, the method includes attaching a strap over the crack such that a first end of the strap is attached to the first surface of the component and the second end of the strap is attached to the second surface of the component.

[0008] These and other features, aspects and advantages of the present invention will become better understood with reference to the following description and appended claims. The accompanying drawings, which are incorporated in and constitute a part of this specification, illustrate embodiments of the invention and, together with the description, serve to explain the principles of the invention.

BRIEF DESCRIPTION OF THE DRAWINGS

[0009] A full and enabling disclosure of the present invention, including the best mode thereof, directed to one of ordinary skill in the art, is set forth in the specification, which makes reference to the appended Figs., in which:

[0010] FIG. 1 illustrates a cross-sectional view of one embodiment of a gas turbine engine that may be utilized within an aircraft in accordance with aspects of the present subject matter;

[0011] FIG. 2 illustrates a partial, cross-sectional view of one embodiment of a turbine suitable for use within the gas turbine engine shown in FIG. 1, particularly illustrating access ports defined in the engine for providing internal access to the turbine;

[0012] FIG. 3 illustrates a partial, cross-sectional view of one embodiment of a compressor suitable for use within the gas turbine engine shown in FIG. 1, particularly illustrating access ports defined in the engine for providing internal access to the compressor;

[0013] FIG. 4 illustrates a simplified view of one embodiment of a system for performing an in situ repair of an internal component of a gas turbine engine in accordance with aspects of the present subject matter, particularly illustrating a repair tool inserted through an access port of the engine to access a defect of the internal component;

[0014] FIG. 5 illustrates a partial view of the repair tool temporarily secured to a tip of an airfoil in order to perform an in situ repair thereon;

[0015] FIG. 6 illustrates a partial view of one embodiment of the repair tool temporarily secured to a surface of an internal component of the gas turbine engine in order to perform an in situ repair thereon;

[0016] FIG. 7 illustrates a partial view of one embodiment of the repair tool temporarily secured to a surface of an internal component of the gas turbine engine in order to form a hole within a defect for in situ repair;

[0017] FIG. 8 illustrates a partial view after formation of the hole according to FIG. 7;

[0018] FIG. 9 illustrates a partial view of one embodiment of the repair tool temporarily secured to a surface of an internal component of the gas turbine engine in order to heat the base of a defect for in situ repair;

[0019] FIG. 10 illustrates a partial view of one embodiment of the repair tool temporarily secured to a surface of an internal component of the gas turbine engine in order to supply material within a defect for in situ repair;

[0020] FIG. 11 illustrates a simplified view of one embodiment of a system for performing an in situ repair of an internal component of a gas turbine engine in accordance with aspects of the present subject matter, particularly illustrating a repair tool inserted through an access port of the engine to access a defect of the internal component and supply a fill material into a defect on the component; and

[0021] FIG. 12 illustrates a partial view of one embodiment of a strap spanning a defect within the surface of an internal component of the gas turbine engine for in situ repair.

[0022] Repeat use of reference characters in the present specification and drawings is intended to represent the same or analogous features or elements of the present invention.

DETAILED DESCRIPTION OF THE INVENTION

[0023] Reference now will be made in detail to embodiments of the invention, one or more examples of which are illustrated in the drawings. Each example is provided by way of explanation of the invention, not limitation of the invention. In fact, it will be apparent to those skilled in the art that various modifications and variations can be made in the present invention without departing from the scope or spirit of the invention. For instance, features illustrated or described as part of one embodiment can be used with another embodiment to yield a still further embodiment. Thus, it is intended that the present invention covers such modifications and variations as come within the scope of the appended claims and their equivalents.

[0024] As used herein, the terms “first”, “second”, and “third” may be used interchangeably to distinguish one component from another and are not intended to signify location or importance of the individual components.

[0025] The terms “upstream” and “downstream” refer to the relative direction with respect to fluid flow in a fluid pathway. For example, “upstream” refers to the direction from which the fluid flows, and “downstream” refers to the direction to which the fluid flows.

