US010443385B2

(12) **United States Patent**
Diwinsky et al.

(10) **Patent No.: US 10,443,385 B2**
(45) **Date of Patent: Oct. 15, 2019**

(54) **IN SITU GAS TURBINE PREVENTION OF CRACK GROWTH PROGRESSION VIA LASER WELDING**

(71) Applicant: **General Electric Company**, Schenectady, NY (US)

(72) Inventors: **David Scott Diwinsky**, West Chester, OH (US); **Herbert Chidsey Roberts**, Middletown, OH (US)

(73) Assignee: **General Electric Company**, Schenectady, NY (US)

(*) Notice: Subject to any disclaimer, the term of this patent is extended or adjusted under 35 U.S.C. 154(b) by 279 days.

(21) Appl. No.: **15/014,095**

(22) Filed: **Feb. 3, 2016**

(65) **Prior Publication Data**

US 2017/0218763 A1 Aug. 3, 2017

(51) **Int. Cl.**
F01D 5/00 (2006.01)
B23K 26/08 (2014.01)
(Continued)

(52) **U.S. Cl.**
CPC ***F01D 5/005*** (2013.01); ***B23K 26/032*** (2013.01); ***B23K 26/0884*** (2013.01);
(Continued)

(58) **Field of Classification Search**
CPC F01D 5/005; B23P 6/045; B23P 6/007; B23K 26/342; B23K 26/144;
(Continued)

(56) **References Cited**

U.S. PATENT DOCUMENTS

2,011,484 A 8/1935 Harman
2,252,986 A 8/1941 Scott
(Continued)

FOREIGN PATENT DOCUMENTS

CN 2760614 Y 2/2006
CN 101342628 A 1/2009
(Continued)

OTHER PUBLICATIONS

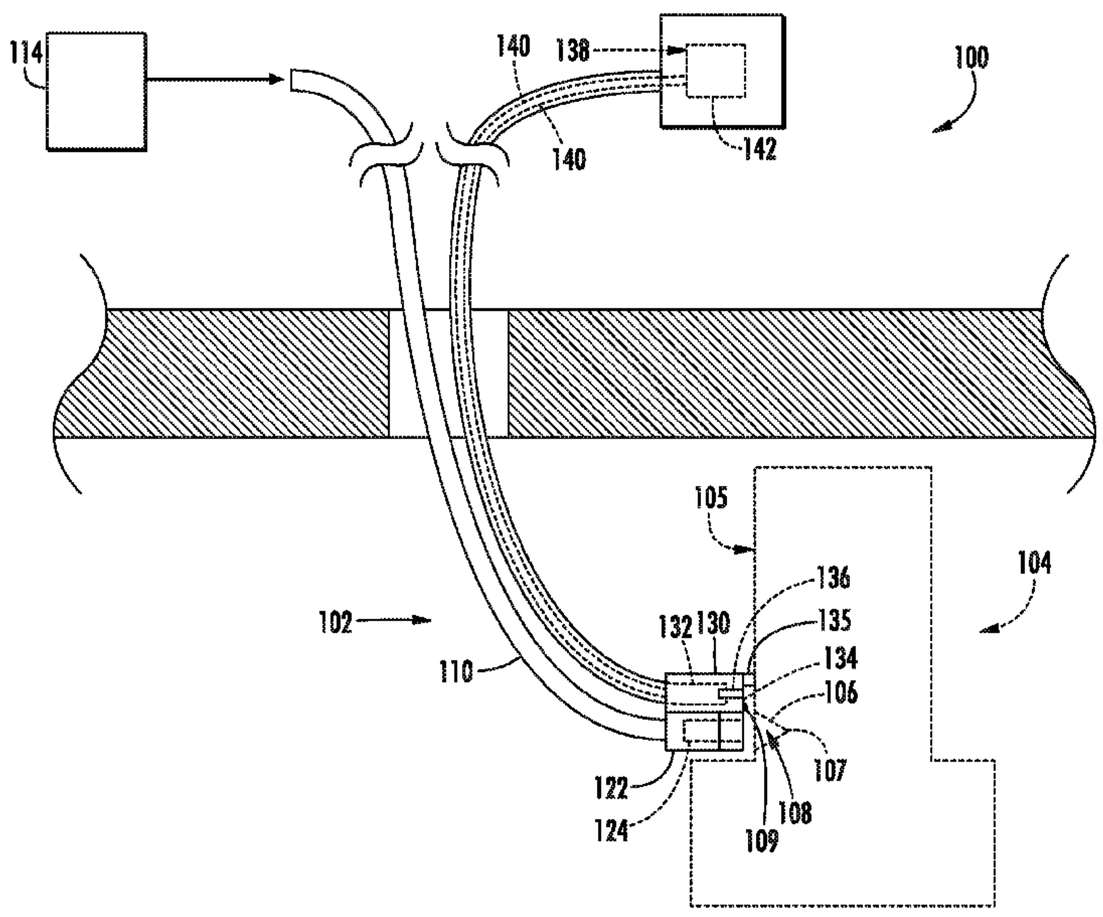
Extended European Search Report and Opinion issued in connection with corresponding EP Application No. 17153816.8 dated Jul. 10, 2017.
(Continued)

Primary Examiner — Ryan J. Walters
Assistant Examiner — Lawrence Averick
(74) *Attorney, Agent, or Firm* — Dority & Manning, P.A.

(57) **ABSTRACT**

Methods for material build-up on a tip of a blade of a gas turbine engine are provided. The method can include inserting a material supply adjacent to the tip and directing a laser onto the interface of the material supply and the tip such that the material supply melts and attaches to the tip. Methods are also provided for remotely stopping a crack in a component of a gas turbine engine. The method can include inserting an integrated repair interface attached to a cable delivery system within a gas turbine engine; positioning the tip adjacent to a defect within a surface of the component; supplying a new material to the fillable area to fill the defect; and directing a laser to the new material within the fillable area to fuse the new material to the component within the defect.

14 Claims, 7 Drawing Sheets



(51) **Int. Cl.**

B23P 6/00	(2006.01)
B23K 26/03	(2006.01)
G01N 21/954	(2006.01)
B23K 26/342	(2014.01)
B23K 26/70	(2014.01)
B23K 26/144	(2014.01)
B23P 6/04	(2006.01)
B23K 101/00	(2006.01)
B23K 103/18	(2006.01)

(52) **U.S. Cl.**

CPC ***B23K 26/144*** (2015.10); ***B23K 26/342*** (2015.10); ***B23K 26/702*** (2015.10); ***B23P 6/007*** (2013.01); ***B23P 6/045*** (2013.01); ***G01N 21/954*** (2013.01); *B23K 2101/001* (2018.08); *B23K 2103/26* (2018.08); *F05D 2230/234* (2013.01); *F05D 2230/80* (2013.01); *F05D 2260/83* (2013.01)

(58) **Field of Classification Search**

CPC B23K 26/702; B23K 26/0884; B23K 26/032; B23K 2201/001; B23K 2203/26; F05D 2260/83; F05D 2230/80; F05D 2230/234

See application file for complete search history.

