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Section 51-00 General

The Liberty XL-2 airplane uses a mixture of structural technologies including:

- Conventional aluminum construction for flying and control surfaces
- Welded steel tube construction for the fuselage center section space frame which accepts and consolidates loads from the wings, fuselage, landing gear, and engine mount
- Aluminum and steel elements for the landing gear
- Composite (fiber/epoxy laminate) constructions for the fuselage, belly fairing and engine cowlings.

Prior to contacting Liberty Aerospace Inc customer support it is imperative that the mechanic has made a major/minor repair decision or has a view to the significance of the repair. If the mechanic is unable to make the decision due to a lack of data, Liberty Aerospace Inc. customer support will furnish this data to help the mechanic make the major/minor determination.

As it is impossible to define criteria and catalogue every type of damage, inspection or repair permutation, please contact Liberty Aerospace Inc customer support for further clarification if any doubt exists with interpretation of this document.

Minor repairs to the aluminum flying surfaces and controls may be carried out using standard materials and techniques in accordance with FAA Advisory Circular 43.13-1B. These repairs should be limited to patching of small holes < 0.1in or tears in aluminum skins, replacement of moving parts such as hinges, etc. Any repairs involving damage to underlying structural components of flying surfaces (significant skin damage, any damage to underlying structures such as spars, ribs, stringers, etc.) should be referred to Liberty Aerospace Inc. Customer Support.

Composite components for the Liberty XL-2 airplane are manufactured from specialized pre-impregnated (“Pre-Preg”) materials and structural foam. The solid (no foam core) and sandwich (with a foam core) composite laminate are closely controlled in the fabrication of the Liberty XL-2 aircraft, in which the mixture of fiber reinforcing materials (fiberglass, carbon fiber, etc.) and resin matrix is very closely controlled.

The definitions of the different composite components are:

- **Primary Structures**: "The structure that carries flight, ground, loads, and whose failure would reduce the structural integrity of the airplane or may result in injury or death to passengers or crew is defined as primary structure." Table 51-1 shows a list of components that are Primary Structures.
  - Examples of primary structures made of composite materials on the XL-2 are the seats which carry crash loads, the majority of the fuselage structure which carries flight and ground loads, and the vertical tail which carries flight loads.
Interior structures that carry crash loads, as required by 14 CFR, part 23, FARs 23.561 and 23.562 are primary structure. These are the primary load carrying members. Their failure would reduce the structural integrity of the airframe. These components are an integral portion of the fuselage.

### Table 51-1 Primary Structures

<table>
<thead>
<tr>
<th>Part Number</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>135A-10-105</td>
<td>Fuselage Assembly*</td>
</tr>
<tr>
<td>135A-10-109</td>
<td>Lower Fuselage</td>
</tr>
<tr>
<td>135A-10-411</td>
<td>Upper Fuselage</td>
</tr>
<tr>
<td>135A-10-413</td>
<td>Fin Close Out (Vertical Close Out)</td>
</tr>
<tr>
<td>135A-10-423</td>
<td>Bulkhead Baggage Bay</td>
</tr>
<tr>
<td>135A-10-425</td>
<td>Bulkhead Mid Fuselage</td>
</tr>
<tr>
<td>135A-10-427</td>
<td>Fin Spar</td>
</tr>
<tr>
<td>135A-10-431</td>
<td>Fin Rib #1</td>
</tr>
<tr>
<td>135A-10-433</td>
<td>Fin Rib #2</td>
</tr>
<tr>
<td>135A-10-435</td>
<td>Fin Rib #3</td>
</tr>
<tr>
<td>135A-10-437</td>
<td>Baggage Bay Floor</td>
</tr>
<tr>
<td>135A-10-439</td>
<td>Duct. NACA - Port Cabin Air</td>
</tr>
<tr>
<td>135A-10-440</td>
<td>Duct. NACA - Stbd Cabin Air</td>
</tr>
<tr>
<td>135A-10-444</td>
<td>Hoop Reinforcement, Fwd</td>
</tr>
<tr>
<td>135A-10-445</td>
<td>Baggage Bay Floor Support, Port</td>
</tr>
<tr>
<td>135A-10-446</td>
<td>Baggage Bay Floor Support, Starboard</td>
</tr>
<tr>
<td>135A-10-465</td>
<td>Fuselage Bond Line Reinforcement Strap</td>
</tr>
<tr>
<td>135A-10-481</td>
<td>Closeout, Seat Back, Port</td>
</tr>
<tr>
<td>135A-10-482</td>
<td>Closeout, Seat Back, Starboard</td>
</tr>
<tr>
<td>135A-10-483</td>
<td>Headliner</td>
</tr>
<tr>
<td>135A-11-107</td>
<td>Seat Back Stiffener Installation*</td>
</tr>
<tr>
<td>135A-11-187</td>
<td>Seat Base Stiffener Installation*</td>
</tr>
<tr>
<td>135A-11-403</td>
<td>Fuselage Bond Line Reinforcement Strap for Rollover Hoop</td>
</tr>
<tr>
<td>135A-11-409</td>
<td>Seat Base Stiffener</td>
</tr>
<tr>
<td>135A-11-422</td>
<td>Seat Back Stiffener, Rear</td>
</tr>
<tr>
<td>135A-11-424</td>
<td>Seat Back Stiffener, Top</td>
</tr>
<tr>
<td>135A-11-426</td>
<td>Seat Back Stiffener, Side</td>
</tr>
<tr>
<td>135A-50-215</td>
<td>Bulkhead Reinforcement installation*</td>
</tr>
<tr>
<td>135A-50-413</td>
<td>Reinforcement, Fwd Bulkhead</td>
</tr>
</tbody>
</table>

**Secondary Structures:** These are not primary load carrying members AND their failure would not reduce the structural integrity of the airframe. These components do not form an integral portion of the fuselage, for example access panels. Table 51-2 shows a list of components that are Secondary Structures.
### Part Number | Description
---|---
135A-10-123 | Door Frame Assembly, Port*
135A-10-124 | Door Frame Assembly, Starboard*
135A-10-407 | Belly Fairing
135A-10-487 | Inner Door Frame Port
135A-10-488 | Inner Door Frame Starboard
135A-11-401 | Footstep Hard point - Port
135A-11-402 | Footstep Hard point – Starboard
135A-11-407 | Outer Door Shell, Port
135A-11-408 | Outer Door Shell, Starboard
135A-50-401 | Footstep Hard point - Port
135A-50-403 | Lower Cowl
135A-80-413 | Instrument Console Untrimmed

### Table 51-2 Secondary Structures

- **Tertiary Structures:** These are not primary or secondary load carrying members. These components do not form an integral portion of the fuselage. Table 51-3 shows a list of components that are Tertiary Structures.