[0026] In general, a system and method is provided for performing an in situ repair of an internal component of a gas turbine engine. In several embodiments, the system may include a repair tool configured to be inserted through an access port of the gas turbine engine to allow a repair tip or tip end of the tool to be positioned adjacent to a defect of an internal component of the engine, such as a crack, void, distressed area or any other defect defining a fillable volume. As will be described below, the repair tool may be configured to temporarily attach to the surface of the component, allowing precision work to be performed on the component. For

example, the repair tool can supply a filler material (solid or liquid), a drill, a cleaning agent, a heating element, etc.

[0027] For example, a drill can be utilized to drill a hole into the base of a crack (e.g., a fiber optic laser drill, a water jet, a bead blast, a mechanical twist drill, a grinding ball, etc.). In other embodiments, a filler material can be supplied from a location exterior to the engine to the internal location of the defect to allow the fillable volume defined by the defect to be filled with the filler material. For example, in one embodiment, the repair tool may be configured to supply liquid metal from the exterior of the engine into the fillable volume of the defect. The liquid metal may then cool and solidify within the volume, thereby filling-in and repairing the defect. In another embodiment, the repair tool may be configured to supply high velocity powder particles from the exterior of the engine into the fillable volume of the defect. Upon impacting a surface of the defect, the high velocity projectiles may plastically deform and adhere to the surface, thereby filling-in the fillable volume and repairing the defect.

[0028] It should be appreciated that the disclosed system and method may generally be used to perform in situ repairs of internal components located within any suitable type of gas turbine engine, including aircraft-based turbine engines and land-based turbine engines, regardless of the engine's current assembly state (e.g., fully or partially assembled). Additionally, with reference to aircraft engines, it should be appreciated that the present subject matter may be implemented on-wing or off-wing.

[0029] Referring now to the drawings, FIG. 1 illustrates a cross-sectional view of one embodiment of a gas turbine engine 10 that may be utilized within an aircraft in accordance with aspects of the present subject matter, with the engine 10 being shown having a longitudinal or axial centerline axis 12 extending therethrough for reference purposes. In general, the engine 10 may include a core gas turbine engine (indicated generally by reference character 14) and a fan section 16 positioned upstream thereof. The core engine 14 may generally include a substantially tubular outer casing 18 that defines an annular inlet 20. In addition, the outer casing 18 may further enclose and support a booster compressor 22 for increasing the pressure of the air that enters the core engine 14 to a first pressure level. A high pressure, multi-stage, axial-flow compressor 24 may then receive the pressurized air from the booster compressor 22

and further increase the pressure of such air. The pressurized air exiting the high-pressure compressor 24 may then flow to a combustor 26 within which fuel is injected into the flow of pressurized air, with the resulting mixture being combusted within the combustor 26. The high energy combustion products are directed from the combustor 26 along the hot gas path of the engine 10 to a first (high pressure) turbine 28 for driving the high pressure compressor 24 via a first (high pressure) drive shaft 30, and then to a second (low pressure) turbine 32 for driving the booster compressor 22 and fan section 16 via a second (low pressure) drive shaft 34 that is generally coaxial with first drive shaft 30. After driving each of turbines 28 and 32, the combustion products may be expelled from the core engine 14 via an exhaust nozzle 36 to provide propulsive jet thrust.

[0030] Additionally, as shown in FIG. 1, the fan section 16 of the engine 10 may generally include a rotatable, axial-flow fan rotor assembly 38 that is configured to be surrounded by an annular fan casing 40. It should be appreciated by those of ordinary skill in the art that the fan casing 40 may be configured to be supported relative to the core engine 14 by a plurality of substantially radially-extending, circumferentially-spaced outlet guide vanes 42. As such, the fan casing 40 may enclose the fan rotor assembly 38 and its corresponding fan rotor blades 44. Moreover, a downstream section 46 of the fan casing 40 may extend over an outer portion of the core engine 14 so as to define a secondary, or by-pass, airflow conduit 48 that provides additional propulsive jet thrust.

