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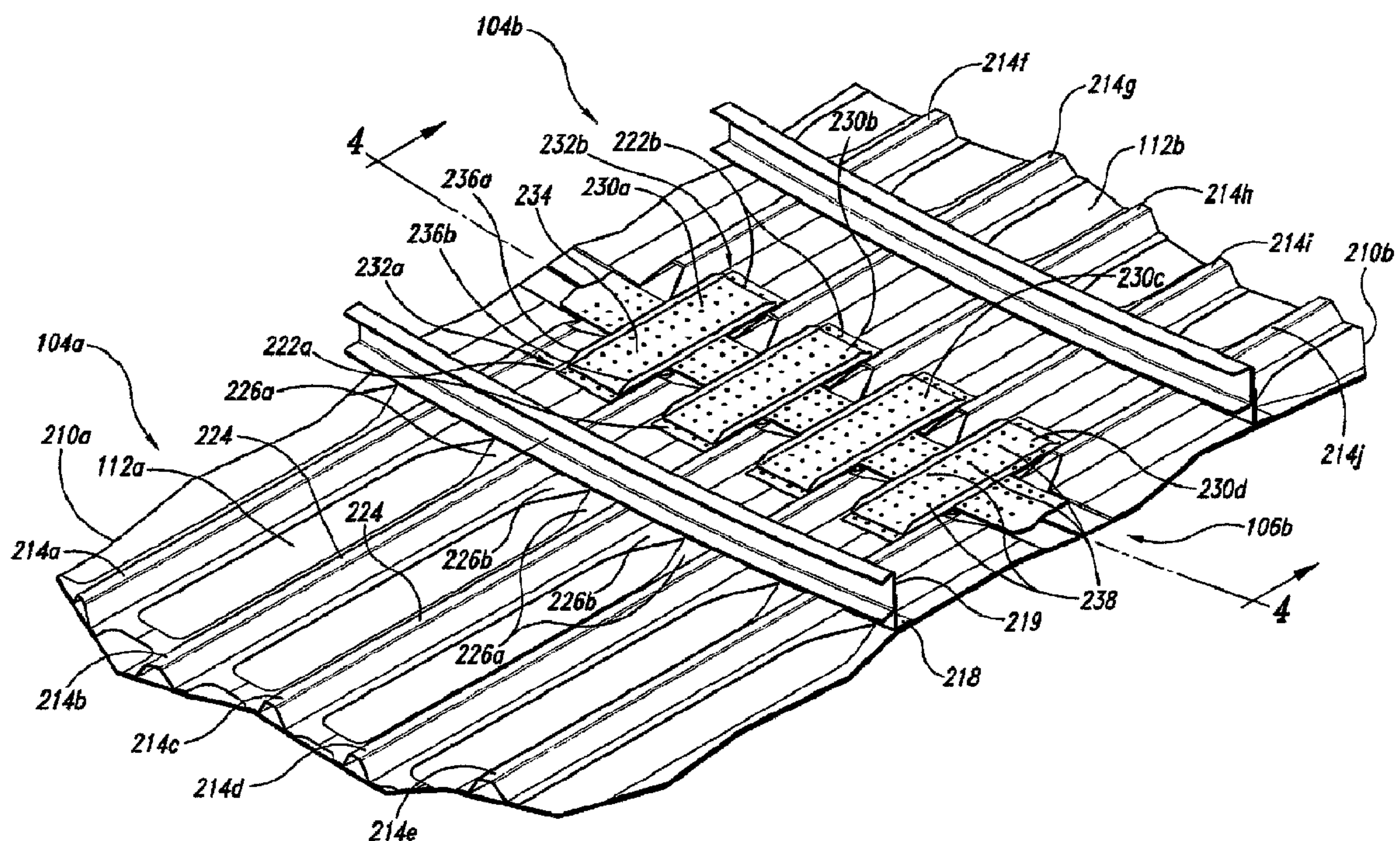
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(57) Abrégé/Abstract:

Structures and methods for joining composite fuselage sections and other panel assemblies together are disclosed herein. In one embodiment, a shell structure configured in accordance with the present invention includes a first panel portion (210a) positioned adjacent to a second panel portion (210b). The first panel portion (210a) can include a first stiffener (214a-214e) attached to a first composite skin (112a), and the second panel portion (210b) can include a second stiffener (214i-214j) attached to a second composite skin (112b). The shell structure can further include a fitting (230) extending across a first edge region of the first panel portion (210a) and a second edge region of the second panel portion (210b). A first end portion (232a) of the fitting (230) can be attached to the first stiffener (214a-214e) and the first composite skin (112a), and a second end portion (232b) of the fitting (230) can be attached to a second stiffener (214i-214j) and a second composite skin (112b), to join the first panel portion (210a) to the second panel portion (210b).

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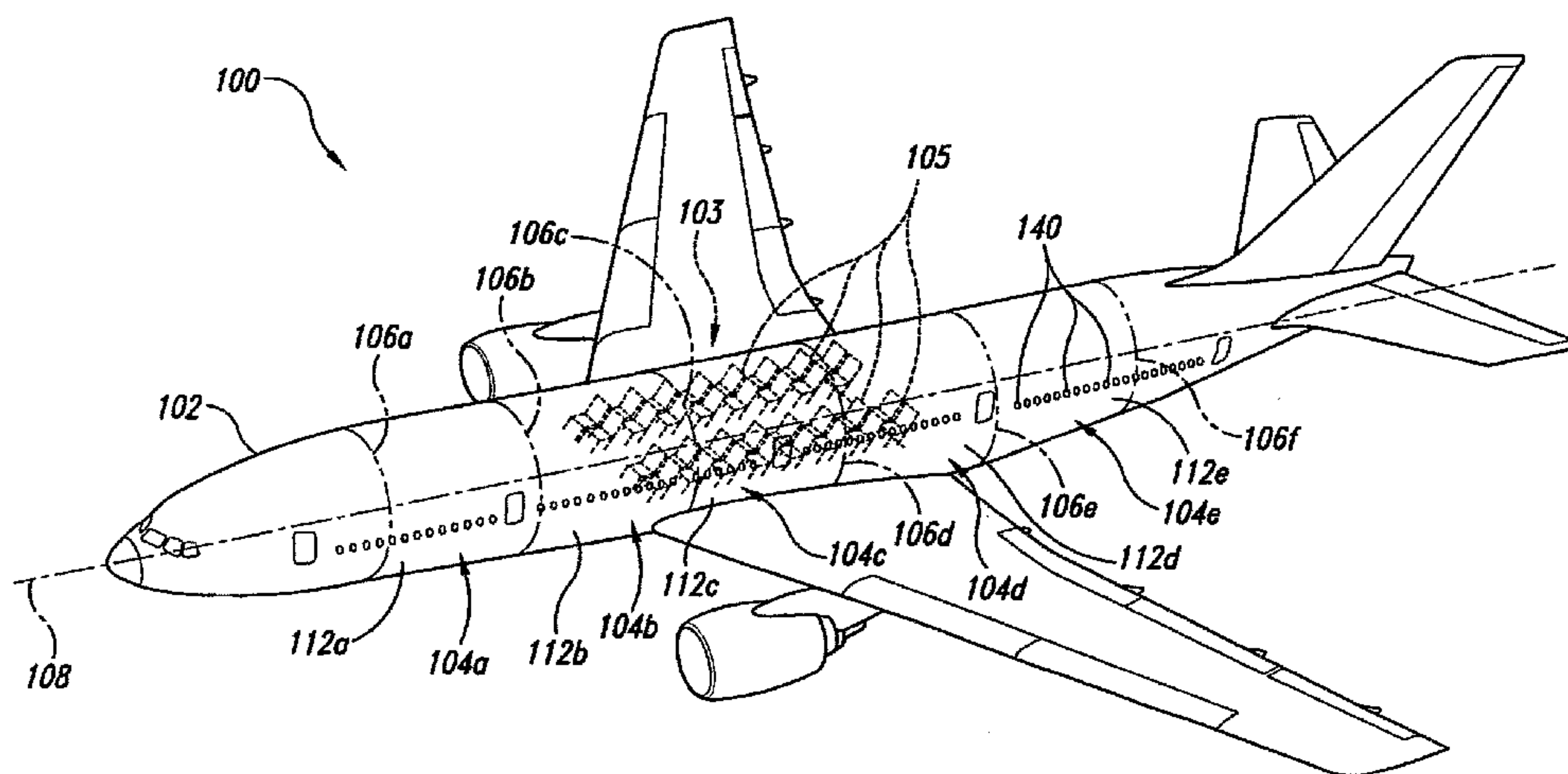
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(54) Title: SPLICE JOINTS FOR COMPOSITE AIRCRAFT FUSELAGES AND OTHER STRUCTURES



(57) Abstract: Structures and methods for joining composite fuselage sections and other panel assemblies together are disclosed herein. In one embodiment, a shell structure configured in accordance with the present invention includes a first panel portion (210a) positioned adjacent to a second panel portion (210b). The first panel portion (210a) can include a first stiffener (214a-214e) attached to a first composite skin (112a), and the second panel portion (210b) can include a second stiffener (214i-214j) attached to a second composite skin (112b). The shell structure can further include a fitting (230) extending across a first edge region of the first panel portion (210a) and a second edge region of the second panel portion (210b). A first end portion (232a) of the fitting (230) can be attached to the first stiffener (214a-214e) and the first composite skin (112a), and a second end portion (232b) of the fitting (230) can be attached to a second stiffener (214i-214j) and a second composite skin (112b), to join the first panel portion (210a) to the second panel portion (210b).