(56) **References Cited**

U.S. PATENT DOCUMENTS

2,361,106	A	10/1944	Jensen	
4,611,744	A	9/1986	Fraser et al.	
4,743,733	A	5/1988	Mehta et al.	
5,019,429	A *	5/1991	Moskowitz	B05B 7/205 427/422
5,062,904	A *	11/1991	Yoden	G11B 5/712 148/105
5,076,660	A	12/1991	Messinger	
5,160,822	A *	11/1992	Aleshin	F01D 5/20 219/121.64
5,302,414	A *	4/1994	Alkhimov	B05B 7/144 427/191
5,741,965	A	4/1998	Hernandez et al.	
6,206,067	B1	3/2001	Kociemba et al.	
6,269,540	B1	8/2001	Islam et al.	
6,270,603	B1	8/2001	Westerman et al.	
6,302,625	B1	10/2001	Carey et al.	
6,525,500	B2	2/2003	Hatley et al.	
6,542,230	B1 *	4/2003	Luke	B23K 26/032 356/241.1
6,593,540	B1	7/2003	Baker et al.	
6,620,457	B2	9/2003	Farmer et al.	
6,912,446	B2	6/2005	Wang et al.	
6,919,956	B2	7/2005	Kitagawa et al.	
7,000,303	B2	2/2006	Talwar et al.	
7,016,035	B2 *	3/2006	Wu	G01J 3/02 356/318
7,032,279	B2	4/2006	McCarvill et al.	
7,034,262	B2	4/2006	Fischer et al.	
7,171,279	B2	1/2007	Buckingham et al.	
7,174,797	B2	2/2007	Brostmeyer et al.	
7,278,208	B2	10/2007	LeSaint et al.	
7,392,713	B2	7/2008	Barkhoudarian	
7,449,658	B2	11/2008	Mielke	
7,458,768	B2	12/2008	Dube et al.	
7,499,772	B2	3/2009	Wilcox et al.	
7,513,024	B2	4/2009	Keller	
8,039,773	B2	10/2011	Spallek et al.	
8,181,528	B2	3/2012	Reed et al.	
8,247,722	B2 *	8/2012	Ominato	D05B 69/04 200/314
8,247,733	B2 *	8/2012	Zhu	B23K 26/03 219/121.63
8,348,136	B1	1/2013	Potter et al.	
8,400,501	B2	3/2013	Heyworth et al.	
8,563,080	B2	10/2013	Hopkins	
8,786,300	B2	7/2014	Bendall	
8,801,502	B2	8/2014	Ng et al.	
8,895,887	B2	11/2014	Trapp et al.	
8,904,635	B2	12/2014	Berkebile et al.	
8,910,359	B2	12/2014	Jones et al.	
8,927,897	B2	1/2015	Xu et al.	
9,003,880	B2	4/2015	Sbihli et al.	
9,046,892	B2	6/2015	Jang et al.	
2005/0068524	A1 *	3/2005	Wu	G01J 3/02 356/316
2006/0042083	A1 *	3/2006	Baker	B23K 26/0096 29/889.1
2009/0243176	A1 *	10/2009	Alexander	B25B 5/14 269/21
2010/0176097	A1 *	7/2010	Zhu	B23K 26/03 219/121.63
2012/0000890	A1	1/2012	Ito et al.	
2012/0121382	A1 *	5/2012	Xu	F01D 25/002 415/118
2012/0286109	A1 *	11/2012	Clark	F01D 21/003 248/70
2013/0167375	A1 *	7/2013	Roesing	B23P 15/00 29/888.3
2013/0174869	A1	7/2013	Rösing et al.	
2013/0289766	A1	10/2013	Hafenrichter et al.	
2013/0305508	A1 *	11/2013	Sachacht	B23P 6/002 29/402.18
2013/0316183	A1 *	11/2013	Kulkarni, Jr.	B23P 6/007 428/557
2014/0063228	A1	3/2014	Boles et al.	
2014/0144473	A1	5/2014	Martin	
2014/0202286	A1 *	7/2014	Yokoyama	B22F 9/08 75/331
2014/0223737	A1	8/2014	Kell et al.	
2014/0278221	A1	9/2014	Troy et al.	
2015/0034266	A1	2/2015	Bruck et al.	
2015/0036127	A1	2/2015	Konomura et al.	
2015/0174838	A1	6/2015	Kittleson et al.	
2016/0229005	A1 *	8/2016	Ryan	F01D 25/285
2016/0279722	A1 *	9/2016	Capriotti	B23H 3/04
2016/0318135	A1 *	11/2016	Raulerson	F01D 5/005
2017/0218761	A1 *	8/2017	Roberts	B23K 26/032
2017/0218762	A1 *	8/2017	Roberts	F01C 21/10
2017/0218764	A1 *	8/2017	Roberts	F01D 5/005
2017/0254205	A1	9/2017	Scott et al.	
2017/0326685	A1 *	11/2017	Kell	B23K 26/0096

FOREIGN PATENT DOCUMENTS

DE	3813157	A1	12/1988
DE	10 2009 004661	A1	7/2010
EP	1609957	A2	12/2005
EP	2711759	A1	3/2014
WO	WO00/06336		2/2000
WO	WO2005/049263	A1	6/2005
WO	WO2013/097944	A1	7/2013

OTHER PUBLICATIONS

Machine Translation and First Office Action and Search issued in connection with corresponding CN Application No. 201710056205.5 dated May 4, 2018.

Office Action issued in connection with corresponding CA Application No. 2956355 dated Aug. 1, 2018.