[0031] It should be appreciated that, in several embodiments, the second (low pressure) drive shaft 34 may be directly coupled to the fan rotor assembly 38 to provide a direct-drive configuration. Alternatively, the second drive shaft 34 may be coupled to the fan rotor assembly 38 via a speed reduction device 37 (e.g., a reduction gear or gearbox) to provide an indirect-drive or geared drive configuration. Such a speed reduction device(s) may also be provided between any other suitable shafts and/or spools within the engine 10 as desired or required.

[0032] During operation of the engine 10, it should be appreciated that an initial air flow (indicated by arrow 50) may enter the engine 10 through an associated inlet 52 of the fan casing 40. The air flow 50 then passes through the fan blades 44 and splits into a first compressed air flow (indicated by arrow 54) that moves through

conduit 48 and a second compressed air flow (indicated by arrow 56) which enters the booster compressor 22. The pressure of the second compressed air flow 56 is then increased and enters the high pressure compressor 24 (as indicated by arrow 58). After mixing with fuel and being combusted within the combustor 26, the combustion products 60 exit the combustor 26 and flow through the first turbine 28. Thereafter, the combustion products 60 flow through the second turbine 32 and exit the exhaust nozzle 36 to provide thrust for the engine 10.

[0033] The gas turbine engine 10 may also include a plurality of access ports defined through its casings and/or frames for providing access to the interior of the core engine 14. For instance, as shown in FIG. 1, the engine 10 may include a plurality of access ports 62 (only six of which are shown) defined through the outer casing 18 for providing internal access to one or both of the compressors 22, 24 and/or for providing internal access to one or both of the turbines 28, 32. In several embodiments, the access ports 62 may be spaced apart axially along the core engine 14. For instance, the access ports 62 may be spaced apart axially along each compressor 22, 24 and/or each turbine 28, 32 such that at least one access port 62 is located at each compressor stage and/or each turbine stage for providing access to the internal components located at such stage(s). In addition, the access ports 62 may also be spaced apart circumferentially around the core engine 14. For instance, a plurality of access ports 62 may be spaced apart circumferentially around each compressor stage and/or turbine stage.

[0034] It should be appreciated that, although the access ports 62 are generally described herein with reference to providing internal access to one or both of the compressors 22, 24 and/or for providing internal access to one or both of the turbines 28, 32, the gas turbine engine 10 may include access ports 62 providing access to any suitable internal location of the engine 10, such as by including access ports 62 that provide access within the combustor 26 and/or any other suitable component of the engine 10.

[0035] Referring now to FIG. 2, a partial, cross-sectional view of the first (or high pressure) turbine 28 described above with reference to FIG. 1 is illustrated in accordance with embodiments of the present subject matter. As shown, the first turbine 28 may include a first stage turbine nozzle 66 and an annular array of rotating

turbine blades 68 (one of which is shown) located immediately downstream of the nozzle 66. The nozzle 66 may generally be defined by an annular flow channel that includes a plurality of radially-extending, circularly-spaced nozzle vanes 70 (one of which is shown). The vanes 70 may be supported between a number of arcuate outer bands 72 and arcuate inner bands 74. Additionally, the circumferentially spaced turbine blades 68 may generally be configured to extend radially outwardly from a rotor disk (not shown) that rotates about the centerline axis 12 (FIG 1) of the engine 10. Moreover, a turbine shroud 76 may be positioned immediately adjacent to the radially outer tips of the turbine blades 68 so as to define the outer radial flowpath boundary for the combustion products 60 flowing through the turbine 28 along the hot gas path of the engine 10.

[0036] As indicated above, the turbine 28 may generally include any number of turbine stages, with each stage including an annular array of nozzle vanes and follow-up turbine blades 68. For example, as shown in FIG. 2, an annular array of nozzle vanes 78 of a second stage of the turbine 28 may be located immediately downstream of the turbine blades 68 of the first stage of the turbine 28.