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SPLICE JOINTS FOR COMPOSITE AIRCRAFT FUSELAGES AND OTHER STRUCTURES

TECHNICAL FIELD

The following disclosure relates generally to shell structures and, more particularly,
5 to splice joints for joining composite fuselage sections and other shell structures together.

BACKGROUND

The primary structural elements of large passenger jets and other large aircraft are typically made from metal. Fuselage shells for such aircraft, for example, are typically manufactured from high-strength aluminum alloys or similar metals. In an effort to
10 increase performance, however, many aircraft manufacturers are turning to fiber-reinforced resin materials (i.e., "composite" materials) that have relatively high strength-to-weight ratios. Conventional composite materials typically include glass, carbon, or polyaramide fibers in a matrix of epoxy or another type of resin. The use of such materials for primary structures has mostly been limited to smaller aircraft, such as fighter aircraft, high-
15 performance private aircraft, and business jets.

One known method for manufacturing business jet airframes with composite materials is employed by the Raytheon Aircraft Company of Wichita, Kansas, to manufacture the Premier I and Hawker Horizon business jets. This method involves wrapping carbon fibers around a rotating mandrel with an automated fiber placement
20 system. The mandrel provides the basic shape of a longitudinal fuselage section. The carbon fibers are preimpregnated with a thermoset epoxy resin, and are applied over the rotating mandrel in multiple plies to form an interior skin of the fuselage section. The interior skin is then covered with a layer of honeycomb core. The fiber placement system then applies additional plies of preimpregnated carbon fibers over the honeycomb core to
25 form an exterior skin that results in a composite sandwich structure.

The Premier I fuselage includes two 360-degree sections formed in the foregoing manner. The Hawker Horizon fuselage includes three such sections formed in this manner. The two 70-inch diameter sections of the Premier I fuselage are riveted and then bonded together at a circumferential splice joint to form the complete fuselage structure.

5 The much larger Hawker Horizon fuselage, with an 84-inch diameter, uses aluminum splice plates at two circumferential joints to join the three fuselage sections together into a complete structure.

To precisely install the aluminum splice plates on the Hawker Horizon fuselage, Raytheon created a special, automated splice machine. This machine aligns the three
10 fuselage sections using a computer-aided laser alignment system, and then drills attachment holes through the aluminum splice plates and the underlying sandwich structure. The machine then probes each hole for size quality and records statistical process control data on each hole. The drill heads also apply sealant and install hi-shear fasteners in approximately 1,800 places along each of the splice joints. (See Raytheon
15 Aircraft news release at <http://www.beechcraft.de/presse/2000/100900b.htm> entitled "RAYTHEON AIRCRAFT'S HAWKER HORIZON REACHES FUSELAGE MILESTONE," October 9, 2000).

SUMMARY

The present invention is directed generally toward structures and methods for
20 joining composite fuselage sections and other panel assemblies together. A shell structure configured in accordance with one aspect of the invention includes a first panel portion positioned adjacent to a second panel portion. The first panel portion can include a first stiffener attached to a first composite skin, and the second panel portion can include a second stiffener attached to a second composite skin. The shell structure can further
25 include a fitting extending across a first edge region of the first panel portion and a second edge region of the second panel portion. A first end portion of the fitting can be attached to the first stiffener and the first composite skin, and a second end portion of the fitting can be attached to the second stiffener and the second composite skin, to join the first panel portion to the second panel portion.

A method for manufacturing a shell structure in accordance with another aspect of the invention includes attaching at least a first stiffener to a first composite skin, and attaching at least a second stiffener to a second composite skin. The method can further include positioning the first composite skin in edgewise alignment with the second composite skin, attaching a first end of a fitting to the first stiffener and the first composite skin, and attaching a second end of the fitting to the second stiffener and the second composite skin. In one embodiment, the method can additionally include attaching a strap to a first edge region of the first composite skin and an adjacent second edge region of the second composite skin to splice the first and second composite skins together before the fitting is attached.

BRIEF DESCRIPTION OF THE DRAWINGS

Figure 1 is an isometric view of an aircraft having a composite fuselage configured in accordance with an embodiment of the invention.

Figures 2A-2C together illustrate a method of joining a first fuselage barrel section to a second fuselage barrel section in accordance with an embodiment of the invention.

Figures 3A-3C together illustrate a method of joining the first fuselage barrel section to the second fuselage barrel section in the vicinity of a window cutout, in accordance with another embodiment of the invention.

Figure 4 is a cross-sectional end view of the splice joint of Figure 2C taken substantially along line 4-4 in Figure 2C.

DETAILED DESCRIPTION

The following disclosure describes structures and methods for joining composite fuselage sections and other panel assemblies together. Certain details are set forth in the following description and in Figures 1-3C to provide a thorough understanding of various embodiments of the invention. Other details describing well-known structures and systems often associated with composite parts and related assembly techniques are not set forth in the following disclosure to avoid unnecessarily obscuring the description of the various embodiments of the invention.

Many of the details, dimensions, angles, and other features shown in the Figures are merely illustrative of particular embodiments of the invention. Accordingly, other embodiments can have other details, dimensions, angles, and features without departing from the spirit or scope of the present invention. In addition, further embodiments of the invention can be practiced without several of the details described below.

In the Figures, identical reference numbers identify identical or at least generally similar elements. To facilitate the discussion of any particular element, the most significant digit or digits of any reference number refer to the Figure in which that element is first introduced. For example, element 106 is first introduced and discussed with reference to Figure 1.

Figure 1 is an isometric view of an aircraft 100 having a composite fuselage 102 configured in accordance with an embodiment of the invention. In one aspect of this embodiment, the fuselage 102 includes a plurality of composite barrel sections 104 (identified individually as barrel sections 104a-e) joined together by a plurality of corresponding splice joints 106 (identified individually as splice joints 106a-f). Each of the barrel sections 104 includes a composite skin 112 (identified individually as composite skins 112a-112e) extending 360 degrees around a longitudinal axis 108. In the illustrated embodiment, each of the composite skins 112 can have a cross-sectional width of at least about 10 feet, such as about 15 feet to about 35 feet. In one embodiment, for example, the composite skins 112 can have a cross-sectional width of about 18 feet. Throughout this disclosure, the term "barrel section" is used for convenience to refer to any shell structure extending 360 degrees around an axis. Accordingly, the term is not limited to cylindrical structures or structures having barrel shapes, but can include structures having circular, elliptical, oval, egg-shaped, rectilinear, tapered, or other cross-sectional shapes. In addition, in one embodiment, the barrel sections 104 can be "one-piece" barrel sections in which the composite skins 112 are "one-piece" skins extending continuously for 360 degrees around the axis. In other embodiments, however, the skins 112 can be formed from two or more skin segments spliced or otherwise joined together to form the full 360-degree barrel section.

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The fuselage 102 can further include a passenger cabin 103 configured to hold a plurality of passenger seats 105 ranging in number from about 50 to about 700 seats. For example, in the illustrated embodiment, the passenger cabin 103 can hold from about 150 to about 600 passenger seats 105. In other embodiments, the passenger cabin 103 can be configured to hold more or fewer passenger seats without departing from the spirit or scope of the present disclosure. Each of the barrel sections 104 can include a plurality of window cutouts 140 to provide the passengers seated in the passenger cabin 103 with views out of the aircraft 100.

Figures 2A-2C together illustrate a method of joining the first barrel section 104a to the second barrel section 104b in accordance with an embodiment of the invention. Referring first to Figure 2A, this view is a partially exploded, enlarged isometric view looking outwardly at a portion of the second splice joint 106b from within the fuselage 102 (Figure 1). The portion of the first barrel section 104a illustrated in Figure 2A includes a first panel portion 210a. The portion of the second barrel section 104b illustrated in Figure 2A includes a second panel portion 210b positioned in edgewise alignment with the first panel portion 210a. In one embodiment, the panel portions 210 can be at least generally similar in structure and function to the panel assemblies described in detail in co-owned U.S. Patent No. 7,527,222, granted May 5, 2009, and U.S. Patent No. 7,316,629, granted November 14, 2006. For example, the first panel portion 210a can include a plurality of stiffeners 214 (identified individually as stiffeners 214a-214e) attached to the first skin 112a. Each of the stiffeners 214 can include a raised portion 224 projecting away from the first skin 112a, and a plurality of flange portions (identified individually as first flange portions 226a and second flange portions 226b) attached directly to the first skin 112a. In the illustrated embodiment, the stiffeners 214 have hat-shaped cross-sections. In other embodiments, however, the stiffeners 214 can have other cross-sectional shapes, including "L" shapes, "C" shapes, inverted "T" shapes, "I" shapes, etc. In yet other embodiments, the panel portions 210 can include other features, including those disclosed in co-owned U.S. Patent No. 7,159,822 granted January 9, 2007.