### Part Number | Description
---|---
135A-10-419 | Baggage Bay Close Out
135A-10-418 | Fin Horn Closeout
135A-10-447 | Baggage Bay Floor Access Panel
135A-10-449 | Access Panel, Fuel Sender
135A-10-456 | Access Panel Torque Tube, Upper
135A-10-458 | Access Panel Torque Tube, Lower
135A-10-460 | Access Panel, Trim Motor
135A-10-469 | Access Cover, Port Door Actuator
135A-10-470 | Access Cover, Stbd Door Actuator
135A-20-415 | Wing Root Fairing, (Port)
135A-20-416 | Wing Root Fairing, (Stbd)
135A-20-421 | Wing Tip Port
135A-20-422 | Wing Tip Stbd
135A-30-401 | Tailplane Tip (Port & Stbd)
135A-30-407 | Rudder Horn
135A-30-409 | Rudder Tip - Lower
135A-40-021 | Main Wheel Fairing Assy, Port*
135A-40-022 | Main Wheel Fairing Assy, Stbd*
135A-40-401 | Nose Gear Fairing, Fwd
135A-40-402 | Aft Nose Gear Fairing Assembly
135A-40-403 | Main Wheel Fairing, Port
135A-40-404 | Main Wheel Fairing, Stbd
135A-40-405 | Main Wheel Fairing Panel, Port
### Table 51-3 Tertiary Structures

<table>
<thead>
<tr>
<th>Part Number</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>135A-40-406</td>
<td>Main Wheel Fairing Panel, Stbd</td>
</tr>
<tr>
<td>135A-40-411</td>
<td>Wheel Fairing Bulkhead, Port</td>
</tr>
<tr>
<td>135A-40-412</td>
<td>Stbd Wheel Fairing Bulkhead, Aft</td>
</tr>
<tr>
<td>135A-50-406</td>
<td>Access Panel, Oil</td>
</tr>
<tr>
<td>135A-50-415</td>
<td>Spinner Fwd Plate</td>
</tr>
<tr>
<td>135A-50-417</td>
<td>Spinner Back Plate</td>
</tr>
<tr>
<td>135A-50-423</td>
<td>Spinner</td>
</tr>
</tbody>
</table>

If the A&P technician is unable to determine the nature of whether the structure is primary or secondary, or whether additional clarification is required prior to the mechanic making the determination whether a repair is major or minor, the mechanic must contact Liberty customer support for further clarification.

If the A&P technician considers that the damage is extensive and/or that the impact may warrant additional Non Destructive Inspection beyond the definition provided within this document, the mechanic must contact Liberty customer support for further clarification.

### Section 00-01 Qualified Facilities And Documentation

Repairs to the Liberty XL-2 composite components (sandwich and solid laminate) or metal components may be carried out by any appropriately rated composite repair facility familiar with primary structural composite repairs. Example repair processes are delineated in the later section of this chapter.

Documentation of approved repairs should go into the aircraft maintenance logbook and documented in accordance with FAR part 43.9.

![NOTE](image)

The following materials and processing specifications are required to be reviewed prior to performing the composite repairs listed herein. These material and processing specifications are available from Liberty Aerospace, Inc. Customer Service upon request.

<table>
<thead>
<tr>
<th>Document number</th>
<th>Title</th>
</tr>
</thead>
<tbody>
<tr>
<td>135A-911-042</td>
<td>Liberty aircraft inspection plan</td>
</tr>
<tr>
<td>135A-925-001</td>
<td>Airex Core material specification</td>
</tr>
<tr>
<td>135A-925-985</td>
<td>Epibond paste adhesive material spec</td>
</tr>
<tr>
<td>135A-925-997</td>
<td>Toray plain weave material specification</td>
</tr>
<tr>
<td>135A-926-012</td>
<td>Prime and paint process specification</td>
</tr>
<tr>
<td>135A-926-994</td>
<td>Secondary bonding of composite materials</td>
</tr>
<tr>
<td>135A-926-998</td>
<td>Processing of composite materials</td>
</tr>
</tbody>
</table>
Section 51-10 Structural Composite Repairs

The fuselage of the Liberty XL-2 is composed of structural composite materials. The fuselage aft of the engine cowling is fabricated from carbon fiber reinforced fabrics that are used as facing plies adhered to core materials to form a structural sandwich. As the fuselage is a load bearing monocoque structure, care must be taken when performing repairs to the fuselage. The processes used to perform the structural repair must be followed and the fabrication steps must be adhered with in accordance with the FAA approved technical documents noted on page 6 and per instructions delineated in the later section of this chapter.

**WARNING**

LIBERTY AEROSPACE INC CUSTOMER SUPPORT MUST BE CONTACTED PRIOR TO PERFORMING THESE REPAIRS.

MODIFICATIONS TO THE FUSELAGE STRUCTURE THAT ARE TO BE PERFORMED ON FORM 337 OR VENDOR STC’ MUST BE COORDINATED WITH LIBERTY AEROSPACE INC. CUSTOMER SUPPORT. IF COORDINATION DOES NOT OCCUR, ANY FIELD REPAIRS OR INSTALLATION MAY LEAD TO A NON-AIRWORTHY FUSELAGE STRUCTURE.

Section 10-01 An Extensive Fuselage Or Flying Surface Skin Repair

In the event that large areas > 1.0” of the aircraft skin require repair, it may be difficult to reform the correct surface profile without proper rigid tooling. In addition, the structure may be weakened by the extensive removal and replacement of a load bearing skin. For performing structural repair of such type, contact Liberty Aerospace Inc Customer Support for additional instructions.

Section 10-02 Fittings Requiring Jigging For Positional Location

In the event that fittings have been torn from their original location, this may require special jigging to ensure that they are correctly re-located relative to neighboring components. In the event that a repair of this nature is required, contact Liberty Aerospace Inc. Customer Support for additional instructions.
PAGE LEFT INTENTIONALLY BLANK
Section 51-11 Composites - Solid Laminate And Sandwich Construction

The majority of the fuselage of the Liberty XL-2 is carbon fiber reinforced sandwich construction. Different thicknesses of core material are used to form a structural sandwich in order to support the distribution of stresses through the structure. There are areas of the structure, like the rollover hoop structure for example that are solid laminate. Care must be taken to inspect and determine the nature of the composite base materials prior to performing the repair. Solid laminate has the ability to support concentrated bearing and clamp-up loads better than sandwich construction. Prior to performing any structural repair, the base material should be reviewed and compared with the appropriate Liberty XL-2 drawings to ensure that the correct repair method is performed.

The repair methods noted below make the distinction between solid laminate and sandwich construction repairs.
Section 51-12 Composites Damage Classification

Damage to the fuselage can occur because of accident, negligence, or corrosion. These may comprised of scratches, dents, tears, holes or cracks. Such damage must be classified to ensure the adequacy of the repairs.