[0037] Moreover, as shown in FIG. 2, a plurality of access ports 62 may be defined through the turbine casing and/or frame, with each access port 62 being configured to provide access to the interior of the turbine 28 at a different axial location. Specifically, as indicated above, the access ports 62 may, in several embodiments, be spaced apart axially such that each access port 62 is aligned with or otherwise provides interior access to a different stage of the turbine 28. For instance, as shown in FIG. 2, a first access port 62A may be defined through the turbine casing/frame to provide access to the first stage of the turbine 28 while a second access port 62B may be defined through the turbine casing/frame to provide access to the second stage of the turbine 28.

[0038] It should be appreciated that similar access ports 62 may also be provided for any other stages of the turbine 28 and/or for any turbine stages of the second (or low pressure) turbine 32. It should also be appreciated that, in addition to the axially spaced access ports 62 shown in FIG. 2, access ports 62 may be also provided at differing circumferentially spaced locations. For instance, in one embodiment, a plurality of circumferentially spaced access ports may be defined through the turbine

casing/frame at each turbine stage to provide interior access to the turbine 28 at multiple circumferential locations around the turbine stage.

[0039] Referring now to FIG. 3, a partial, cross-sectional view of the high pressure compressor 24 described above with reference to FIG. 1 is illustrated in accordance with embodiments of the present subject matter. As shown, the compressor 24 may include a plurality of compressor stages, with each stage including both an annular array of fixed compressor vanes 80 (only one of which is shown for each stage) and an annular array of rotatable compressor blades 82 (only one of which is shown for each stage). Each row of compressor vanes 80 is generally configured to direct air flowing through the compressor 24 to the row of compressor blades 82 immediately downstream thereof.

[0040] Moreover, the compressor 24 may include a plurality of access ports 62 defined through the compressor casing/frame, with each access port 62 being configured to provide access to the interior of the compressor 24 at a different axial location. Specifically, in several embodiments, the access ports 62 may be spaced apart axially such that each access port 62 is aligned with or otherwise provides interior access to a different stage of the compressor 24. For instance, as shown in FIG. 3, first, second, third and fourth access ports 62a, 62b, 62c, 62d are illustrated that provide access to four successive stages, respectively, of the compressor 24.

[0041] It should be appreciated that similar access ports 62 may also be provided for any of the other stages of the compressor 24 and/or for any of the stages of the low pressure compressor 22. It should also be appreciated that, in addition to the axially spaced access ports 62 shown in FIG. 3, access ports 62 may be also provided at differing circumferentially spaced locations. For instance, in one embodiment, a plurality of circumferentially spaced access ports may be defined through the compressor casing/frame at each compressor stage to provide interior access to the compressor 24 at multiple circumferential locations around the compressor stage.

[0042] Referring now to FIG. 4, a simplified view of one embodiment of a system 100 for performing an in situ repair of an internal component of a gas turbine engine 10 are illustrated in accordance with aspects of the present subject matter. As shown, the system 100 may include a repair tool 102 configured to be inserted through an access port 62 of the gas turbine engine 10, such as any of the access ports 62

described above with reference to FIGS. 1-3, to allow an in situ repair procedure to be performed on an internal component(s) (indicated by dashed lines 104) of the engine 10.

[0043] In general, the repair tool 102 may correspond to any suitable tool(s) and/or component(s) that may be inserted through an access port 62 of the gas turbine engine 10 and attach onto the surface 105 of the component 104 to perform precision work thereon. For example, an attachment mechanism 135 can temporarily attach onto the surface 105 so that the tool 102 can perform work at or near an identified defect 106 of the internal engine component(s) 104 being repaired (e.g., a turbine blade(s)). As such, the repair tool 102 may be temporarily attached to the surface 105 so as to allow for precision work at the defect 106 (e.g., with precision accuracy within about 0.5 mm or less, such as about 0.25 mm or less). As generically shown in FIG. 4, a conduit 110 is attached to a working head 122 includes a work mechanism 124 controllable via a controller 114 (e.g., a computer or other programmable machine).