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The stiffeners 214 can be positioned on the first skin 112a so that the first flange portions 226a of one stiffener 214 are aligned with the corresponding second flange portions 226b of an adjacent stiffener 214. By aligning the flange portions 226 in the foregoing manner, the flange portions 226 can form a plurality of at least approximately
5 continuous support surfaces 228 (identified individually as support surfaces 228a and 228b) extending between the raised portions 224 of the stiffeners 214.

The first panel portion 210a can further include part of a support member or frame 216a. In the illustrated embodiment, the frame 216a is a two-piece frame that includes a first frame section 218 and a second frame section 219. The first frame section 218 can be
10 attached directly to the support surfaces 228 as described in detail in U.S. Patent No. 7,527,222. In other embodiments, the first frame section 218 can be attached to the first panel portion 210a using other methods. In still further embodiments, the first panel portion 210a can include parts of other frames composed of more or fewer frame sections. Alternatively, the frame 216a can be omitted.

15 The second panel portion 210b can be at least generally similar in structure and function to the first panel portion 210a described above. Accordingly, the second panel portion 210b can include a plurality of stiffeners 214 (identified individually as stiffeners 214f-j) attached to the second skin 112b. The second panel portion 210b can further include a second frame 216b that is attached to flange portions of the stiffeners 214 in the
20 manner described above for the first panel portion 210a.

Referring next to Figure 2B, an elongate strap 220 is attached to a first edge region 213a of the first skin 112a and an adjacent second edge region 213b of the second skin 112b to splice the first skin 112a to the second skin 112b. The strap 220 is attached to the inner side of the respective skins 112 to maintain a smooth, aerodynamic surface on the
25 exterior of the fuselage 102 (Figure 1). In one embodiment, the strap 220 can include composite materials, such as graphite-epoxy or similar material. In other embodiments, the strap 220 can include other materials, including metallic materials such as aluminum, titanium, steel, etc. The strap 220 can be attached to the skins 112 with a plurality of fasteners 221 extending through the strap 220 and the skins 112. In other embodiments,
30 the strap 220 can be bonded to the skins 112, or bonded and fastened to the skins 112.

Further, in embodiment, the strap 220 can extend continuously, or at least approximately continuously, around the splice joint 106b. In other embodiments, the strap 220 can be segmented around the splice joint 106b. For example, in one embodiment, the splice joint 106b can include six segments of the strap 220. In other embodiments, more (e.g., eight) or less segments of the strap 220 can be used.

In the illustrated embodiment, the strap 220 can be at least approximately as thick as the skins 112, but thicker than the adjacent flange portions 226 of the stiffeners 214. To avoid a step between adjacent surfaces, shim pads or fillers 222 (identified individually as first fillers 222a and second fillers 222b) are positioned on the flange portions 226 adjacent to the strap 220. In one embodiment, the fillers 222 can include composite materials, including graphite-epoxy or similar materials. In other embodiments, the fillers 222 can include aluminum and other metals. In yet other embodiments, the strap 220, the skins 112, and/or the flange portions 226 can have other relative thicknesses and/or the fillers 222 can be omitted.

Referring next to Figure 2C, a plurality of fittings 230 are positioned on the strap 220 and extend across the splice joint 106b between the stiffeners 214. A first end portion 232a of each fitting 230 overlays the corresponding first filler 222a and the flange portions 226 of the adjacent stiffeners 214. Similarly, a second end portion 232b of each fitting 230 overlays the corresponding second filler 222b and the flange portions 226 of the adjacent stiffeners 214. In the illustrated embodiment, each of the fittings 230 has a channel or "U-shaped" cross section that includes a base portion 234, a first upstanding edge portion 236a positioned toward a first side of the base portion 234, and a second upstanding edge portion 236b positioned toward a second side of the base portion 234. In other embodiments, the fittings 230 can have other cross-sectional shapes, including "C" shapes, "L" shapes, inverted "P" shapes, and flat shapes, to name a few. A plurality of fasteners 238 extending through the fittings 230 and the underlying structures (i.e., the fillers 222, the flange portions 226, the strap 220, and the skins 112) attach the fittings 230 to the underlying structures to form a structural load path across the splice joint 106b.

The fittings 230, the stiffeners 214, the strap 220, and the skins 112 can include composite materials, including graphite-epoxy and/or other suitable composite materials.

For example, in one embodiment, the skins 112 can be manufactured with toughened epoxy resin and carbon fibers, e.g., intermediate carbon fibers from Toray Composites America, Inc. of 19002 50th Avenue East, Tacoma, WA 98446. In this embodiment, the skins 112 can include fiber tape pre-impregnated with resin (i.e., "prepreg") and outer plies of prepreg fabric. In another embodiment, the strap 220 and the fittings 230 can also be manufactured from epoxy resin and carbon fibers. The skins 112, the strap 220, and the fittings 230 can have quasi-isotropic lay-ups, i.e., lay-ups having an equal (or approximately equal) number of plies with 0, +45, -45, and 90 degree orientations. The stiffeners 214 can have axial-dominated fiber orientations. In other embodiments, the skins 112, the strap 220, the fittings 230, and the stiffeners 214 can have other fiber orientations.

One advantage of using composite materials instead of metals is that the fittings 230 and the underlying structures (e.g., the skins 112 and the stiffeners 214) will have at least generally similar coefficients of thermal expansion. As a result, temperature fluctuations experienced during operation of the aircraft 100 (Figure 1) will not cause disparate thermal expansion between the fittings 230 and the underlying structures, and hence will not induce significant stresses in the splice joint 106b. In other embodiments, however, the fittings 230 can include metal materials such as aluminum, titanium, steel, etc. The use of metals may be appropriate in those situations in which the aircraft is not expected to experience wide temperature fluctuations during operation.

In addition to composites and metal materials, in yet other embodiments, the skins 112, the strap 220, the fittings 230, and the stiffeners 214, and combinations thereof, can include other materials, including hybrid materials such as fiber/metal laminates. Such laminates include fiberglass/aluminum laminates and titanium reinforced graphite laminates (Ti/Gr). One hybrid laminate that includes alternating layers of aluminum and fiberglass is referred to as "GLARE™." This laminate may offer better fatigue properties than conventional aluminum. A Ti/Gr laminate may offer weight advantages over conventional aluminum or graphite-epoxy, but this laminate may also be more expensive.

One feature of the splice joint 106b illustrated in Figure 2C is that the fittings 230 overlap the strap 220. One advantage of this feature is that it provides a fail-safe, redundant load path in the unlikely event that a crack or other structural flaw propagates

through a portion of the strap 220. In such an event, the fittings 230 alone can carry the structural load across the splice joint 106b. In addition, the fittings 230 also provide a redundant load path across the splice joint 106b from where the stiffeners 214 terminate. Further, if a segmented strap 220 is used, then the fittings 230 can also be used as splice
5 plates for adjacent strap segments. Another feature of the splice joint 106b is that the ends of the stiffeners 214 are left open. One advantage of this feature is that it enables moisture caused by condensation and other sources to escape the stiffeners 214 for sufficient drainage.

One feature of the fittings 230 of the illustrated embodiment are the first and second
10 upstanding edge portions 236a and 236b. The upstanding edge portions 236 can add stiffness to the fittings 230, and can be positioned proximate to the raised portions 224 of the stiffeners 214. One advantage of this configuration is that it can increase the stability of the splice joint 106b, especially under compression loads.

Yet another feature of the illustrated embodiment is that the raised portions 224 of
15 opposing stiffeners 214 are not spliced together across the splice joint 106b. One advantage of this feature is that it makes the fittings 230 relatively easy to install because the raised portions 224 do not have to be in perfect alignment. While the raised portions 224 could be spliced together in other embodiments, doing so would most likely add time and cost to manufacturing of the splice joint because of the various alignment and
20 shimming considerations involved. Further, splicing the raised portions 224 together could close off the ends of the stiffeners 214, thereby preventing sufficient water drainage and preventing visual inspection of any fasteners positioned under the raised portions 224.

Although the splice joint 106b of the illustrated embodiment is built up from a number of separate parts (e.g., the strap 220 and the fittings 230), in other embodiments,
25 two or more of these parts can be integrated into a single part that performs the function and/or has the features of the two or more parts. For example, in one other embodiment, the splice joint 106b can be at least partially formed by a single part that integrates the features of the strap 220 and the fittings 230. In another embodiment, the splice joint 106b can include a single part that integrates the features of the strap 220 and the adjacent
30 fillers 222. Although integrating parts may have the advantages of reducing part count

and/or increasing strength, using separate parts may have the advantage of simplifying part construction and/or simplifying installation procedures.