Damage to the fuselage can be classified into four distinct type of damages based on the following criteria:

- Load-path type (primary/secondary structure)
- Location on fuselage ((any sandwich/solid laminate/cosmetic)
- Damage size (length/diameter)
- Damage number (per unit length / area)

Critical damage dimensions have been determined for the Liberty XL-2 fuselage. The acceptable limit for the quantity of ‘damage’ per unit length or area has also been determined. This allows the fuselage damage to be specified as one of four with each types (1 – 4) having two classes (I and II). Damage should therefore be recorded by the maintenance facility as Type 1, Class II etc.

Laminate classifications are defined in Table 51-4 below. These classifications are used solely to define allowable laminate defects.

<table>
<thead>
<tr>
<th>Laminate Classification</th>
<th>Maximum Cumulative Defect Size</th>
<th>Defect Accumulation Threshold (DAT)</th>
</tr>
</thead>
<tbody>
<tr>
<td>I</td>
<td>1.00”</td>
<td>1.0 ft²</td>
</tr>
<tr>
<td>II</td>
<td>2.00”</td>
<td>1.0 ft²</td>
</tr>
</tbody>
</table>

**Table 51-4 Laminate Defect Classifications**

Laminate defects include:

- Impact damage (such as obvious fracture or penetration of matrix or fibers)
- Inclusions (foreign matter in laminate)
- Extreme porosity (frequent small , 0.050in voids in laminate)

Shallow (less than 0.005 in) scratches and mild porosity (less than 0.050 sq in within a 1sq inch region) do not count as structural defects. When there is doubt regarding classifying a potential defect as a defect, treat it as a defect.

The greatest linear dimension of a defect defines the severity of the defect. For example, circular defects are defined by their diameter, crack defects are defined by their length, and other defects are defined by their largest measurement.

All defects within the Defect Accumulation Threshold, or DAT, are cumulative. For example, if the DAT allows for 1.0” within a 1.0 ft² area, then add up all defect sizes within a given square foot. If the total exceeds 1.0 inch, then submit the component to Liberty Aerospace Inc. for determination.
The examples of laminate defect size definition show how the DAT is used to determine cumulative defects. After all defects are identified, move the DAT “window” around to determine which defects must be treated as cumulative.

**Section 12-01  Type 1 - Damage**

The definition of Type 1-damage is critical damage inflicted to primary or secondary structure at locations such as highly stressed (contact Liberty Aerospace Customer support for clarification) regions and underlying structural elements.

Damage is classified as Type 1 when the size, location, and number of damages per unit length or area endanger the structural integrity of the aircraft.

This type of damage requires partial or complete reconstruction of parts or repairs of large areas.

**Section 12-02  Type 2 - Damage**

The definition of Type 2-damage is damage inflicted to primary or secondary structure involving complete penetration of the sandwich or laminate materials. Damage is classified as Type 2 if it has the potential to affect the structural integrity of the aircraft in flight.
Section 12-03  Type 3 - Damage

The definition of Type 3-damage is damage limited to outer skin only (no damage to internal facing plies or core material). Damages are classified Type 3 when the size, location and number of damage per unit length/area does not endanger the structural integrity of the aircraft.

Section 12-04  Type 4 - Damage

The definition of Type 4-damage is damage that is inflicted to parts of minimal structural importance. Type 4 damage includes light surface erosion, scratches, grooves, dents (no ply and core damage), etc. that do not penetrate the composite outer skin. This includes damage to replaceable access covers, etc.
Section 51-13 Laminated Composites Inspection Criterion

Table 51-5 below defines laminate composite inspection classifications.

<table>
<thead>
<tr>
<th>Laminate Classification</th>
<th>Minimum Required Inspection</th>
</tr>
</thead>
<tbody>
<tr>
<td>NS</td>
<td>Inspection type not specified</td>
</tr>
<tr>
<td>A</td>
<td>100% visual + 100% NDI ultrasound</td>
</tr>
<tr>
<td>B</td>
<td>100% visual + 100% NDI tap testing</td>
</tr>
<tr>
<td>C</td>
<td>100% visual only</td>
</tr>
</tbody>
</table>

Table 51-5 Inspection Classifications

The classification “NS” means that the inspection method is not specified. Any practical method(s) may be used if the method chosen demonstrates itself to the A&P Mechanic that the defect criterion may be met for the component, i.e. the defect can be picked up by the method.

“Visual” means optical. Actual inspection may be done via the naked eye, a bore scope, or other imaging means.

A component may always be inspected more intensively than its classification requires. For example, if a laminate is classified as “C” (visual only), it may still be inspected with ultrasound, tap testing, or other means in addition to the visual inspection as deemed necessary to make a major or minor determination by the A&P Mechanic. If question exists over the methods to employ for inspection or over the need for a more intensive inspection technique, Liberty Aerospace Inc Customer Support must be contacted to obtain clarification.

The specified inspection type is only prescribed for the initial inspection. If defects are found, the defects must be thoroughly mapped using whatever additional inspection means may be deemed necessary by the A&P Mechanic. For example, if tap testing is required and a defect is found, it may be necessary to use ultrasound to fully define the defect. Liberty Aerospace Inc Customer Support must be contacted to obtain these additional inspection instructions.

Section 13-01 Composite Laminates

All composite laminates in the fuselage and vertical stabilizer must be inspected for defects or to evaluate any structural damage. The classification is I-B for all fuselage and vertical stabilizer laminates. Laminate classification defines acceptable defect criteria.
Section 13-02 Tap Testing

The fuselage has been inspected prior to release, the methods used were visual and tap testing. At the time of aircraft release no ultrasound/radiographic testing was performed, but should be considered by an A&P Mechanic as an alternate means of evaluation if doubt exists as to whether damage exists in the structure that requires repair. The use of ultrasound must be coordinated with Liberty Aerospace Inc Customer Support.

Tap testing is widely used to detect the presence of delaminations or debonding. The tap testing procedure consists of lightly tapping the surface of the part with a coin, or light special hammer with a maximum of 2 ounces (see figure below) or any other suitable object.

A flat or dead response is considered unacceptable. The acoustic response of a good part can vary with changes in geometry and laminate thickness. Care should be taken to compare good solid laminate with solid laminate of interest. Similarly good adhesive joints should be compared with the adhesive joint of interest and good sandwich structure of comparable lay-up and thickness should be compared with sandwich structure of interest.

By removing internal access panels, access to both sides of the laminate can be obtained. A review of the fuselage illustration located later within this chapter highlights the bonded areas of the structure for clarity.

The entire area of interest must be tap tested. The surface should be dry and free from oil, grease, and dirt.

NOTE

The accuracy of this test depends on the subjective interpretation of the test response; therefore, only A&P technicians familiar with composite tap testing should perform this test.

As it is impossible to define criteria for all permutations of damage that could require inspection, Liberty customer support should be contacted in the event that a good comparison cannot be obtained from the initial tap test.