[0044] In one embodiment, the attachment mechanism 135 can be a tripod grip for a component 104 having a known shape and/or size. As shown in Fig. 5, the component 104 is an airfoil tip 200 with a known shape and size (e.g., a nozzle and/or blade). In other embodiments, the component 104 can be a trailing edge and/or leading edge of the airfoil. The attachment mechanism 135 includes a plurality of grip arms 150 that attach the repair tool 102 onto the surface 105. The grip arms 150 are brought together onto the edge of the tip 200 until the repair tool 102 is secured onto the tip 200. In the embodiment shown, three grip arms 150 are included in the attachment mechanism 135, although any suitable number of grip arms 150 may be utilized (e.g., three or more grip arms).

[0045] In another embodiment, the attachment mechanism 135 can be a suction cup attached onto the repair tool 102. As shown in Fig. 6, the attachment mechanism 135 includes a suction cup 160 that attach the repair tool 102 onto the surface 105. In one embodiment, a vacuum can be applied within the suction cup 160 to hold the repair tool 102 onto the surface in place. The suction cup 160 can be constructed of a deformable, air-impervious material (e.g., a rubber material) that can form a suction attachment with the surface 105. Although shown with one suction cup 160, any

number of suction cups can be utilized to secure the repair tool 102 onto the surface 105. In yet another embodiment, an adhesive can be utilized to secure the repair tool 102 onto the surface 105, such as a hot melt adhesive, epoxy material, etc. Then, the adhesive material can be melted to remove the repair tool 102 from the surface 105.

[0046] Through the attachment mechanism 135, the location of repair tool 102 can be precisely controlled and temporarily secured in place, which allows for precision work to be performed. In one embodiment, a working head 122 is positioned and secured adjacent to the identified defect 106 of the internal engine component(s) 104 being repaired (e.g., a turbine blade(s)). For example, as particularly shown in FIG. 4, the defect 106 corresponds to a crack, void or other defective area formed along the exterior of the component 104 that defines an open or fillable volume 108 with a base 107 of the crack, void or other defective area.

[0047] As shown in FIGS. 5-9, the working head 122 includes a work mechanism 124 configured for addressing the defect 106. In the embodiment of FIGS. 6-7, the work mechanism 124 may include a drill component 125, such as a fiber optic laser drill, a water jet, a bead blaster, a mechanical twist drill, a grinding ball, etc., that is directed to the base 107 of the defect 106 in the surface 105 of the component 104. Referring to FIG. 8, the drill component 125 can be used to form a precision hole 109 within the base 107 of the defect 106 (e.g., at the deepest point from the surface 105 within the component 104). In one embodiment, the hole 109 can be formed to be about 10% to about 50% of the size of the base 107 of the defect 106. For example, the hole 109 can have a hole size (i.e., average diameter in the plane perpendicular to or tangent to the surface 105 at the defect 106) that is about 0.1 mm to about 3.25 mm. The precision hole 109 can effectively stop the propagation of the defect 106 through the component 104 by forming a more flexible joint at the base 107.

[0048] Alternatively or additionally, as shown in FIG. 9, the working head 122 may include a heating component 127 to locally heat the base of the defect 106. For example, the heating component 127 may direct thermal energy (represented as arrow 129) to the base 107 of the defect 106 in the surface 105 of the component 104, as shown in Fig. 9. The heating component 127 can heat a precision weld 111 within the base 107 of the defect 106 (e.g., at the deepest point from the surface 105 within the

component 104) to effectively stop the propagation of the defect 106 through the component 104.

[0049] For example, the base 107 may be heated to a temperature of about 1000 °C to about 2000 °C (e.g., about 1800° C to about 2000° C), particularly with the component 104 is constructed from a metal alloy or super-alloy such as a nickel-based alloy, a chromium-based alloy, etc.