Figures 3A-3C together illustrate a method of joining the first barrel section 104a to the second barrel section 104b in the vicinity of one of the window cutouts 140, in accordance with an embodiment of the invention. Referring first to Figure 3A, this view is a partially exploded, enlarged isometric view looking outwardly at a portion of the second splice joint 106b around the window cutout 140. The portion of the first barrel section 104a illustrated in Figure 3A includes a third panel portion 310a. The portion of the second barrel section 104b illustrated in Figure 3A includes a fourth panel portion 310b positioned in edgewise alignment with the third panel portion 310a. The panel portions 310 can be at least generally similar in structure and function to the panel portions 210 described in detail above with reference to Figures 2A-2C. For example, the third panel portion 310a can include a plurality of stiffeners 214 (identified individually as stiffeners 214k-214m) attached to the first skin 112a. Similarly, the fourth panel portion 310b can include a plurality of stiffeners 214 (identified individually as stiffeners 214n-214p) attached to the second skin 112b. In one aspect of the illustrated embodiment, however, the window cutout 140 is formed in a third edge region 313a of the first skin 112a, and in an adjacent fourth edge region 313b of the second skin 112b.

Referring next to Figure 3B, an elongate strap 320 is attached to the third edge region 313a of the first skin 112a and the adjacent fourth edge region 313b of the second skin 112b. With the exception of an aperture 324 that extends through a flared-out portion of the strap 320, the strap 320 can be at least generally similar in structure and function to the strap 220 described above with reference to Figures 2A-2C. For installation, the aperture 324 is aligned with the window cutout 140 and the strap 320 is attached to the skins 112 with a plurality of the fasteners 221. In other embodiments, the strap 320 can be bonded to the skins 112, or bonded and fastened to the skins 112.

One feature of the strap 320 is that the aperture 324 extends completely around the window cutout 140. One advantage of this feature is that the strap 320 acts as a one-piece doubler, thereby providing an efficient load path around the window cutout 140. A

further advantage of this feature is that it reduces part count by combining the window doubler feature with the splice strap feature in a single, integrated part.

In the illustrated embodiment, the strap 320 is thicker than the adjacent flange portions 226 of the stiffeners 214. To avoid a step between adjacent surfaces, the first fillers 222a and the second fillers 222b are positioned on the flange portions 226 adjacent to the strap 320 in those portions of the splice joint 106b positioned away from the window cutout 140. Narrower fillers 322 (identified individually as third fillers 322a and fourth fillers 322b) are positioned on the stiffener flange portions 226 in those areas proximate to the window cutout 140.

Referring next to Figure 3C, a plurality of the fittings 230 extend across the splice joint 106b in the stiffener bays away from the window cutout 140 as described above with reference to Figures 2A-2C. Narrower fittings 330 are attached across the splice joint 106b in similar fashion at opposing ends of the window cutout 140. The narrow fittings 330 of the illustrated embodiment have "L" shaped cross sections. In other embodiments, however, the narrower fittings 330 can have other cross sectional shapes, including "U" shapes, "C" shapes, and flat shapes. A window frame 350 can be fastened or otherwise attached to the strap 320 and any underlying structures around the window cutout 140. In one embodiment, the window frame 350 can be machined or otherwise formed from a high-strength metal material, such as aluminum. In other embodiments, the window frame 350 can include composites and/or other suitable materials.

One feature of the embodiments described above and illustrated in Figures 3A-3C is that the splice joint 106b extends through the middle of the window cutout 140. One advantage of this feature is that it provides design flexibility. For example, this feature allows window patterns and barrel section lengths to be selected irrespective of splice location. Figure 4 is a cross-sectional end view of the splice joint 106b taken substantially along line 4-4 in Figure 2C. This view illustrates that, in this embodiment, the fittings 230 are positioned over the strap 220, and the fasteners 238 extend through the fittings 230, the strap 220, and the skin 112b. This view further illustrates that the fittings 230 are positioned between, but proximate to, respective stiffeners 214.

The subject matter of copending U.S. Patent Application Nos. 10/646,509, entitled "MULTIPLE HEAD AUTOMATED COMPOSITE LAMINATING MACHINE FOR THE FABRICATION OF LARGE BARREL SECTION COMPONENTS," filed August 22, 2003; 10/717,030, entitled "METHOD OF TRANSFERRING LARGE UNCURED COMPOSITE LAMINATES," filed November 18, 2003; 10/646,392, entitled "AUTOMATED COMPOSITE LAY-UP TO AN INTERNAL FUSELAGE MANDREL," filed August 22, 2003; 10/630,594, entitled "COMPOSITE FUSELAGE MACHINE," filed July 28, 2003; 10/646,316, entitled "UNIDIRECTIONAL, MULTI-HEAD FIBER PLACEMENT," filed August 22, 2003; 10/301,949, entitled "PARALLEL CONFIGURATION COMPOSITE MATERIAL FABRICATOR," filed November 22, 2002; 10/799,306, entitled "SYSTEMS AND METHODS ENABLING AUTOMATED RETURN TO AND/OR REPAIR OF DEFECTS WITH A MATERIAL PLACEMENT MACHINE," filed March 12, 2004; 10/726,099, entitled "SYSTEMS AND METHODS FOR DETERMINING DEFECT CHARACTERISTICS OF A COMPOSITE STRUCTURE," filed December 2, 2003; 10/628,691, entitled "SYSTEMS AND METHODS FOR IDENTIFYING FOREIGN OBJECTS AND DEBRIS (FOD) AND DEFECTS DURING FABRICATION OF A COMPOSITE STRUCTURE," filed July 28, 2003; and [Attorney Docket No. 7784-000696], entitled "SYSTEMS AND METHODS FOR USING LIGHT TO INDICATE DEFECT LOCATIONS ON A COMPOSITE STRUCTURE," filed April 12, 2004, is incorporated herein in its entirety by reference. In addition, the subject matter of U.S. Patent No. 6,168,358 is also incorporated herein in its entirety by reference.

From the foregoing, it will be appreciated that specific embodiments of the invention have been described herein for purposes of illustration, but that various modifications may be made without deviating from the spirit and scope of the invention. For example, aspects described in the context of particular vehicles, such as aircraft, can equally apply to other vehicles, such as helicopters, rockets, watercraft, etc. Further, aspects described in the context of particular embodiments can be combined or eliminated in other embodiments. Accordingly, the invention is not limited, except as by the appended claims.

CLAIMS

We claim:

1. A shell structure comprising:
a first panel portion, the first panel portion including:
a first skin; and
a first stiffener attached to the first skin, wherein the first stiffener includes a first flange portion attached to the first skin and a first raised portion projecting away from the first skin;
a second panel portion positioned adjacent to the first panel portion, the second panel portion including:
a second skin; and
a second stiffener attached to the second skin, wherein the second stiffener includes a second flange portion attached to the second skin and a second raised portion projecting away from the second skin; and
a fitting extending across a first edge region of the first panel portion and a second edge region of the second panel portion, wherein a first end portion of the fitting overlies and is attached to the first flange portion of the first stiffener and the first skin, and wherein a second end portion of the fitting overlies and is attached to the second flange portion of the second stiffener and the second skin to join the first panel portion to the second panel portion.
2. The shell structure of claim 1 wherein the first stiffener is longitudinally aligned with the second stiffener.
3. The shell structure of claim 1 wherein the first skin is positioned in edgewise alignment with the second skin.

4. The shell structure of claim 1, further comprising a strap attached to a first edge region of the first skin and a second edge region of the second skin to splice the first skin to the second skin.

5. The shell structure of claim 1, further comprising a strap attached to a first edge region of the first skin and a second edge region of the second skin to splice the first skin to the second skin, wherein at least a portion of the strap is sandwiched between the fitting and the first edge region of the first skin and the second edge region of the second skin.

6. The shell structure of claim 1 wherein the first skin is a first composite skin of an aircraft fuselage, and wherein the second skin is a second composite skin of the aircraft fuselage.

7. An aircraft structure comprising:

a first panel portion, the first panel portion including:

a first skin;

a first stiffener having a first flange portion attached to the first skin and a first raised portion projecting away from the first skin; and

a second stiffener having a second flange portion attached to the first skin and a second raised portion projecting away from the first skin;

a second panel portion positioned adjacent to the first panel portion, the second panel portion including:

a second skin;

a third stiffener having a third flange portion attached to the second skin and a third raised portion projecting away from the second skin; and

a fourth stiffener having a fourth flange portion attached to the second skin and a fourth raised portion projecting away from the second skin; and

a fitting having a first end portion spaced apart from a second end portion, wherein the first end portion is attached to the first flange portion of the first stiffener and the second flange

portion of the second stiffener, and wherein the second end portion is attached to the third flange portion of the third stiffener and the fourth flange portion of the fourth stiffener.