* cited by examiner

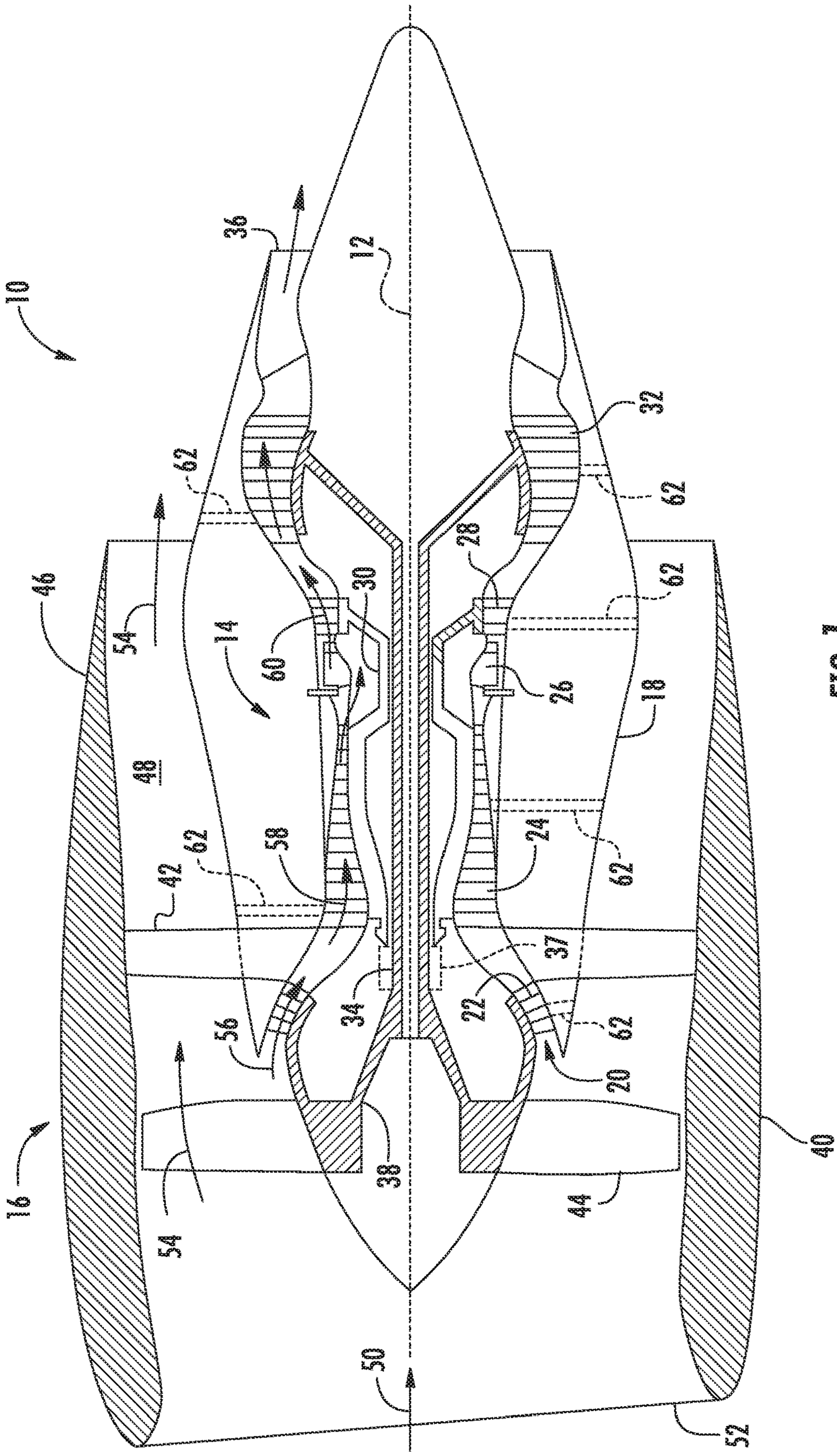

10
36
12
32
62
28
46
54
30
14
60
26
18
48
24
42
58
37
34
22
56
20
38
44
16
40
50
52