Figure 51-3 Sample Of Special Tap Test Hammer For Tap Testing
Section 51-14 Laminated Composites Repair Procedures

All repairs must be coordinated with Liberty Aerospace, Inc. Customer Support. Separate FAA approval will be obtained for aircraft specific repair instructions. Liberty Customer Support will provide these instructions to the customer.

REPAIR METHODS PROVIDED HEREIN ARE BY WAY OF EXAMPLE ONLY. THEY ARE INTENDED TO HELP THE MAINTENANCE TECHNICIAN TO DETERMINE THE NATURE OF THE REPAIR REQUIRED FOR THE DAMAGE AND TO ELABORATE ON THE ACCEPTED PROCESS THAT WILL BE SPECIFIED TO PERFORM THE REPAIR. FOR SPECIFIC FAA APPROVED INSTRUCTIONS ON THE REPAIR OF COMPOSITE DAMAGE, CONTACT LIBERTY AEROSPACE, INC. CUSTOMER SUPPORT.

AS IT IS IMPOSSIBLE TO DEFINE CRITERIA AND CATALOGUE EVERY DAMAGE, INSPECTION OR REPAIR PERMUTATION, PLEASE CONTACT LIBERTY AEROSPACE INC. CUSTOMER SUPPORT FOR FURTHER CLARIFICATION IF ANY DOUBT EXISTS WITH INTERPRETATION OF THIS DOCUMENT.

This section has the procedures to effect repairs to the four types of damage. There are different repair procedures depending on the severity and location of the damage.

Type 1 - damage is critical damage inflicted to primary or secondary structure at locations such as highly stressed regions and underlying structural elements.

Type 2 - damage is damage inflicted to a primary or secondary structure involving complete penetration of the sandwich or laminate materials. Employ the same repair procedures as Type 1 – damage.

The following is a list of procedure to effect repairs for Type 1 and Type 2 damage.

<table>
<thead>
<tr>
<th>Procedure Title</th>
<th>Page</th>
</tr>
</thead>
<tbody>
<tr>
<td>Type 1 or 2 Damage To One Side Of Sandwich (Co-Cure Repair)</td>
<td>21</td>
</tr>
<tr>
<td>Type 1 or 2 Damage To One Side Of Sandwich (Alternative Secondary Bonding Repair)</td>
<td>24</td>
</tr>
<tr>
<td>Type 1 or 2 Damage To Both Sides Of Sandwich Structure (Flat Surface)-Secondary Bonding</td>
<td>26</td>
</tr>
<tr>
<td>Type 1 or 2 Damage To Both Sides Of Sandwich Structure (Both Flat Surface – Alternate Co-Cure Repair)</td>
<td>29</td>
</tr>
<tr>
<td>Type 1 or 2 Damage To Both Sides Of Sandwich Structure (Curved Surface)</td>
<td>32</td>
</tr>
</tbody>
</table>

Type 3 - damage is damage limited to outer skin only. No damage is caused to the underlying foam core. This damage may be of following types:

- Delamination between two plies
- Fibers damage to the top plies
- No foam core is damaged for TYPE 3 damage inflicted to the XL-2 composite structure.

In any of the above cases, there is to be no damage to the underlying foam core.

The follow is a list of procedures for all Type 3 case.

<table>
<thead>
<tr>
<th>Procedure Title</th>
<th>Page</th>
</tr>
</thead>
<tbody>
<tr>
<td>Type 3 Single Skin Repair</td>
<td></td>
</tr>
<tr>
<td>One Side Accessible (Co-Cure Repair)</td>
<td>35</td>
</tr>
<tr>
<td>Type 3 Single Skin Repair</td>
<td></td>
</tr>
<tr>
<td>One Side Accessible (Alternative Secondary Repair)</td>
<td>37</td>
</tr>
<tr>
<td>Type 3 Single Skin Repair</td>
<td></td>
</tr>
<tr>
<td>Both Sides Accessible (Secondary Bonding Repair)</td>
<td>39</td>
</tr>
</tbody>
</table>

Type 4-damage is defined as damage that is inflicted to parts of minimal structural importance. Rework to the outer surface of the fuselage can be smoothed using aerodynamic filler during final painting in accordance with 135A-926-012.

NOTE

If rework/repair is done on the outer surface of the fuselage, it is to be smoothed using aerodynamic filler during final painting in accordance with 135A-926-012.
TYPE 1 OR 2 DAMAGE TO ONE SIDE OF SANDWICH
(CO-CURE REPAIR)

This procedure covers the repair of crushed foam core for a single skin of a sandwich structure. The belly panel, upper and lower cowl comes under this category.

Perform the following initial preparation as follows:

- Minimum 2-ply of Plain Weave, or PW, (135A-925-997) will be laid up with same orientation as of the plies removed such that the covering repair patch equals or exceeds the lay-up in the deviant region. Thickness of one cured ply is between 0.0083-0.0089 inches. Repair patch (cured or uncured) must replicate the same number of plies \((n)\) removed plus one i.e. \((n+1)\) from the discrepant/deviation section (For example if 2 plies of carbon PW are present prior to the 3mm /5 mm foam, then the patch repair must be \(n+1\) i.e. 3 plies)
- During core removal or surface preparation, no damage should be caused to the underlying ply.
- The size of the repair is to be such that the crushed foam core region is removed and the patch extends minimum +1” all around the deviant region.
- The Plies need to be staggered such that each ply extends 0.5”-1” beyond the previous ply with the innermost ply of the patch being the smallest.
- Crushed/indented foam core must be removed with chamfered edges 30°-45°.
- Shape and size of the repair foam core must be same as that of the core removed.
- The Patches should be rectangular (rounded edge) or elliptical or circular in shape.

NOTE

This covers both flat and curved surfaces. Flat surface is shown for clarity.

Perform this procedure to affect a repair to the composite.

Crushed Core Discrepancy

![Figure 51-4 Original View of the Damage](image)

1. Completely grind out the crushed foam from the deviant region without damaging the underlying ply.
2. Clean and prep the surface +1" of the core removed and apply two plies of Hysol EA9696 (0.03psf) Film adhesive (135A-925-992) such that it extends +1" beyond the exposed foam core region.

3. Create a chamfered 3mm or 5mm Airex foam core piece such that it matches the sanded/prepped region and locate it in the repair section as shown in Figure 51-7.

4. Lay up at least two plies of Carbon Plain weave (135A-925-997) at +/- 45° such that it covers the film adhesive. (Stagger Plies)

5. Vacuum bag the repair and perform a standard AGATE/TCA cure cycle in accordance with 135A-926-998 specifications.

6. Upon completion of the cure, inspect the following aspects:


NOTE

The number of plies and orientation shown are for illustration purpose only. The number of plies laid-up (n+1) depends upon the number of plies removed (n).
- The cure cycle is within process spec limits (135A-926-998)
- DMA coupon was created and tested TG passed the requirement detailed within 135A-925-997
- The consolidation of the repair is good with no void/discontinuity as observed by inspection procedures in accordance with 135A-911-042

Based on a successful post repair inspection the part may be released to service.
TYPE 1 OR 2 DAMAGE TO ONE SIDE OF SANDWICH
(ALTERNATIVE SECONDARY BONDING REPAIR)

Perform this procedure to affect a repair to the composite material.