8. The aircraft structure of claim 7 wherein the first end portion of the fitting overlays the first flange portion of the first stiffener and the second flange portion of the second stiffener, and wherein the second end portion of the fitting overlays the third flange portion of the third stiffener and the fourth flange portion of the fourth stiffener.

9. The aircraft structure of claim 7 wherein at least one of the first flange portion of the first stiffener and the second flange portion of the second stiffener extends toward the other to form an at least approximately continuous surface extending between the first raised portion of the first stiffener and the second raised portion of the second stiffener, and wherein the approximately continuous surface is sandwiched between the first end portion of the fitting and the first skin.

10. The aircraft structure of claim 7, further comprising a strap attached to a first edge region of the first skin and a second edge region of the second skin to splice the first skin to the second skin, wherein at least a portion of the strap is sandwiched between the fitting and the first edge region of the first skin and the second edge region of the second skin.

11. The aircraft structure of claim 7 wherein the first and second panel portions form an exterior portion of a fuselage, and wherein the aircraft structure further comprises:
the fuselage; and
means for generating lift positioned at least proximate to the fuselage.

12. The aircraft structure of claim 7 wherein the fitting has a U-shaped cross-section.

13. The aircraft structure of claim 7 wherein the fitting has a base portion and at least one upstanding edge region, and wherein the base portion is fastened to the first flange portion

of the first stiffener, the second flange portion of the second stiffener, the third flange portion of the third stiffener, and the fourth flange portion of the fourth stiffener.

14. The aircraft structure of claim 7 wherein the fitting has a base portion, a first upstanding edge portion positioned toward a first side of the base portion, and a second upstanding edge portion positioned toward a second side of the base portion, wherein the first upstanding edge portion is positioned proximate to the first raised portion of the first stiffener and the third raised portion of the third stiffener, and wherein the second upstanding edge portion is positioned proximate to the second raised portion of the second stiffener and the fourth raised portion of the fourth stiffener.

15. The aircraft structure of claim 7 wherein the first and second skins include composite materials.

16. The aircraft structure of claim 7 wherein the first and second skins and the fitting include composite materials.

17. The aircraft structure of claim 7 wherein at least the first and second stiffeners form enclosed passages with open ends.

18. The aircraft structure of claim 7 wherein the first skin includes a first portion of a window cutout and the second skin includes a second portion of the window cutout.

19. An aircraft fuselage aligned along a longitudinal axis, the aircraft fuselage comprising:

a first composite skin extending 360 degrees around the longitudinal axis;

a second composite skin positioned adjacent to the first composite skin and extending 360 degrees around the longitudinal axis;

at least a first stiffener attached to the first composite skin, wherein the first stiffener includes a first flange portion attached to the first skin and a first raised portion projecting away from the first skin;

at least a second stiffener attached to the second composite skin, wherein the second stiffener includes a second flange portion attached to the second skin and a second raised portion projecting away from the second skin; and

a fitting extending across a first edge region of the first composite skin and a second edge region of the second composite skin, wherein a first end portion of the fitting overlies and is attached to the first composite skin and a second end portion of the fitting overlies and is attached to the second composite skin to join the first composite skin to the second composite skin.

20. The aircraft fuselage of claim 19 wherein the first end portion of the fitting is attached to the first stiffener and the second end portion of the fitting is attached to the second stiffener.

21. The aircraft fuselage of claim 19 wherein the first stiffener is longitudinally aligned with the second stiffener.

22. The aircraft fuselage of claim 19, further comprising:
a passenger cabin.

23. The aircraft fuselage of claim 19 wherein the first composite skin at least partially forms a barrel section having a cross-sectional width of at least about 10 feet.

24. The aircraft fuselage of claim 19 wherein the first edge region of the first composite skin includes a first portion of a window cutout, and wherein the second edge region of the second composite skin includes a second portion of the window cutout.

25. The aircraft fuselage of claim 19 wherein the first edge region of the first composite skin includes a first portion of a window cutout, wherein the second edge region of the second composite skin includes a second portion of the window cutout, and wherein the aircraft fuselage further comprises a strap having an aperture aligned with the window cutout, the strap being fastened to the first edge region of the first composite skin and the second edge region of the second composite skin.

26. A method for manufacturing a shell structure, the method comprising:
positioning a first composite skin in edgewise alignment with a second composite skin;
attaching a first end of a fitting to the first composite skin and a first stiffener on the first composite skin, wherein the first stiffener has a first flange portion attached to the first composite skin and a first raised portion projecting away from the first composite skin; and
attaching a second end of the fitting to the second composite skin and a second stiffener on the second composite skin to join the first composite skin to the second composite skin, wherein the second stiffener has a second flange portion attached to the second composite skin and a second raised portion projecting away from the second composite skin, wherein attaching the first end portion of the fitting to the first stiffener includes attaching the first end portion to the first flange portion, and
wherein attaching the second end portion of the fitting to the second stiffener includes attaching the second end portion to the second flange portion.

27. The method of claim 26, further comprising attaching a strap to a first edge region of the first composite skin and a second edge region of the second composite skin to splice the first and second composite skins together before the fitting is attached.

28. The method of claim 26, further comprising attaching a strap to a first edge region of the first composite skin and a second edge region of the second composite skin to splice the first and second composite skins together before the fitting is attached, and wherein attaching the fitting includes sandwiching a portion of the strap between the fitting and the first

edge region of the first composite skin, and sandwiching a second portion of the strap between the fitting and the second edge region of the second composite skin.

29. The method of claim 26 wherein the first composite skin extends 360 degrees around a first axis, and wherein the second composite skin extends 360 degrees around a second axis.

30. The method of claim 26 wherein attaching a first end of a fitting to the first stiffener and a second end of the fitting to the second stiffener includes leaving end portions of the first and second stiffeners open for liquid drainage.

31. A method for manufacturing an aircraft fuselage, the method comprising:

- attaching at least a first stiffener to a first composite skin, wherein the first stiffener has a first flange portion attached to the first composite skin and a first raised portion projecting away from the first composite skin,
- the first composite skin extending 360 degrees around a first axis;
- attaching at least a second stiffener to a second composite skin, wherein the second stiffener has a second flange portion attached to the second composite skin and a second raised portion projecting away from the second composite skin, the second composite skin extending 360 degrees around a second axis;
- positioning the first composite skin adjacent to the second composite skin so that the first and second axes are at least approximately aligned; and
- attaching a first end portion of a fitting to the first stiffener and the first composite skin and a second end portion of the fitting to the second stiffener and the second composite skin to join the first composite skin to the second composite skin, wherein attaching the first end portion of the fitting to the first stiffener includes attaching the first end portion to the first flange portion, and wherein attaching the second end portion of the fitting to the second stiffener includes attaching the second end portion to the second flange portion.

32. The method of claim 31, further comprising:

attaching at least a third stiffener to the first composite skin;

attaching at least a fourth stiffener to the second composite skin; and attaching the first end portion of the fitting to the third stiffener and the second end portion of the fitting to the fourth stiffener.

33. The method of claim 31, further comprising:

attaching at least a third stiffener to the first composite skin so that a first flange portion of the first stiffener and a third flange portion of the third stiffener form an at least approximately continuous first support surface;

attaching at least a fourth stiffener to the second composite skin so that the second flange portion of the second stiffener and a fourth flange portion of the fourth stiffener form an at least approximately continuous second support surface, wherein attaching a first end portion of a fitting to the first stiffener includes attaching the first end portion to the first support surface, and wherein attaching a second end portion of the fitting to the second stiffener includes attaching the second end portion to the second support surface.

34. The method of claim 31 wherein positioning the first composite skin adjacent to the second composite skin includes positioning a first barrel section having a first width of about 15 feet or more with a second barrel section having a second width of about 15 feet or more.

35. The method of claim 31, further comprising:

forming a first portion of a window cut-out in a first edge region of the first composite skin; and

forming a second portion of a window cut-out in a second edge region of the second composite skin, wherein positioning the first composite skin adjacent to the second composite skin includes aligning the first portion of the window cut-out with the second portion of the window cut-out prior to joining the first composite skin to the second composite skin.

36. The method of claim 35, further comprising attaching a strap to the first edge region of the first skin and the second edge region of the second skin to splice the first and second composite skins together, wherein the strap includes an aperture at least approximately aligned with the first and second portions of the window cut-out.

37. The method of claim 31, further comprising attaching a strap to a first edge region of the first composite skin and a second edge region of the second composite skin to splice the first and second composite skins together before attaching the first end portion of the fitting to the first stiffener and the first composite skin and the second end portion of the fitting to the second stiffener and the second composite skin.

38. The method of claim 31, further comprising attaching a strap to a first edge region of the first composite skin and a second edge region of the second composite skin to splice the first and second composite skins together, and wherein attaching the fitting includes sandwiching a portion of the strap between the fitting and the first edge region of the first composite skin, and sandwiching a second portion of the strap between the fitting and the second edge region of the second composite skin.