FIG. 1

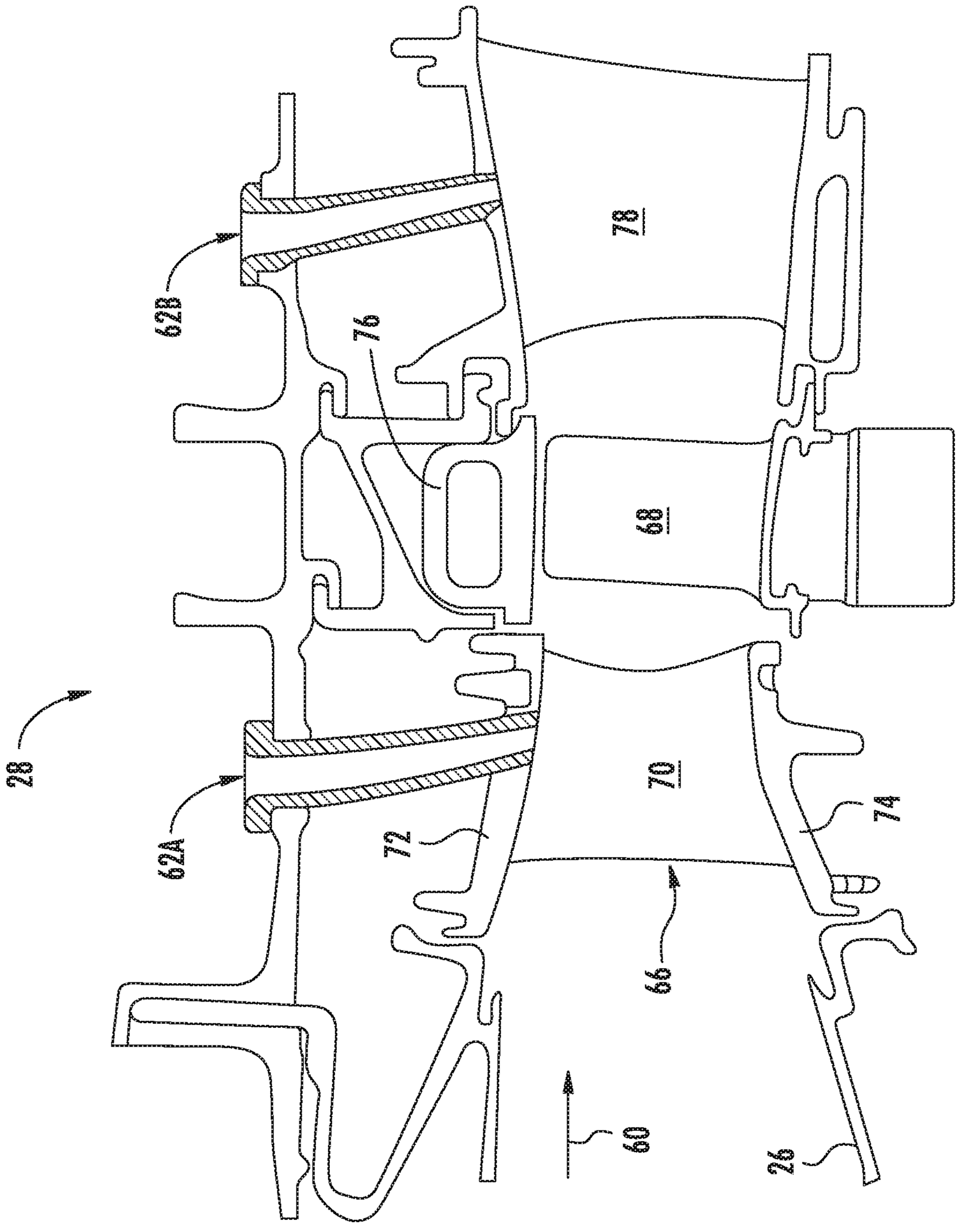


FIG. 2

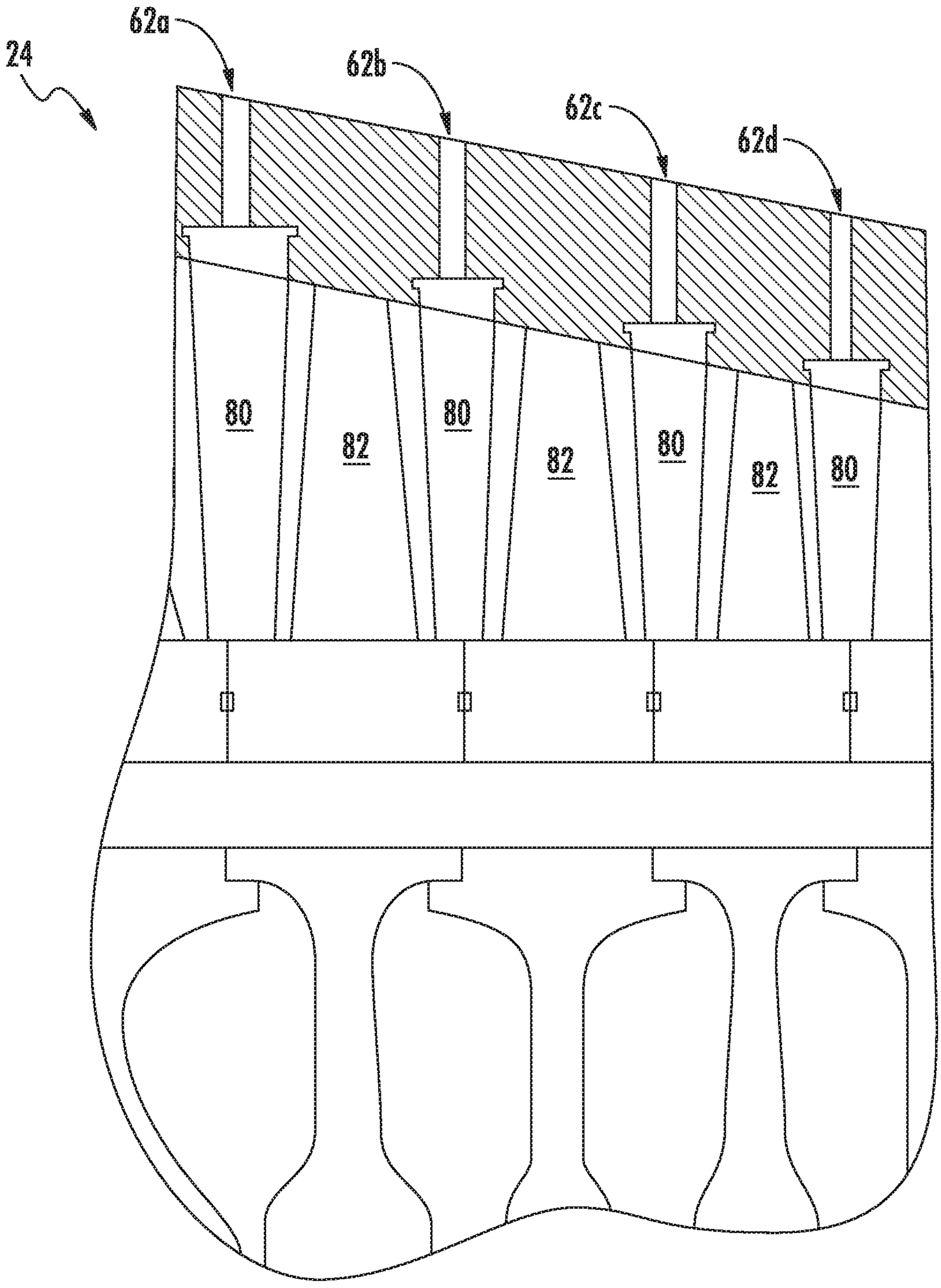


FIG. 3

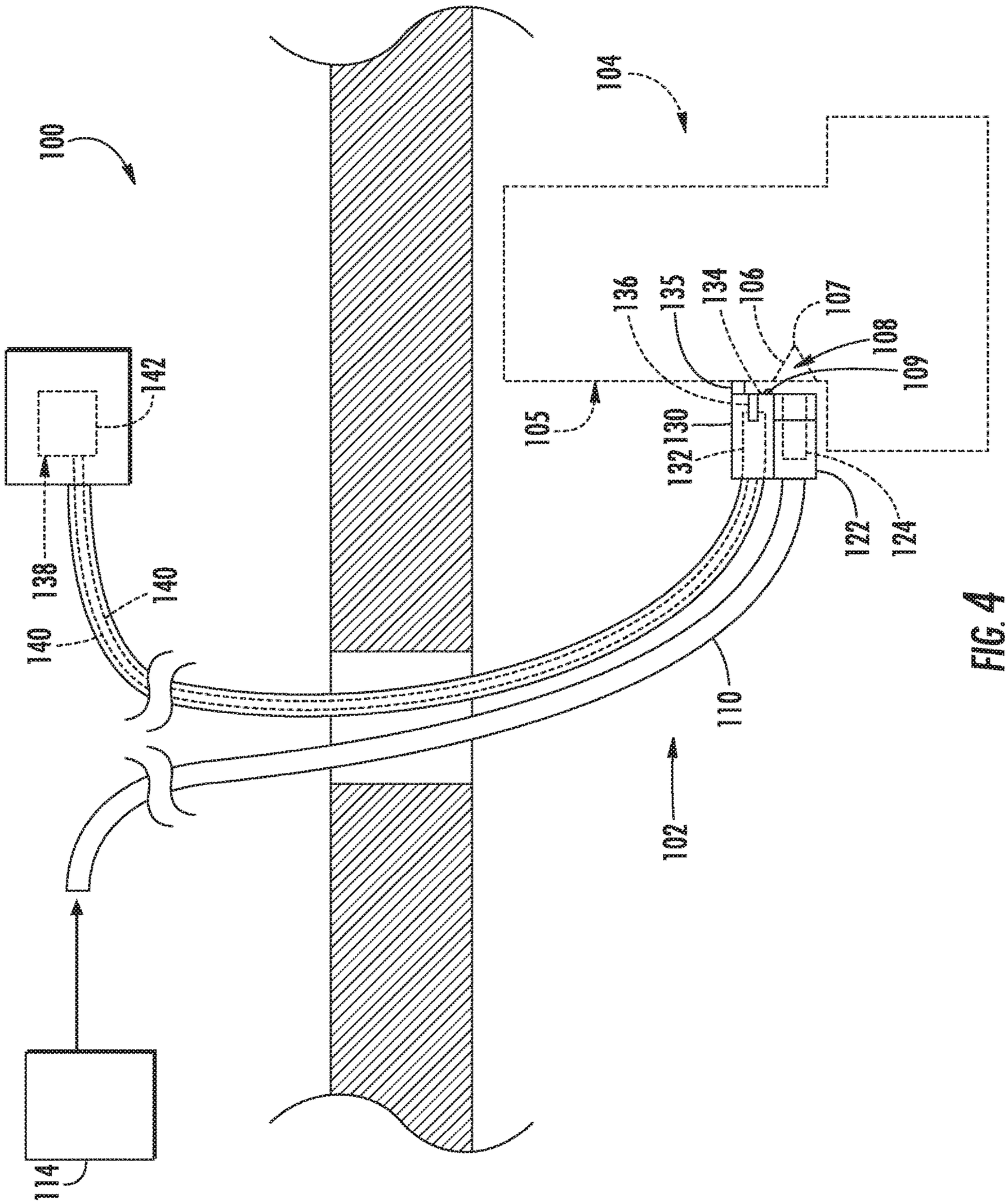

100
104
105
102
110
106
107
108
109
122
124
130
132
134
135
136
138
140
140
142
144