1. Completely grind out the crushed foam core region without damaging the underlying ply. Clean and prep the surface +1" all around the discrepant/deviant section as shown:

   Grind region enough
to remove crushed core

2. Secondarily bond a pre-cured patch of at least 2 ply PW@45°/ 3mm or 5mm Foam Core using Epibond 1590 adhesive (135A-925-985) such that it extends minimum +1" beyond the repair region. Mechanically apply a distributed load on the back of the patch to ensure the Epibond adhesive joint is 0.015in (+0.025, -0.010) thick in accordance with 135A-926-994-G until handling strength, 6hr per 135A-926-994-H.

3. Elevated temperature post cured the patch in accordance with 135A-926-994.

4. After completion of the cure, inspect the following aspects:

The number of plies and orientation shown are for illustration purpose only. The number of plies laid-up (n+1) depends upon the number of plies removed (n).
- Pre cured patch must have good Material traceability (135A-925-997 & 135A-925-001) and must be processed in accordance with 135A-926-998 with good cure and good DMA.
- Check that a good SHORE D and DMA for the mixed batch of Epibond adhesive in accordance with 135A-925-985 & 135A-926-994.
- The consolidation of the repair is good with no voids/discontinuities observed during tap testing of the repaired region.
TYPE 1 OR 2 DAMAGE TO BOTH SIDES OF SANDWICH STRUCTURE (FLAT SURFACE)-SECONDARY BONDING

This procedure covers the repair of crushed foam core from both sides of a sandwich structure.

**NOTE**

If the rework/repair is done on the outer surface of the fuselage, it is to be smoothed using aerodynamic filler during final painting in accordance with 135A-926-012.

Damaged regions under this section: Type 1 and 2 damages are categorized under this section. A critical damage inflicted to primary or secondary structures from both sides, at locations such as highly stressed (contact Liberty Aerospace Customer support for clarification) regions, bonding areas etc. The belly panel, upper and lower cowl comes under this category.

The initial preparation will be done as follows:

- Minimum two Plies of Plain weave (PW) (135A-925-997) will be laid up with same orientation as of the plies removed such that the covering repair patch equals or exceeds the lay-up in the deviant region. Thickness of one cured ply is between 0.0083-0.0089 inches. Repair patch (cured or uncured) must replicate the same number of plies (n) removed plus one i.e. (n+1) from the discrepant/deviation section (For example if 2 plies of carbon PW are present prior to the 3mm /5 mm foam, then the patch repair must be n+1 i.e. 3 plies)
- The size of the Repair is to be such that the crushed foam core region is removed and the patch extends minimum +1” all around the deviant region.
- Crushed/indented foam core must be removed with chamfered edges 30º-45º.
- Shape and size of the repair foam core must be same as that of the core removed.
- The Patches should be rectangular (rounded edge) or elliptical or circular in shape.

Perform this procedure to affect repair of the composite.

![Diagram of Indentations and Core](image)

**Figure 51-12 Cross Section of Damaged Composite Sandwich**

1. Completely grind out the crushed foam core region as follows:
Grind region enough to remove crushed core

Figure 51-13 Section of the Composite after Grinding to Remove Damaged Area

2. Clean and surface prep the Outer and the inner surface +1” all around the discrepant section.

3. Create a secondarily bond of a pre-cured patch of at least two ply PW@45 using Epibond 1590 adhesive (LAI 135A-925-985) such that it extends minimum +1” beyond the repair region all around. Mechanically apply a distributed load on the back of the patch to ensure the Epibond adhesive joint is 0.015in (+0.025, -0.010) thick in accordance with 135A-926-994-G until handling strength, 6hr per 135A-926-994-H.

Figure 51-14 Application of the Epibond Adhesive and the Staggered Pliés of Plain Weave with a Backing Support

If damage is accessible from both sides, backing fixture/support (Aluminum caul plate or similar) can be used after this step.

The number of plies and orientation shown are for illustration purpose only. The number of plies laid-up (n+1) depends upon the number of plies removed (n).

4. Secondarily bond pre-cured 2 ply PW@45°/ 3mm or 5mm Foam Core using Epibond 1590 adhesive (LAI 135A-925-985) such that it extends minimum +1” beyond the repair region all around. Mechanically apply a distributed load on the back of the patch to ensure the Epibond adhesive joint is 0.015in (+0.025, -0.010) thick in accordance with 135A-926-994-G until handling strength, 6hr per 135A-926-994-H.
Figure 51-15 Insertion of the New Foam Core and Application of the Epibond Adhesive and Staggered Plies of Plain Weave

5. Elevated temperature post cure the repair in accordance with 135A-926-994.

6. After completion of the cure, inspect the following aspects:
   - Pre cured patch must have good Material traceability (135A-925-997 & 135A-925-001) and must be processed in accordance with 135A-926-998 with good cure and good DMA.
   - Check for a good SHORE D and DMA for the mixed batch of Epibond adhesive in accordance with 135A-925-985 & 135A-926-994.
   - The consolidation of the repair is good with no voids/discontinuities observed during tap testing of the repaired region.

Based on a successful post repair inspection the part may be released to service.
TYPE 1 OR 2 DAMAGE TO BOTH SIDES OF SANDWICH STRUCTURE  
(BOTH FLAT SURFACE – ALTERNATE CO-CURE REPAIR)

Damaged regions under this section: Co-cured repair is an alternative method to repair the major damage inflicted to both sides of the sandwich structure. This covers both flat and curved surfaces. Flat surface is shown for clarity only.

NOTE

If the rework/repair is done on the outer surface of the fuselage, it is to be smoothed using aerodynamic filler during final painting in accordance with 135A-926-012.

The initial preparation will be done as follows:

- Minimum two Plies of Plain weave (PW) (135A-925-997) will be laid up with same orientation as of the plies removed such that the covering repair patch equals or exceeds the lay-up in the deviant region. Thickness of one cured ply is between 0.0083-0.0089 inches. Repair patch (cured or uncured) must replicate the same number of plies (n) removed plus one i.e. (n+1) from the discrepant/deviation section (For example if 2 plies of carbon PW are present prior to the 3mm /5 mm foam, then the patch repair must be n+1 i.e. 3 plies)
- The Plies need to be staggered such that each ply extends 0.5”-1” beyond the previous ply with the innermost ply of the patch being the smallest.
- Crushed/indented foam core must be removed with chamfered edges 30°-45°.
- Shape and size of the repair foam core must be same as that of the core removed.
- The Patches should be rectangular (rounded edge) or elliptical or circular in shape.