39. The method of claim 31, further comprising:
attaching a third flange portion of a third stiffener to the first composite skin; attaching a fourth flange portion of a fourth stiffener to the second composite skin; attaching the first end portion of the fitting to the third flange portion of the third stiffener; and
attaching the second end portion of the fitting to the fourth flange portion of the fourth stiffener.

40. The method of claim 31, further comprising forming the fitting from composite materials.

41. The method of claim 31, further comprising: forming the first and second stiffeners from composite materials; and forming the fitting from composite materials.

42. A shell structure comprising:

a first panel portion, the first panel portion including:

a first skin having a first edge cut-out portion; and

a first stiffener attached to the first skin, wherein the first stiffener includes a first flange portion attached to the first skin and a first raised portion projecting away from the first skin;

a second panel portion, the second panel portion including:

a second skin having a second edge cut-out portion, wherein the second panel portion is positioned adjacent to the first panel portion so that the first edge cut-out portion is aligned with the second edge cut-out portion; and

a second stiffener attached to the second skin, wherein the second stiffener includes a second flange portion attached to the second skin and a second raised portion projecting away from the second skin; and

a fitting extending across a first edge region of the first panel portion and a second edge region of the second panel portion, wherein a first end portion of the fitting overlies and is attached to the first stiffener and the first skin and a second end portion of the fitting overlies and is attached to the second stiffener and the second skin to join the first panel portion to the second panel portion.

43. The shell structure of claim 42, further comprising a strap attached to a first edge region of the first skin and a second edge region of the second skin to splice the first skin to the second skin, wherein the strap includes an aperture aligned with the first edge portion cut-out and the second edge portion cut-out.

44. The shell structure of claim 42, further comprising a strap attached to a first edge region of the first skin and a second edge region of the second skin to splice the first skin to the second skin, wherein the strap includes an aperture aligned with the first edge portion cut-out and the second edge portion cut-out, and wherein at least a portion of the strap is sandwiched between the fitting and the first edge region of the first skin and the second edge region of the second skin.

45. The shell structure of claim 42 wherein the first skin is a first composite skin of an aircraft fuselage, wherein the second skin is a second composite skin of the aircraft fuselage,

and wherein the first and second edge cut-out portions form a window cutout in the aircraft fuselage.

46. A method for manufacturing a shell structure, the method comprising:
attaching at least a first stiffener to a first skin having a first edge cut-out portion;
attaching at least a second stiffener to a second skin having a second edge cut-out portion;
positioning the first skin in edgewise alignment with the second skin whereby the first edge cut-out portion is at least approximately aligned with the second edge cut-out portion; and
attaching a first end of a fitting to the first stiffener and the first skin and a second end of the fitting to the second stiffener and the second skin.

47. The method of claim 46, further comprising attaching a strap to a first edge region of the first skin and a second edge region of the second skin to splice the first and second skins together, wherein the strap includes an aperture at least approximately aligned with the first and second edge cut-out portions of the first and second skins.

48. The method of claim 46, further comprising attaching a strap to a first edge region of the first skin and a second edge region of the second skin to splice the first and second skins together, wherein the strap includes an aperture at least approximately aligned with the first and second edge cut-out portions of the first and second skins, and wherein attaching the fitting includes sandwiching a portion of the strap between the fitting and the first edge region of the first skin and the second edge region of the second skin.

49. An aircraft fuselage structural system comprising:
a first composite skin extending 360 degrees around an axis;
a second composite skin positioned adjacent to the first composite skin and extending 360 degrees around the axis;

at least first stiffening means for stiffening the first composite skin, wherein the first stiffening means includes a first flange portion attached to the first skin and a first raised portion projecting away from the first skin;

at least second stiffening means for stiffening the second composite skin, wherein the second stiffening means includes a second flange portion attached to the second skin and a second raised portion projecting away from the second skin;

splice means for attaching a first edge region of the first composite skin to a second edge region of the second composite skin; and

fitting means separate from the splice means for overlying and structurally connecting the first stiffening means to the second stiffening means.

50. The aircraft fuselage of claim 49, further comprising:

third stiffening means positioned adjacent to the first stiffening means for stiffening the first composite skin; and

fourth stiffening means positioned adjacent to the second stiffening means for stiffening the second composite skin, wherein the fitting means extends across the splice means and structurally connects the first and third stiffening means to the second and fourth stiffening means.

51. The aircraft of claim 49 wherein the first stiffening means have a first raised portion projecting away from the first composite skin and at least a first flange portion attached directly to the first composite skin, wherein the second stiffening means have a second raised portion projecting away from the second composite skin and at least second flange portion attached directly to the second composite skin, and wherein the fitting means are attached to the first and second flange portions.

52. The aircraft of claim 49 wherein the first stiffening means have a first raised portion projecting away from the first composite skin and at least a first flange portion attached directly to the first composite skin, wherein the second stiffening means have a second raised portion projecting away from the second composite skin and at least second flange portion

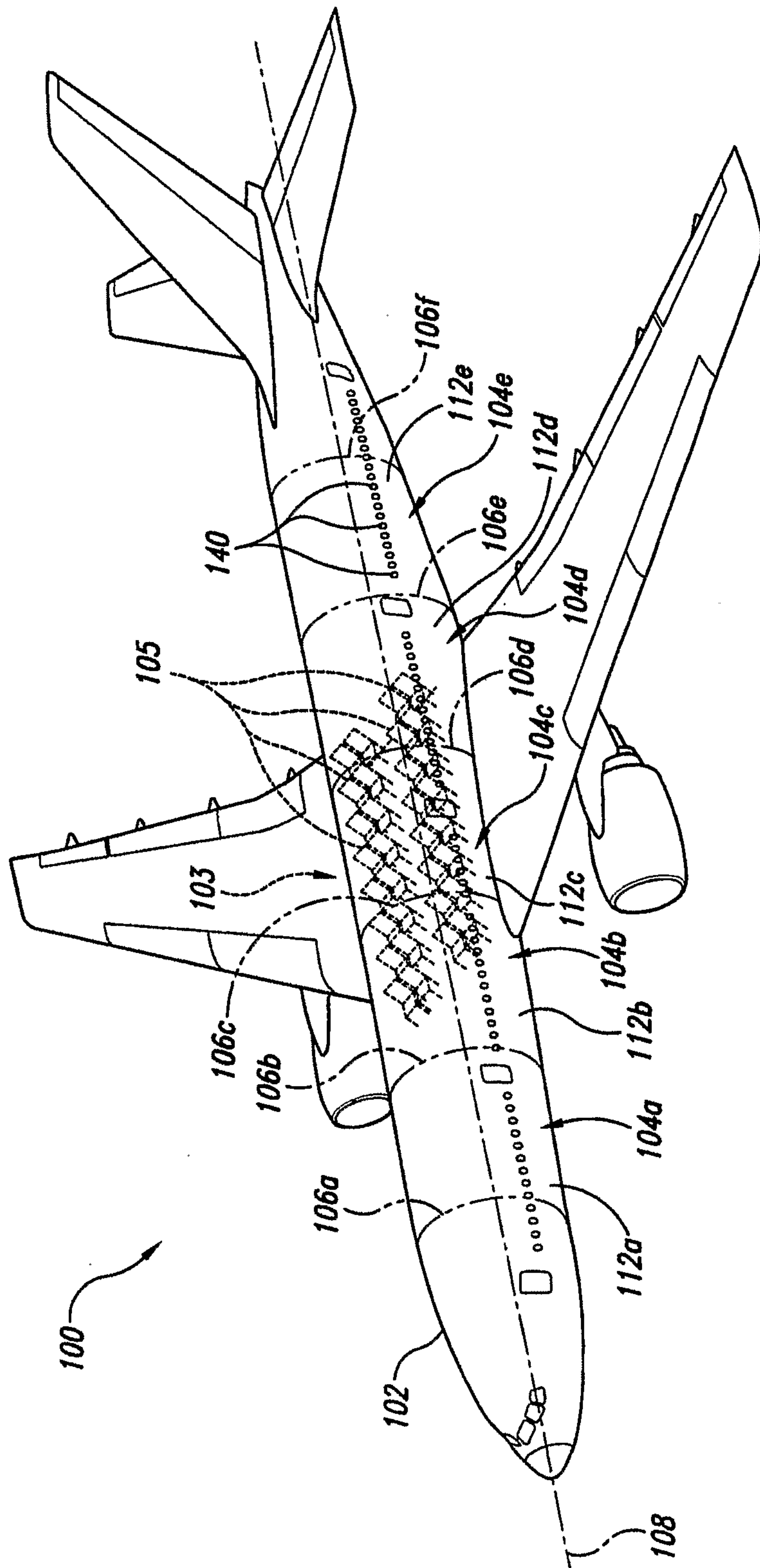
attached directly to the second composite skin, and wherein the fitting means are attached to the first and second flange portions but not to the first and second raised portions.