FIG. 4

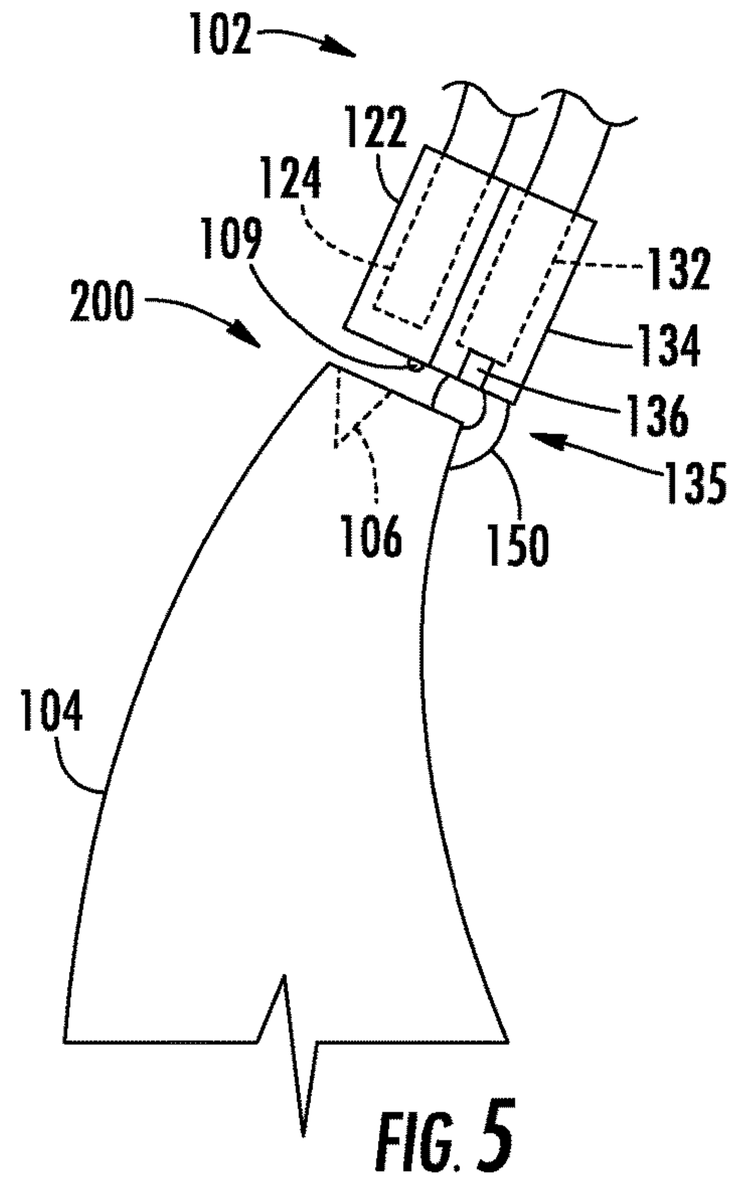

76
102
122
124
109
200
132
134
136
135
106
150
104

FIG. 5

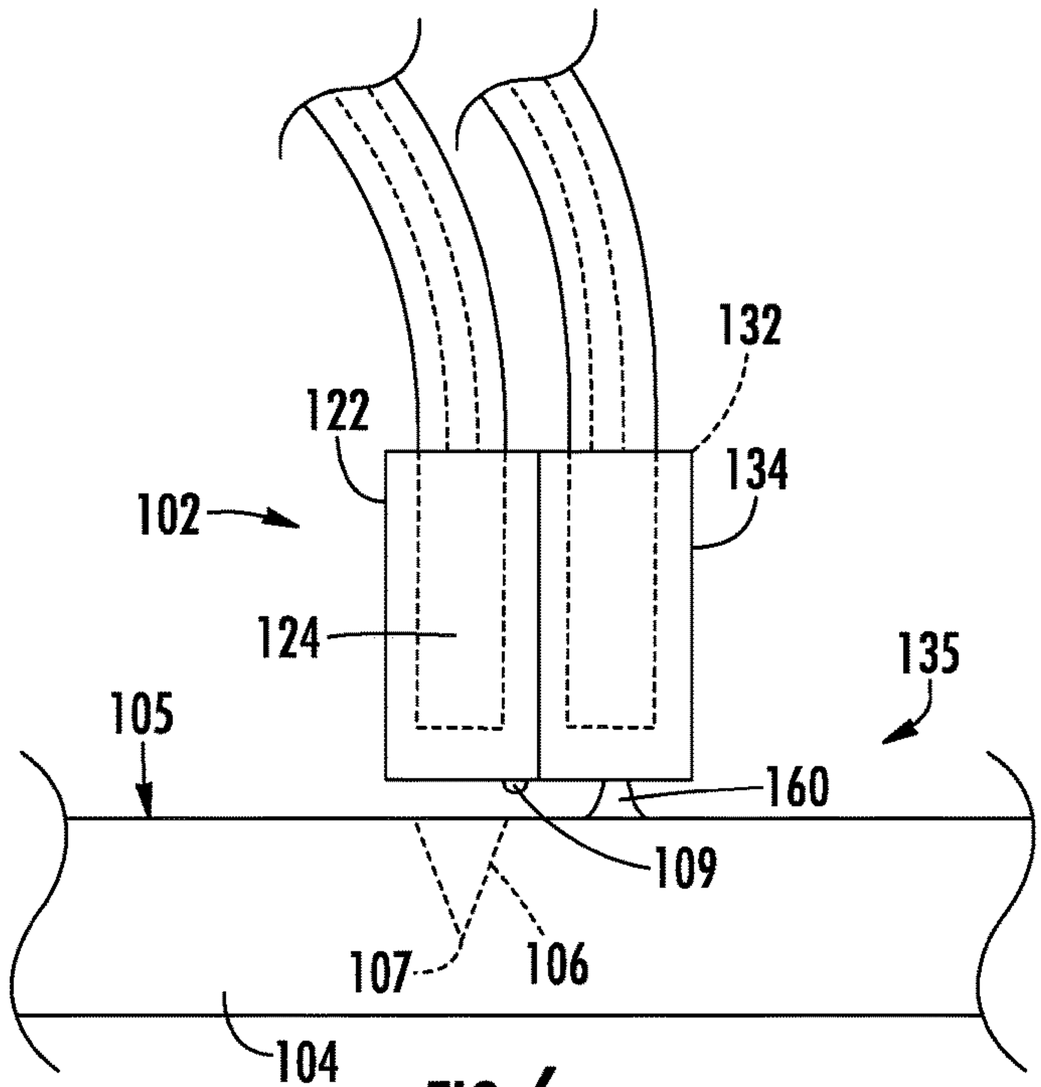

132
122
134
102
124
135
105
160
109
107
106
104

FIG. 6

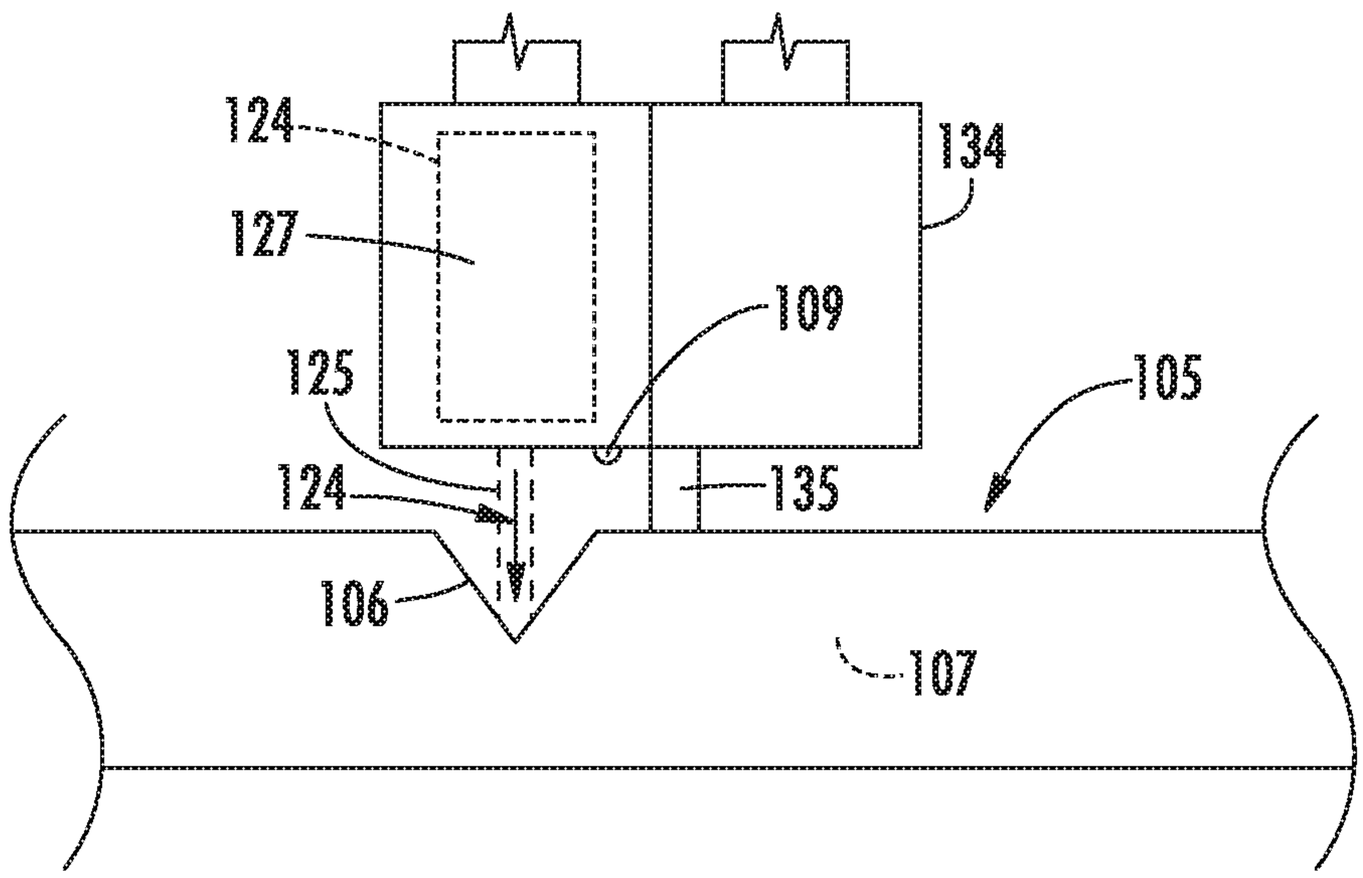

124
127
134
109
125
105
124
135
106
107

FIG. 7

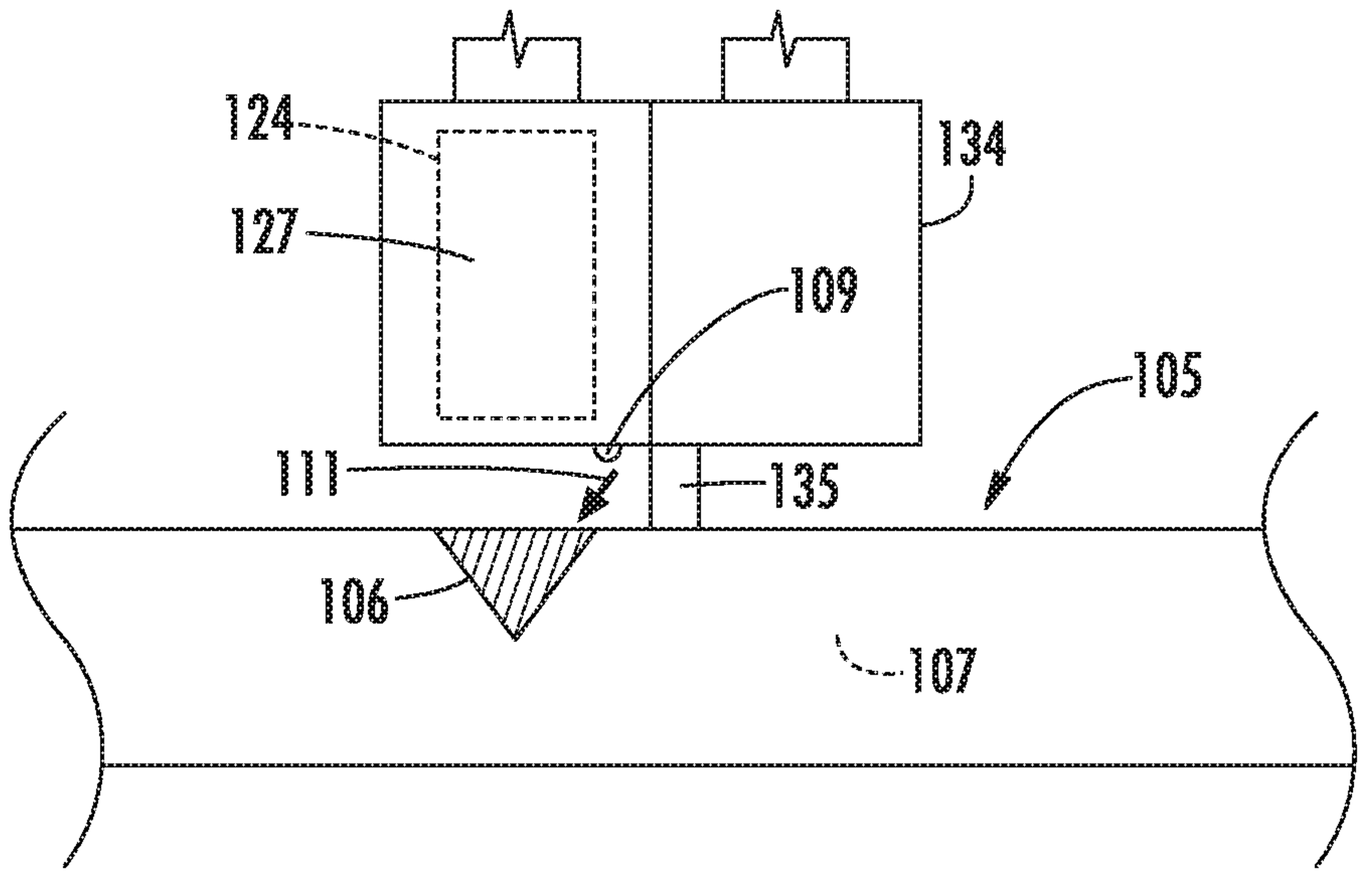

124
127
134
109
105
111
135
106
107

FIG. 8

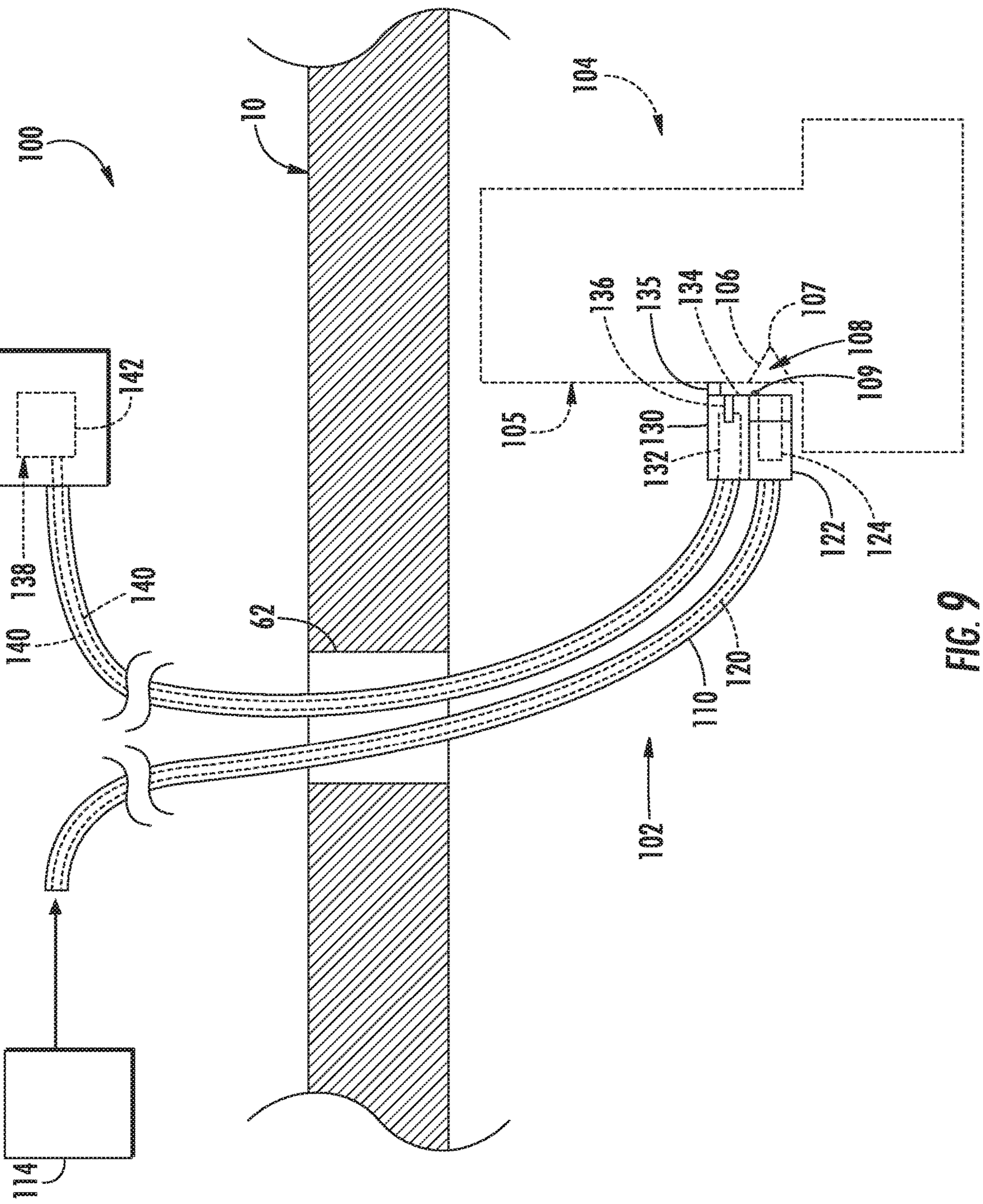

100
10
104
136
135
134
106
107
108
109
105
130
132
142
138
140
140
62
122
124
110
120
102
114

FIG. 9

IN SITU GAS TURBINE PREVENTION OF CRACK GROWTH PROGRESSION VIA LASER WELDING

FIELD OF THE INVENTION

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

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.

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.

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

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.

Methods are generally provided for material build-up on a tip of a blade of a gas turbine engine. In one embodiment, the method includes inserting a material supply adjacent to the tip and directing a laser onto the interface of the material supply and the tip such that the material supply melts and attaches to the tip.

Methods are also 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 within a surface of the component; supplying a new material to the fillable area to fill the defect; and directing a laser to the new material within the fillable area to fuse the new material to the component within the defect.

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

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:

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;

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;

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;

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;

FIG. **5** illustrates a partial view of an exemplary repair tool temporarily secured to a tip of an airfoil in order to perform an in situ repair thereon;

FIG. **6** illustrates a partial view of another exemplary repair tool temporarily secured to a tip of an airfoil in order to perform an in situ repair thereon;

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 supply powder particles within a defect for in situ repair;

FIG. **8** illustrates a partial view of the embodiment of the repair tool of FIG. **8** using a laser to weld the powder particles within the defect; and

FIG. **9** 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.

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

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.

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.

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.

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 new material (solid or liquid) and direct a laser to heat to fuse new material within the crack to repair the defect. For example, if the new material is supplied as a solid, then the laser can heat and weld the material within the crack to repair the defect.

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.

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.

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.

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.

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**.

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.

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**.

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**.

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**.

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 **62**A may be defined through the turbine casing/frame to provide access to the first stage of the turbine **28** while a second access port **62**B may be defined through the turbine casing/frame to provide access to the second stage of the turbine **28**.

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.

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.

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 **62***a*, **62***b*, **62***c*, **62***d* are illustrated that provide access to four successive stages, respectively, of the compressor **24**.

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.

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**.

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).

In one embodiment, the attachment mechanism **135** can be a tripod grip for a component **104** having a known shape and/or size. FIG. **5** illustrates a partial view of an exemplary repair tool temporarily secured to a tip of an airfoil in order to perform an in situ repair thereon such that the material supply is positioned between the tip **200** of the blade and a shroud **76**. 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 bought 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).

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**.

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.