Perform the following procedure to repair the composite

Figure 51-16 Cross Section of Damaged Area

1. Completely grind out the crushed foam core region on the outer and the inner side as shown below:
2. Apply two plies of Hysol EA9696 (0.03psf) Film adhesive (135A-925-992) such that it extends +1" beyond the exposed foam core region as shown:

![Figure 51-17 Damaged Area after Grinding](image)

**Figure 51-17 Damaged Area after Grinding**

3. Lay up at least two plies of Carbon Plain weave (135A-925-997) at +/- 45° such that it covers the film adhesive. (Stagger Plies)

![Figure 51-18 Application of the Film Adhesive](image)

**Figure 51-18 Application of the Film Adhesive**

*The number of plies and orientation shown are for illustration purpose only. The number of plies laid-up (n+1) depends upon the number of plies removed (n).*

4. Apply two plies of Hysol EA9696 (0.03psf) Film adhesive (135A-925-992) such that it extends +1" beyond the exposed foam core region.

![Figure 51-19 Application of Staggered Plies of Plain Weave](image)

**Figure 51-19 Application of Staggered Plies of Plain Weave**

*If damage is accessible from both side, backing fixture/support (Aluminum caul plate or similar) can be used after this step.*
Figure 51-20 Application of Additional Layers of Adhesive Film

5. Create a chamfered 3mm or 5mm Airex foam core piece such that it matches the sanded region and locate it in the repair as shown below

Figure 51-21 Insertion of the New Core

6. Lay up at least two plies of Carbon Plain weave (135A-925-997) at +/- 45° such that it covers the film adhesive. (Stagger Plies)

NOTE

The number of plies and orientation shown are for illustration purpose only. The number of plies laid-up (n+1) depends upon the number of plies removed (n).

Figure 51-22 Application of the Staggered Plies of Plain Weave

7. Vacuum bag the repair and perform a standard AGATE/TCA cure cycle in accordance with 135A-926-998 specifications.

8. Upon completion of the cure check the following aspects are inspected:
   - The cure cycle is within process spec limits (135A-926-998)
   - DMA coupon was created and tested TG passed the requirement detailed within 135A-925-997
   - The consolidation of the repair is good with no void/ discontinuity as observed by inspection procedures in accordance with 135A-911-042

Based on a successful post repair inspection the part may be released to service.
**TYPE 1 OR 2 DAMAGE TO BOTH SIDES OF SANDWICH STRUCTURE (CURVED SURFACE)**

Damaged regions under this section: This type of damage repair will be performed when the damaged region is curved.

If the rework/repair is done on the outer surface of the fuselage, it is to be smoothed using aerodynamic filler during final painting in accordance with 135A-926-012.

The two-ply outer skin and 2-ply inner skin are adhered to the fuselage, thus the ‘cored’ insert will be shorter in overall dimensions such that the two-ply outer skin and inner skin overlay on both the inside and outside of the structures.

Perform the following procedure to repair curved surfaces.

1. Completely grind out the crushed foam core region as follows
2. The ‘cored’ insert must be cut from a lower fuselage with good cure and good DMA.

![Diagram](image1.png)

**Figure 51-26 The Core Insert Ready for Insertion into Damaged Area**

3. This patch is to be bonded using Epibond 1590 adhesive (135A-925-985). The facing sheets (2 ply outer skin and 2-ply inner skin) must overlay the cored insert by at least 1.50" all the way around the cored insert edge. Mechanically apply a distributed load on the back of the patch to ensure the Epibond adhesive joint is 0.015in (+0.025, -0.010) thick in accordance with 135A-926-994-G until handling strength, 6hr per 135A-926-994-H.

![Diagram](image2.png)

**Figure 51-27 Finished Repair Showing the Layers of Plain Weave**

NOTE

*The number of plies and orientation shown are for illustration purpose only. The number of plies laid-up (n+1) depends upon the number of plies removed (n).*

4. After completion of the cure, inspect the following aspects:
   - Pre cured patch must have good Material traceability (135A-925-997 & 135A-925-001) and must be processed in accordance with 135A-926-998 with good cure and good DMA
   - Check for a good SHORE D and DMA for the mixed batch of Epibond adhesive in accordance with 135A-925-985 & 135A-926-994.
   - The consolidation of the repair is good with no voids/discontinuities observed during tap testing of the repaired region.
Based on a successful post repair inspection the part may be released to service.

**NOTE**

The number of plies and orientation shown are for illustration purpose only. The number of plies laid-up (n+1) depends upon the number of plies removed (n).
TYPE 3 SINGLE SKIN REPAIR
ONE SIDE ACCESSIBLE (CO-CURE REPAIR)

Perform the following procedure to repair Type 3 – damage.

If the rework/repair is done on the outer surface of the fuselage, it is to be smoothed using aerodynamic filler during final painting in accordance with 135A-926-012.

The initial preparation is done as follows

- Prior to rework, a visual determination needs to be made to check if there is any damage to the underlying foam core
- It must be ensured that the covering repair patch exceed the lay-up in the deviant region per print post repair and with same orientation of ply or plies removed. Thickness of one cured ply is between 0.0083-0.0089 inches. (For example if 2 plies of carbon PW were removed to get to the foreign object, then the repair patch must have 3 plies of Carbon PW)
- The size of the Repair is to be such that the patch extends minimum +1” all around the deviant region. No damage should be caused to the underlying foam core while performing repair process.
- The Plies need to be staggered such that each ply extends 0.5”-1” beyond the previous ply with the innermost ply of the patch being the smallest.
- The Patch should be rectangular (with rounded edges), elliptical, or circular in shape.

The steps described below illustrate the repair techniques for single skin damage, where the damage is accessible from one side only.

The size of the Repair is to be such that the damage region with the foreign object or Delamination must be completely covered with the patch & extends minimum +1” all around the deviant region.

Perform this procedure to repair the composite surface

1. Carefully grind out the inner/outer ply (or plies) until the damaged area is completely removed as noted below. No damage should be inflicted to the underlying foam core.
Grind region enough to remove damaged surface layers without disturbing the core material.

Figure 51-29 Same Area After Careful Grinding to Remove the Damage but Not the Underlying Core Material

2. Clean and surface prep the outer surface and apply TWO plies of Hysol EA9696 Film adhesive (135A-925-992) such that it extends +1" beyond the repair region.

Figure 51-30 Application of the Adhesive Film

3. Lay up at least 2 plies of Carbon Plain weave (135A-925-997) at +/- 45° such that it covers the film adhesive. (Stagger Plies)

Figure 51-31 Application of the Staggered Layers of Plain Weave

The number of plies and orientation shown are for illustration purpose only. The number of plies laid-up (n+1) depends upon the number of plies removed (n).