[0050] In one embodiment, the working head 122 may be utilized to clean an area on the surface 105 of the component 104. For example, the working head 122 can supply a cleaning agent to the surface 105 of the component 104, such as through a water jet, grit blast, chemical spray, etc.

[0051] In another embodiment, a filler material can be supplied to the location of the defect via the repair tool 102 such that the fillable volume 108 may be filled-in with the filler material, thereby repairing the defect 106. In several embodiments, the repair tool 102 may be configured to supply liquid metal within the interior of the gas turbine engine 10 as the filler material. For example, liquid metal may be transported via the repair tool 102 from a location exterior to the gas turbine engine 10 to a location within the engine 10 to allow the liquid metal to be injected or otherwise directed into the fillable volume 108 defined by the defect 106. The liquid metal may then solidify within the fillable volume 108 as the metal cools.

[0052] It should be appreciated that the liquid metal may generally correspond to any suitable metal material. For example, in one embodiment, the liquid metal may correspond to the parent metal material of the internal component 104 being repaired. In other embodiments, the liquid metal may correspond to any other metal material that is suitable for use as a repair material within a gas turbine engine 10.

[0053] As shown in the illustrated embodiment of FIG. 4, the repair tool 102 may include a high temperature conduit 110 for transporting the liquid metal from outside the engine 10 to the location of the defect 106. Specifically, as shown in FIG. 4, the high temperature conduit 110 may extend lengthwise between working head 122 located within the gas turbine engine 10 and a material supply end 114 located exterior to the engine 10. The tip end of the tool 102 may generally be positioned adjacent to the location of the defect 106 for directing the liquid metal into the fillable volume 108. Additionally, the material supply end 114 of the tool 102 may generally

be configured to receive liquid metal from a liquid metal source. For example, as shown in FIGS. 10 and 11, liquid metal contained within a furnace (or other suitable liquid metal source) located exterior to the gas turbine engine 10 may be supplied to the material supply end 114 of the tool 102. The liquid metal received at the material supply end 114 may then be directed through the high temperature conduit 110 to the tip end of the tool 102 to allow the liquid metal to be delivered to the location of the defect 106.

[0054] It should be appreciated that the high temperature conduit 110 may generally be formed from any suitable high temperature material that allows the conduit 110 to serve as a fluid delivery means for the liquid metal. For example, in several embodiments, the high temperature conduit 110 may be formed from a ceramic material capable of withstanding temperatures above the melting temperature of the metal being supplied to the defect 106. However, in other embodiments, the conduit 110 may be formed from any other suitable high temperature material.

[0055] Additionally, as particularly shown in FIGS. 10 and 11, the repair tool 102 may include one or more heating elements (indicated by dashed lines 120) provided in operative association within the high temperature conduit 110. In general, the heating element(s) 120 may be configured to generate heat within the high temperature conduit 110 as liquid metal is being supplied through the conduit 110 so as to maintain the metal in its liquid state. For example, in one embodiment, the heating element(s) 120 may correspond to a resisting heating element(s), such as one or more resistance wires, that is integrated into or incorporated within a wall(s) of the conduit 110. However, in another embodiment, the heating element(s) 120 may correspond to any other suitable heat generating device(s) and/or component(s) that may be used to provide heating within the conduit 110 so as to maintain the temperature of the liquid metal at or above its melting temperature.

[0056] In one embodiment, the repair tool 102 includes an optical probe 130 adjacent to the working head 122 and configured to be used in association with the repair tool 102. For instance, as shown in FIG. 4, the optical probe 130 corresponds to a separate component configured to be used in combination with the repair tool 102 for repairing the defect 106. However, in other embodiments, the optical probe 130 may be coupled to or integrated within the repair tool 102. Additionally, as shown in

FIG. 4, the optical probe 130 has been inserted through the same access port 62 as the repair tool 102. However, in other embodiments, the probe 130 may be inserted into a different access port 62 than the repair tool 102, such as an access port 62 located adjacent to the access port 62 within which the repair tool 102 has been inserted.