As shown in FIGS. **5-9**, the working head **122** includes a work mechanism **124** configured for addressing the defect **106**. In one embodiment, the new material can be supplied from a location exterior to the engine to the internal location of the defect **106** to allow the fillable volume **108** defined by the defect **106** to be filled with the new material. FIG. **7** shows the repair tool **102** configured to supply high velocity powder particles **125** from the exterior of the engine into the fillable volume **108** of the defect **106**. Upon impacting a surface of the defect **106**, the high velocity particles **125** may plastically deform and adhere to the surface, thereby filling-in the fillable volume **108** and repairing the defect **106**. For example, the particles can impact the surface within the defect **106** at a speed of about 150 meters per second (m/s) to about 900 m/s.

The average size of the powder particles **125** can vary depending on their composition, gun type, nozzle type, gases used, etc. In most embodiments, the particle size and distribution can be about 25 μm to about 150 μm (e.g., about 35 μm to about 75 μm (i.e., 400 to about 200 mesh)). In certain embodiments, no more than about five percent of the particles are larger than about 75 μm (200 mesh) and no more than about fifteen percent of the particles being smaller than about 35 μm (400 mesh).

The powder particles **125** 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 powder particles **125**, thereby repairing the defect **106**. In several embodiments, the repair tool **102** may be configured to supply the powder particles **125** within the interior of the gas turbine engine **10**. For example, the powder particles **125** 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 powder particles **125** to be injected or otherwise directed into the fillable volume **108** defined by the defect **106**.

The particles **125** may be supplied via a carrier fluid (e.g., a carrier gas) that is inert to the coating deposition.

The powder particles **125** may then be heated to fuse the material within the fillable volume **108** to repair the defect **106**. For example, the repair tool **102** may include a laser **109** at its working end to heat the powder particles **125** during and/or after introduction into the defect **106**, thereby filling in the fillable volume **108** to bond the material within the defect **106**. For example, the working head **122** may include a laser **109** directed into the flow path of the powder particles **125** (e.g., into the defect **106**) to locally heat the base of the defect **106**, before, during, and/or after deposition of the new material (e.g., the powder particles **125**). For example, the laser **109** may direct electromagnetic radiation **111** having a wavelength of about 0.5 μm to about 25 μm into the defect **106** in the surface **105** of the component **104** at a power level of about 0.02 W to about 75 kW. The electromagnetic radiation **111** can heat a precision weld 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**. FIG. **8** shows the laser **109** directing electromagnetic radiation **111** onto the new material within the defect **106**.

For example, the base **107** may be locally heated using the laser **109** 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.

In one embodiment, 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**. As shown in the illustrated embodiment of FIG. **9**, the repair tool **102** may include a high temperature conduit **110** for transporting the metal particles from outside the engine **10** to the location of the defect **106**. Specifically, as shown in FIG. **9**, 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 particles **125** into the fillable volume **108**. Additionally, the material supply end **114** of the tool **102** may generally be configured to receive particles **125** from a particle source. For example, as shown in FIG. **9**, particles **125** contained within a chamber (or other suitable powder particle source) located exterior to the gas turbine engine **10** may be supplied to the material supply end **114** of the tool **102**. The particles **125** 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 metal particles to be delivered to the location of the defect **106**.

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.

In general, the heating element(s) **120** may be configured to generate heat within the high temperature conduit **110** as powder particles **125** is being supplied through the conduit **110** so as to allow for particle flow at the desired rate and speed. 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 powder particles **125** at its desired delivery temperature. In one embodiment, the particles **125** are delivered to the defect **106** at a temperature within 25% of its melting point (e.g., within 10% of its melting point).

It should be appreciated that the powder particles **125** may be composed of any suitable metal material. For example, in one embodiment, the powder particles **125** may correspond to the parent metal material of the internal component **104** being repaired. In other embodiments, the powder particles **125** may correspond to any other metal material that is suitable for use as a repair material within a gas turbine engine **10**.

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.

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**.

As shown in FIGS. **4** and **9**, 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**.

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.

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. **1-9**. 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.

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**.

Moreover, the method may include performing precision repair work (e.g., supplying powder particles, heating, etc.) using the repair tool by temporarily attaching the tip end of the repair tool to the surface of the component.

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 in situ material build-up on a tip of a blade of a gas turbine engine, the method comprising:

inserting a working head through an access port of an assembled core gas turbine engine, wherein the working head comprises a laser and a tripod grip;

inserting a material supply adjacent to the tip of the blade-while the blade is an installed internal component of the gas turbine engine, such that the material supply is positioned radially inwardly of a shroud of the gas turbine engine;

positioning the working head adjacent to an interface defined between the material supply and the tip of the blade;

temporarily securing the working head adjacent to the blade with the tripod grip, wherein the tip of the blade has a known shape and size, and wherein the tripod grip comprises a plurality of grip arms oriented to grip the tip of the blade having the known shape and size within the gas turbine engine; and

directing a radiation from the laser onto the interface of the material supply and the tip of the blade such that the material supply melts and attaches to the tip of the blade.

2. The method of claim **1**, wherein the radiation has a wavelength of about 0.5 μm to about 25 μm.

3. The method of claim **1**, wherein the laser emits electromagnetic radiation at a power level of about 0.02 W to about 75 kW.

4. The method of claim **1**, wherein the working head is attached to a robotic arm.

5. The method of claim **4**, wherein the interface precision is 0.05 inches or less.

6. The method of claim **4**, wherein an optical probe is attached to the robotic arm.

7. The method of claim **1**, further comprising:

inserting an optical probe through the access port or an adjacent access port of the gas turbine engine, the optical probe being configured to provide a view of the interface within the gas turbine engine.

8. A method of in situ material build-up on a tip of a blade of a gas turbine engine, the method comprising:

inserting a working head through an access port of an assembled core gas turbine engine, wherein the working head comprises a laser, a hot-melt adhesive, and a heating component for melting the hot melt adhesive;

inserting a material supply adjacent to the tip of the blade-while the blade is an installed internal component of the gas turbine engine, such that the material supply is positioned radially inwardly of a shroud of the gas turbine engine;

positioning the working head adjacent to an interface defined between the material supply and the tip of the blade;

temporarily securing the working head adjacent to the blade with the hot-melt adhesive;

directing a radiation from the laser onto the interface of the material supply and the tip of the blade such that the material supply melts and attaches to the tip of the blade; and

melting the hot melt adhesive with the heating component to remove the working head from the blade.

9. The method of claim **8**, wherein the radiation has a wavelength of about 0.5 μm to about 25 μm.

10. The method of claim **8**, wherein the radiation is electromagnetic radiation emitted from the laser at a power level of about 0.02 W to about 75 kW.

11. The method of claim **8**, wherein the working head is attached to a robotic arm.

12. The method of claim **11**, wherein the interface precision is 0.05 inches or less.

13. The method of claim **11**, wherein an optical probe is attached to the robotic arm.

14. The method of claim **8**, further comprising:

inserting an optical probe through the access port or an adjacent access port of the gas turbine engine, the optical probe being configured to provide a view of the interface within the gas turbine engine.

* * * * *