53. The aircraft of claim 49 wherein the first composite skin at least partially forms a barrel section having a cross-sectional width of at least about 10 feet.

54. A circumferential joint between abutting panels wherein each panel is stiffened with at least two longitudinal stiffeners, the joint comprising:

at least one splice plate overlapping the panels and having first and second end portions, wherein the first and second end portions extend into the regions between the stiffeners to connect the panels together.

55. The circumferential joint of claim 54 wherein the splice plate is attached to portions of the stiffeners.

*Fig. 1*

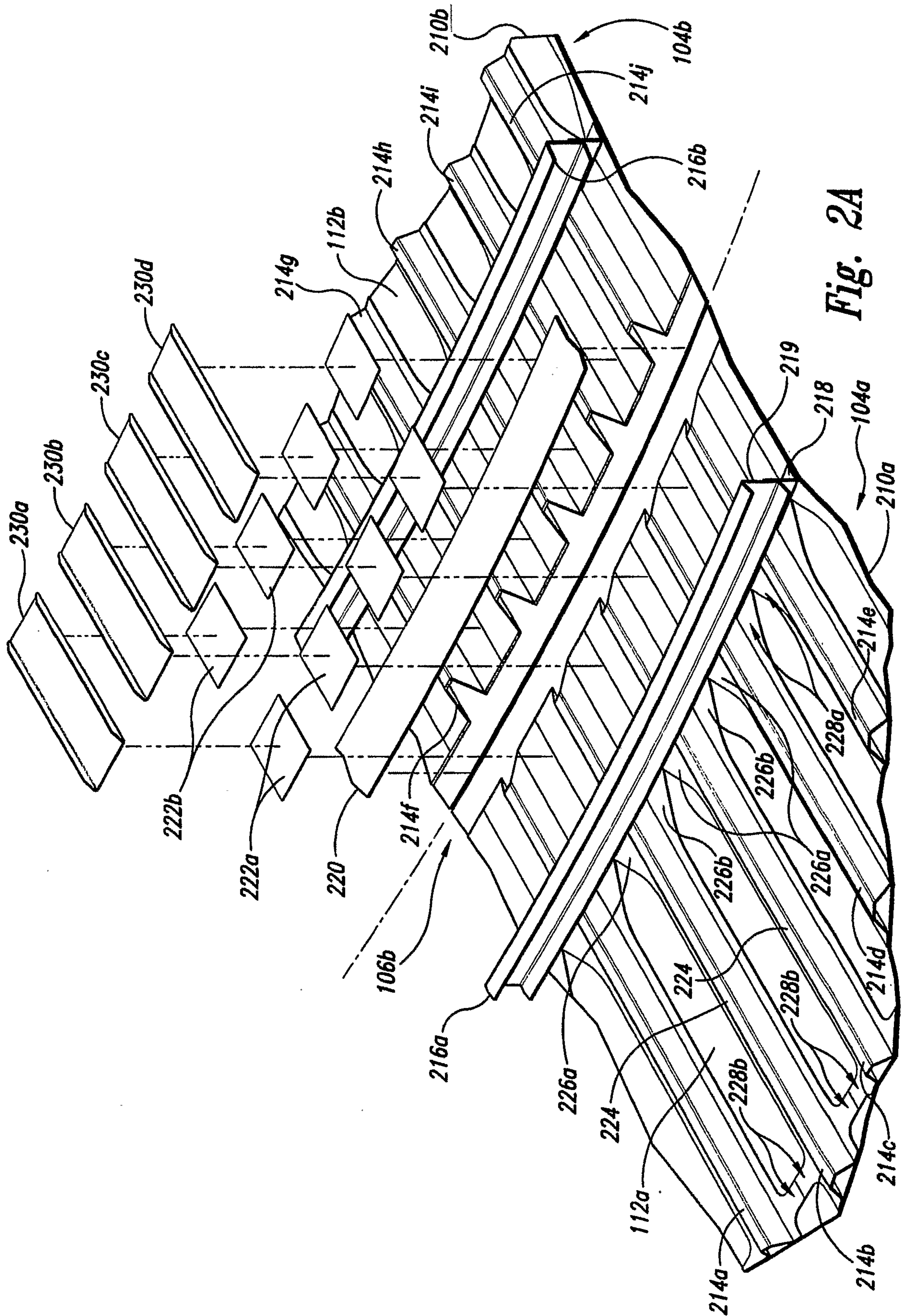


Fig. 2A

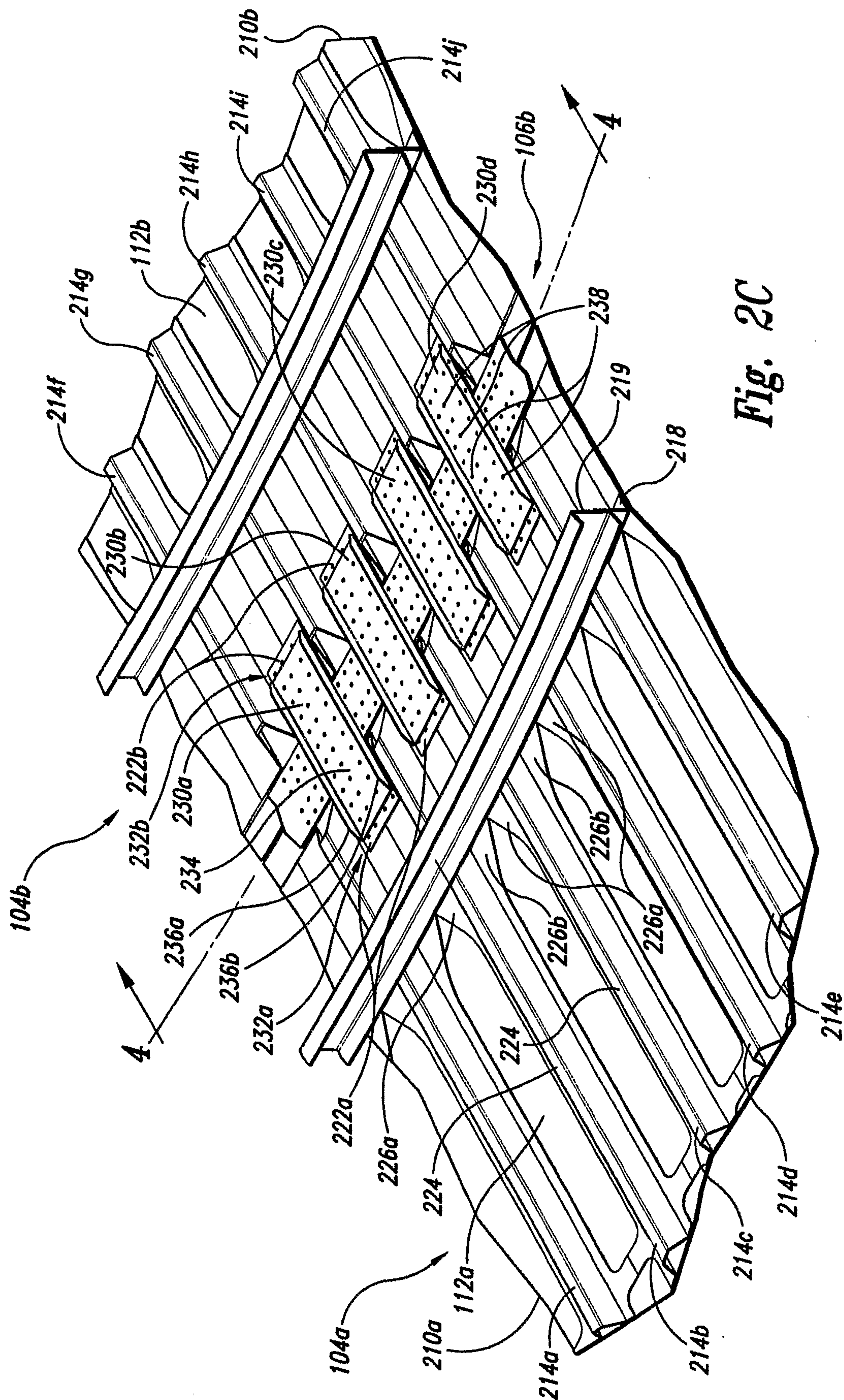


Fig. 2C

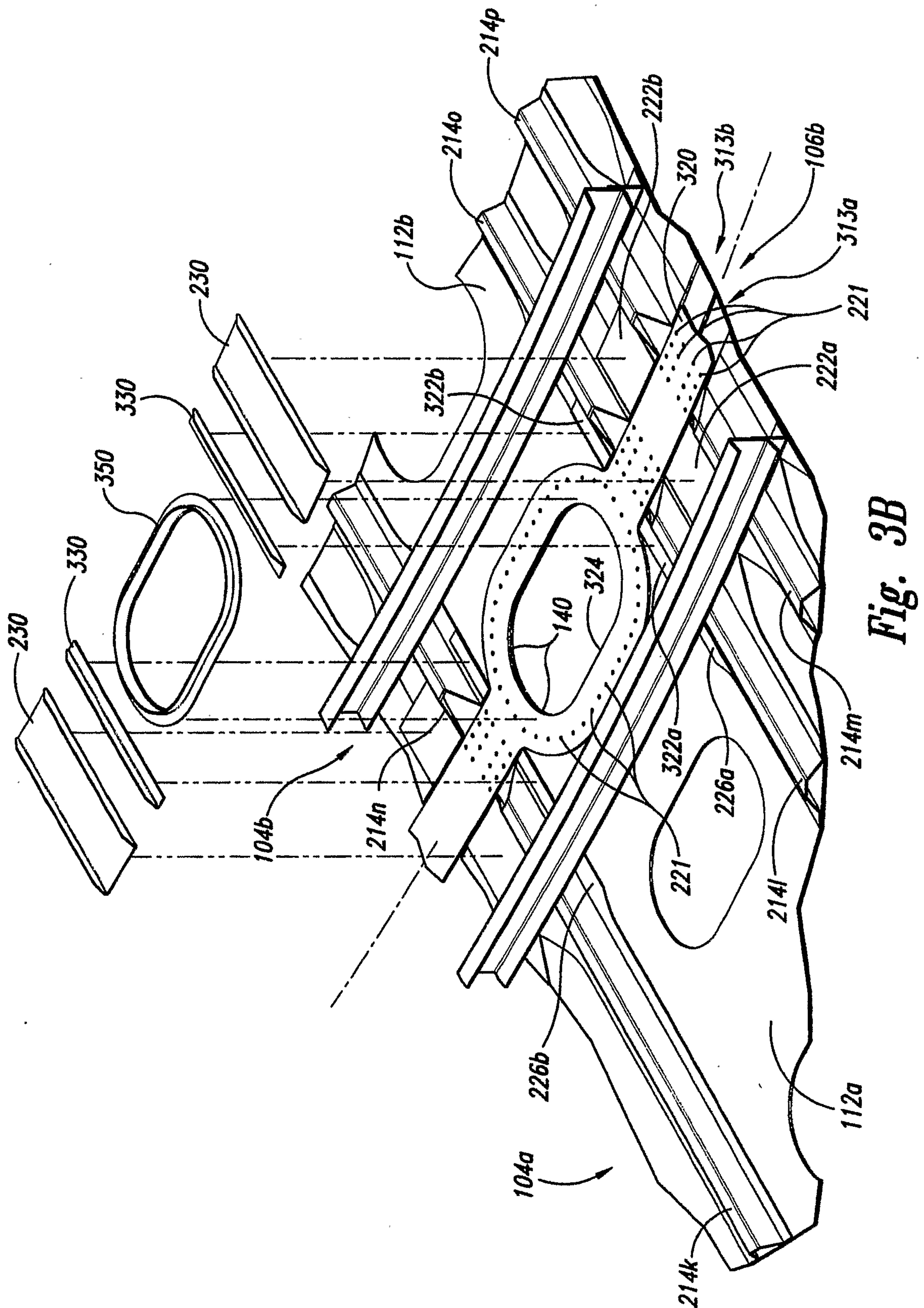
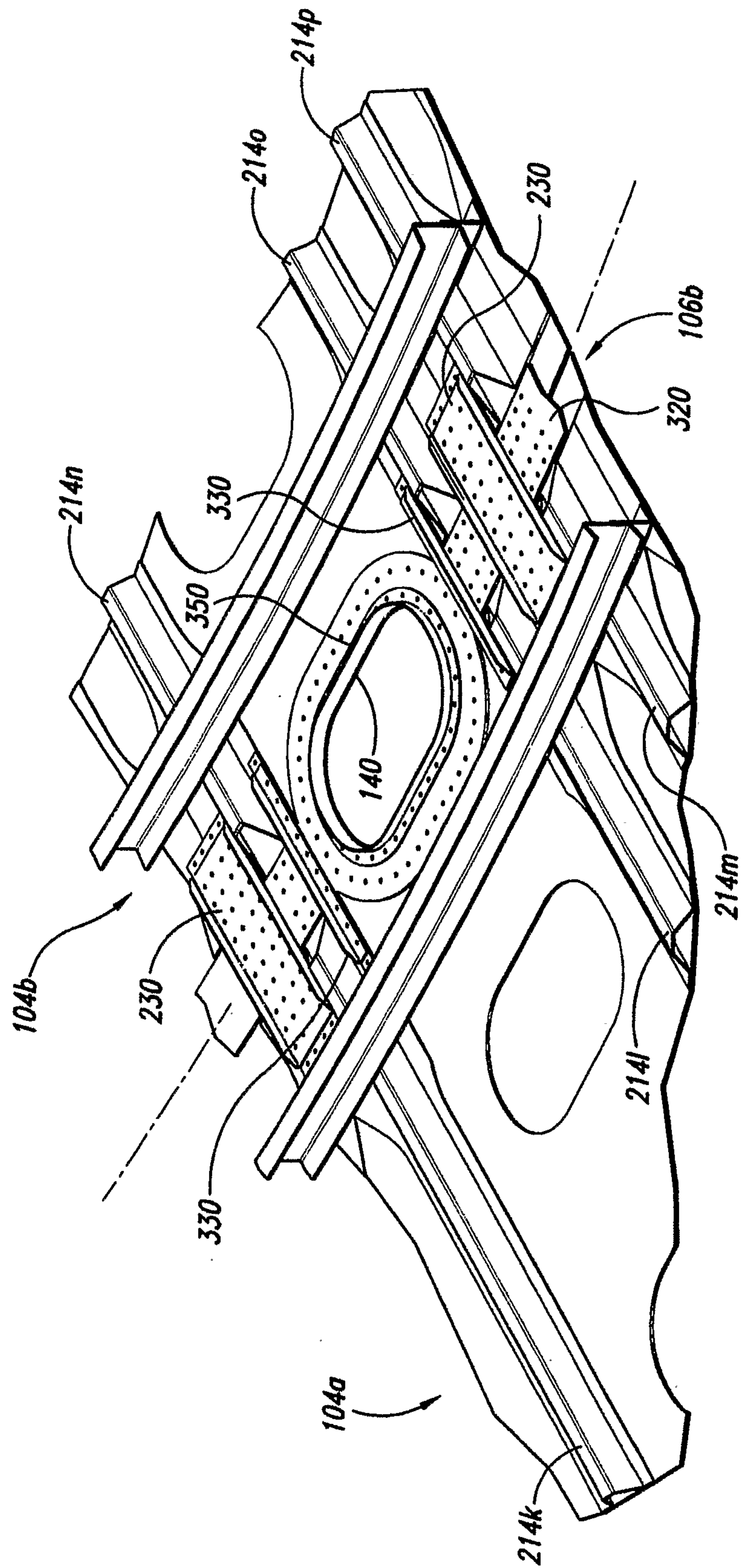
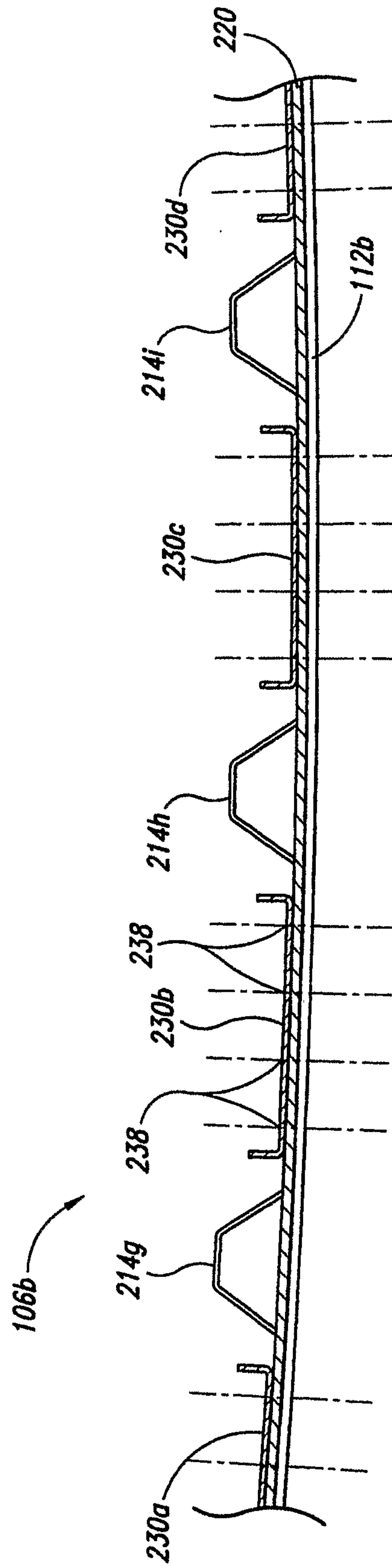


Fig. 3B

*Fig. 3C*

*Fig. 4*