[0057] In general, the optical probe 130 may correspond to any suitable optical device that allows images of the interior of the engine 10 to be captured or otherwise obtained. For instance, in several embodiments, the optical probe 130 may correspond to a borescope, videoscope, fiberscope or any other similar optical device known in the art that allows for the interior of a gas turbine engine 10 to be viewed through an access port 62. In such embodiments, the optical probe 130 may include one or more optical elements (indicated schematically by dashed box 132), such as one or more optical lenses, optical fibers, image capture devices, cables, and/or the like, for obtaining views or images of the interior of the engine 10 at a tip 134 of the probe 130 and for transmitting or relaying such images from the probe tip 134 along the length of the probe 130 to the exterior of the engine 10 for viewing by the personnel performing the repair procedure on the internal component(s) 104. In addition, the probe 130 may include a light source (indicated by dashed box 136) positioned at or adjacent to the probe tip 134 to provide lighting within the interior of the engine 10.

[0058] As shown in FIGS. 4 and 11, the optical probe 130 may also include an articulation assembly 138 that allows the orientation of the probe tip 134 to be adjusted within the interior of the gas turbine engine 10. For example, the articulation assembly 138 may allow for the probe tip 134 to be rotated or pivoted about a single axis or multiple axes to adjust the orientation of the tip 134 relative to the remainder of the probe 130. It should be appreciated that the articulation assembly 138 may generally have any suitable configuration and/or may include any suitable components that allow for adjustment of the orientation of the probe tip 134 relative to the remainder of the probe 130. For example, in one embodiment, a plurality of articulation cables 140 may be coupled between the probe tip 134 and one or more articulation motors 142. In such an embodiment, by adjusting the tension of the cables 140 via the motor(s) 142, the probe tip 134 may be reoriented within the gas turbine engine 10.

[0059] In one particular embodiment, the articulation assembly 138 also controls the attachment mechanism 135 so as to temporarily attach to the surface 105 the component 104 in order to perform the desired work thereon.

[0060] In still another embodiment, a strap 300 may be attached onto opposite surfaces of the defect 106, such as generally depicted in FIG. 12. As shown, the strap 300 spans over the defect 106 such that the first end 304 is attached to the first surface 105a and the second end 306 is attached to the second surface 105b. As shown, each end 304, 306 of the strap 300 defines an attachment aperture 301 allowing an attachment pin 302 (e.g., a rivet, a screw, a bolt, etc.) to secure the ends 304, 306 of the strap 300 onto the respective surface 105a, 105b. Alternatively, adhesive and/or weld can be utilized to attach the ends 304, 306 of the strap 300 to the respective surface 105a, 105b. In certain embodiments, the integrated repair interface shown in FIGS. 1-11 can be utilized and precisely attach the strap 300 over the defect 106.

[0061] Methods are generally provided for performing an in situ repair of an internal component of a gas turbine engine. In general, the methods are discussed herein with reference to the gas turbine engine 10 and the system 100 described above with reference to FIGS. 1-12. However, it should be appreciated by those of ordinary skill in the art that the disclosed methods may generally be implemented with gas turbine engines having any other suitable engine configuration and/or with systems having any other suitable system configuration. In addition, although the methods are discussed in a particular order for purposes of discussion, the methods discussed herein are not limited to any particular order or arrangement. One skilled in the art, using the disclosures provided herein, will appreciate that various steps of the methods disclosed herein can be omitted, rearranged, combined, and/or adapted in various ways without deviating from the scope of the present disclosure.

[0062] The method may include inserting a repair tool through an access port of the gas turbine engine such that the tool includes a tip end positioned within the engine; positioning the tip adjacent to a defect (e.g., a crack or other distress point) within the surface of the component; and temporarily attaching the tip adjacent to the defect to allow precision work to be performed. For example, as indicated above, the method may include positioning the tip end of the repair tool adjacent to a defect of an internal component of the gas turbine engine. As indicated above, the defect 106

may, for example, correspond to a crack, void or other defective area of an internal component 104 of the gas turbine engine 10.