4. Vacuum bag the repair and perform a standard AGATE/TCA cure cycle in accordance with 135A-926-998 specifications.

5. After completion of the cure, inspect following aspects.
   - The cure cycle is within process spec limits (135A-926-998)
   - DMA coupon was created and tested TG passed the requirement detailed within 135A-925-997
   - The consolidation of the repair is good with no void/ discontinuity as observed by inspection procedures in accordance with 135A-911-042

   Based on a successful post repair inspection the part may be released to service.
TYPE 3 SINGLE SKIN REPAIR
ONE SIDE ACCESSIBLE (ALTERNATIVE SECONDARY REPAIR)

Perform the following procedure to repair the composite surface.

Fiber Damage Only

1. Carefully grind out the inner/outer ply (or plies) until the damaged area is completely removed as noted below. No damage should be inflicted to the underlying foam core.

   Grind region enough to remove damaged surface layers without disturbing the core material.

2. Clean and prep the surface +1" all around the discrepant/deviant section.

3. Create a secondarily bond of a pre-cured patch of at least 2 ply PW@ +/- 45° using Epibond 1590 adhesive (135A-925-985) such that it extends 1" beyond the discrepant region. Mechanically apply a distributed load on the back of the patch to ensure the Epibond adhesive joint is 0.015in (+0.025, -0.010) thick in accordance with 135A-926-994-G until handling strength, 6hr per 135A-926-994-H.

   Epibond 1590 Adhesive

   Staggered Plies of Plain Weave
The number of plies and orientation shown are for illustration purpose only. The number of plies laid-up \((n+1)\) depends upon the number of plies removed \((n)\).

4. Elevated temperature post cured in accordance with 135A-926-994.

5. After completion of the cure, inspect the following aspects.
   - Pre cured patch must have good Material traceability (135A-925-997 & 135A-925-001) and must be processed in accordance with 135A-926-998 with good cure and good DMA
   - Check for a good SHORE D and DMA for the mixed batch of Epibond adhesive in accordance with 135A-925-985 & 135A-926-994.
   - The consolidation of the repair is good with no voids/discontinuities observed during tap testing of the repaired region.

Based on a successful post repair inspection the part may be released to service.
TYPE 3 SINGLE SKIN REPAIR  
BOTH SIDES ACCESSIBLE (SECONDARY BONDING REPAIR)

This procedure describes the repair of single skin damage when it is readily accessible from both sides. The effective bonding of the repair may be achieved by this process.

Damaged regions under this section: If an outer region is damaged, a satisfactory repair could be made from outside; however, because of the easy accessibility, the repair may be performed from outside against a support that presses hard on the repair until it’s hardened/cured. This method of repair can give a smooth molded finish to the external surface.

Perform this procedure to repair the composite surface.

1. Carefully grind out the inner/outer ply (or plies) the damaged area is completely removed as shown in Figure 51-37.

   Figure 51-36 Cross Section of the Damaged Area
   1. Carefully grind out the inner/outer ply (or plies) the damaged area is completely removed as shown in Figure 51-37.

2. Clean and prep the surface +1” all around the discrepant/deviant section.

   Figure 51-37 Same Area after Grinding the Damaged Layers
   2. Clean and prep the surface +1” all around the discrepant/deviant section.

3. Create a secondarily bond with the pre-cured patch of at least 2 ply PW@+/- 45° with Epibond 1590 adhesive (135A-925-985) such that it extends 1.0 in beyond the repair region. Mechanically apply a distributed load on the back of the patch to ensure the Epibond adhesive joint is 0.015in (+0.025, -0.010) thick in accordance with 135A-926-994-G until handling strength, 6hr per 135A-926-994-H.

   Figure 51-38 Application of the Epibond adhesive and the Staggered Plies of Plain Weave
NOTE

The number of plies and orientation shown are for illustration purpose only. The number of plies laid-up (n+1) depends upon the number of plies removed (n).

4. Elevate the temperature post cured in accordance with 135A-926-994.

5. After completion of the cure, inspect the following aspects:
   - Pre cured patch must have good Material traceability (135A-925-997 & 135A-925-001) and must be processed in accordance with 135A-926-998 with good cure and good DMA
   - Check for a good SHORE D and DMA for the mixed batch of Epibond adhesive in accordance with 135A-925-985 & 135A-926-994.
   - The consolidation of the repair is good with no voids/discontinuities observed during tap testing of the repaired region.

Based on a successful post repair inspection the part may be released to service.
Section 51-15 Adhesive Joint Structural Repairs:

The fuselage of the Liberty XL-2 is composed of structural composite materials. The fuselage aft of the engine cowling is fabricated from carbon fiber reinforced fabrics that are used as facing plies adhered to core materials to form a structural sandwich. The main load bearing members of this structure are secondarily bonded together using an epoxy paste adhesive. Care must be taken when performing repairs to the fuselage adhesive joints.

NOTE

Modifications to the fuselage structure including adhesive joints that are performed on form 337 must be coordinated with Liberty Aerospace Inc. Customer Support.
Section 51-16 Adhesive Joint Damage Classification:

Adhesive joint damage classifications are defined by the following table. These classifications are used solely to define allowable ‘bondline’ defects.

Bond defects include:

- Disbonds (separation between bonded adherents)
- Inclusions (foreign matter in bondline)
- Porosity (frequent small voids in bondline)
- Lack of adhesive (visible due to no squeeze-out)

When there is a doubt regarding classifying a potential defect as a defect, it should be treated as a defect. For example, if a bondline has several air bubbles but it is not clear if it should be treated as a porosity defect, treat it as a porosity defect.

<table>
<thead>
<tr>
<th>Bond Classification</th>
<th>Maximum Cumulative Defect Size</th>
<th>Defect Accumulation Threshold (DAT)</th>
</tr>
</thead>
<tbody>
<tr>
<td>I</td>
<td>1.0 in²</td>
<td>12.0 linear inches</td>
</tr>
<tr>
<td>II</td>
<td>2.0 in²</td>
<td>12.0 linear inches</td>
</tr>
<tr>
<td>III</td>
<td>3.0 in²</td>
<td>12.0 linear inches</td>
</tr>
<tr>
<td>IV</td>
<td>4.0 in²</td>
<td>12.0 linear inches</td>
</tr>
<tr>
<td>Reserved</td>
<td>Reserved</td>
<td>Reserved</td>
</tr>
</tbody>
</table>

Table 51-6 Bond Defect Classifications

Bond defects are cumulative within the DAT (Defect Accumulation Threshold). In other words, if 1.0 in² of defect is allowed and the DAT is 12.0 linear inches, then all bond defects within a given 12 inch length of bondline must be added.

Figure 51-39 Example Application of Bondline DAT
Figure 51-39 above shows how the DAT is used to determine cumulative defects. After all defects are identified, measure out the DAT on either side of each defect to determine what other defects must be treated as cumulative. Damage should therefore be recorded by the maintenance facility as Type 1, Class II etc.

![Diagram of the airplane structure]

Figure 51-40 The Bond-Line Locations (in Gray) On The Airplane’s Fuselage

The table below gives the bond inspection classifications to be used for determining the bondline defects.