[0063] Moreover, the method may include performing precision repair work (e.g., drilling, cleaning, heating, supplying liquid metal, securing a strap, etc.) using the repair tool by temporarily attaching the tip end of the repair tool to the surface of the component.

[0064] This written description uses examples to disclose the invention, including the best mode, and also to enable any person skilled in the art to practice the invention, including making and using any devices or systems and performing any incorporated methods. The patentable scope of the invention is defined by the claims, and may include other examples that occur to those skilled in the art. Such other examples are intended to be within the scope of the claims if they include structural elements that do not differ from the literal language of the claims, or if they include equivalent structural elements with insubstantial differences from the literal languages of the claims.

WHAT IS CLAIMED IS:

1. A method of remotely stopping a crack in a component of a gas turbine engine, the method comprising:
 - inserting an integrated repair interface attached to a cable delivery system within a gas turbine engine;
 - positioning the tip adjacent to a defect defined on a surface on the component, wherein the defect defines a base; and
 - locally heating the base of the defect.
2. The method as in claim 1, further comprising:
 - temporarily attaching the tip adjacent to the defect defined within the surface of the component.
3. The method as in claim 2, wherein the tip is attached using a tripod grip, a glue, or a vacuum sucker.
4. The method as in claim 1, further comprising:
 - prior to locally heating the hole, drilling a hole into the base of the defect.
5. The method as in claim 4, wherein the hole is drilled using a fiber optic laser drill, a water jet, a bead blast, a mechanical twist drill, or a grinding ball.
6. The method as in claim 4, wherein the hole is formed to be about 10% to about 50% of the size of the base of the defect.
7. The method as in claim 4, wherein the hole is drilled in a desired location with precision accuracy within about 0.5 mm or less.
8. The method as in claim 1, wherein the component includes a metal alloy, and wherein the base is heated to a temperature of about 1000 °C to about 2000 °C.
9. The method as in claim 1, wherein the component includes a metal alloy, and wherein the base is heated to a temperature of about 1800 °C to about 2000 °C.
10. A method of clamping a crack defined between a first surface and a second surface of a component of a gas turbine engine, the method comprising:
 - attaching a strap over the crack such that a first end of the strap is attached to the first surface of the component and the second end of the strap is attached to the second surface of the component.

11. The method as in claim 10, wherein the strap has a thickness of about 250 μm to about 3 mm.
12. The method as in claim 10, wherein the strap has a thickness of about 250 μm to about 1.3 mm.
13. The method as in claim 10, wherein the first end of the strap defines an attachment aperture, and wherein the first end is attached to the first surface via an attachment pin, and further wherein the second end of the strap defines an attachment aperture, and wherein the second end is attached to the second surface via an attachment pin.
14. The method as in claim 10, wherein the first end of the strap is attached to the first surface via a weld or braze attachment, and wherein the second end of the strap is attached to the second surface via a weld or braze attachment.
15. The method as in claim 10, wherein the first end of the strap is attached to the first surface via an adhesive, and wherein the second end of the strap is attached to the second surface via an adhesive.
16. The method as in claim 10, further comprising:
 - inserting an integrated repair interface attached to a cable delivery system within a gas turbine engine;
 - positioning the tip adjacent to the crack; and
 - thereafter, attaching the strap over the crack.
17. The method as in claim 16, further comprising:
 - after positioning the tip adjacent to the crack, temporarily attaching the tip adjacent to the crack; and
 - thereafter, attaching the strap over the crack.
18. The method as in claim 10, wherein the strap spans a surface of the component to cover distress points defined within the surface.
19. The method as in claim 10, wherein the strap is clamped about a curved portion of the component.
20. The method as in claim 20, wherein the strap has a shared attachment point on either end.