<table>
<thead>
<tr>
<th>Bond Classification</th>
<th>Minimum Required Inspection</th>
</tr>
</thead>
<tbody>
<tr>
<td>NS</td>
<td>Inspection type not specified</td>
</tr>
<tr>
<td>A</td>
<td>100% visual + 100% NDI ultrasound</td>
</tr>
<tr>
<td>B</td>
<td>100% visual + 100% NDI tap testing</td>
</tr>
<tr>
<td>C</td>
<td>100% visual only</td>
</tr>
<tr>
<td>Reserved</td>
<td>Reserved</td>
</tr>
</tbody>
</table>

Table 51-7 Bond Inspection Classifications

**NOTE**

The classification “NS” means that the inspection method is not specified. Any practical method(s) may be used if the method chosen demonstrates that the acceptable defect
criterion may be met for the component.

“Visual” means optical. Actual inspection may be done via the naked eye, a bore scope, or other imaging means.

This document does not define detailed procedures for specified inspection methods.

A component may always be inspected more intensively than its classification requires. For example, if a bond is classified as “B” (visual and tap test), it may still be inspected with ultrasound testing, or other means in addition to the visual inspection.

Section 16-01  Type 1- Damage:
Type 1-damage is defined as critical damage inflicted to primary or secondary structural joints at locations such as highly stressed (contact Liberty Aerospace Customer support for clarification) regions and underlying structural elements.

Damage is classified as Type 1 when the size, location, and number of damages per unit length or area endanger the structural integrity of the aircraft.

This type of damage requires partial or complete reconstruction of parts or repairs of large areas.

Section 16-02  Type 2- Damage:
Type 2-damage is defined as damage inflicted to primary or secondary structure involving complete penetration of the joint materials.

Damage is classified as Type 2 if it has the potential to affect the structural integrity of the aircraft in flight.

Section 16-03  Type 3- Damage:
Type 3-damage is defined as damage limited to outer skin only (no damage to internal facing plies or adhesive).

Damages are classified Type 3 when the size, location and number of damage per unit length/area does not endanger the structural integrity of the aircraft.

Section 16-04  Type 4- Damage:
Type 4-damage is defined as damage that is inflicted to parts of minimal structural importance.

Type 4 damage includes light surface erosion, scratches, grooves, small dents, etc. that do not penetrate the composite outer skin. This includes damage to replaceable access covers, etc.

Rework to the outer surface of the fuselage can be smoothed using aerodynamic filler during final painting in accordance with 135A-926-012.
**Section 51-17 Adhesive Joint Inspection Criterion:**

All Liberty XL-2 adhesive joints must meet the requirements of the applicable engineering drawing and material and process specifications.

In addition, bonds must be inspected for defects according to their classification as presented in the table below. The classification defines both the inspection method required and the standard acceptable defect sizes and defect frequencies.

<table>
<thead>
<tr>
<th>Bond</th>
<th>Classification</th>
</tr>
</thead>
<tbody>
<tr>
<td>Upper to lower fuselage bond</td>
<td>II-B</td>
</tr>
<tr>
<td>Roll Over Hoop to upper fuselage bond</td>
<td>II-B</td>
</tr>
<tr>
<td>Bulkhead to fuselage bond</td>
<td>II-B</td>
</tr>
<tr>
<td>Windscreen to fuselage bond</td>
<td>II-B</td>
</tr>
<tr>
<td>Door transparency to door frame bond</td>
<td>II-B</td>
</tr>
<tr>
<td>Vertical stabilizer skin to rib bonds</td>
<td>II-B</td>
</tr>
<tr>
<td>Vertical stabilizer skin to spar bonds</td>
<td>II-B</td>
</tr>
<tr>
<td>Vertical stabilizer closeout spar bond</td>
<td>II-B</td>
</tr>
</tbody>
</table>

**Table 51-8 FUSELAGE BOND CLASSIFICATIONS**

For the purposes of this inspection, bonds shall be classified by combining the defect and inspection classifications listed in the previous sections. For example, a bond may be classified as "II-B", or "I-NS".

The liberty fuselage has a Bond Classification of “B”, as listed in the table above. Qualified personnel using a coin (penny) to find out any damage or defect (hollow sound) can also perform tap testing as defined in Section 51-13 - Laminated Composites Inspection Criterion on page 17 of this chapter. The tester will tap an area small or large enough to ensure that no voids or disbondings exist in the tested component. A void or disbonding is heard to have a “dull” or empty sound. To ensure the area is in fact a void or disbond, testing of the surrounding area is recommended.

**NOTE**

The classification “NS” means that the inspection method is not specified. Any practical method(s) may be used if the method chosen demonstrates that the acceptable defect criterion may be met for the component.

“Visual” means optical. Actual inspection may be done via the naked eye, a bore scope, or other imaging means.

This document does not define detailed procedures for specified inspection methods.

A component may always be inspected more intensively than its classification requires. For example, if a bond is classified as “B” (visual and tap test), it may still be inspected with ultrasound testing, or other means in addition to the visual inspection.
The specified inspection type is only the initial inspection required. If defects are found, the defects must be thoroughly defined using whatever inspection means are required to do so. For example, if tap testing is required and a defect is found, it may be necessary to use ultrasound to define the defect.

**Section 17-01   Tap Testing:**

The aircraft fuselage and all adhesive joints have been inspected prior to release, the methods used were visual and tap tests. At the time of aircraft release no ultrasound/radiographic testing was performed, but should be considered, as an alternate means of evaluation if doubt exists as to whether any disbonding exist in structures that require repair.
Section 51-18 Adhesive Joint Repair Methods:

Below is a general representation of the laminates/Sandwich adhesive joint. Refer to this orientation when reading next sections.

![Diagram](image.png)

**Figure 51-41 Representation of the Laminate/ Sandwich adhesive joint**

Section 18-01 Type 1 Damage:

Type 1-damage is defined as critical damage inflicted to primary or secondary structural joints at locations such as highly stressed regions and underlying structural elements.

Type-1 damage primarily consists of regions where:

- Sandwich structure/laminate-fuselage skin disbond
- Upper and lower fuselage bondline disbond

Section 18-02 Type 2 Damage:

Type-2 damage is defined as damage inflicted to primary or secondary structure involving complete penetration of the adhesive joint. Repair methodologies should be consistent with those employed for Type 1-DAMAGE above.

Section 18-03 Type 3 Damage:

Type-3 damage is defined as no damage to the fuselage structure. Type-3 damage is limited to presence of a foreign object and of porosity in the Epibond adhesive.

**NOTE**

This repair work is performed from inside of the fuselage (IML).

Section 18-04 Type 4- Damage:

Type 4-damage is defined as damage that is inflicted to parts of minimal structural importance.
TYPE-1 OR 2 DISBOND OF ADHESIVE JOINT

Perform this procedure to repair an adhesive joint.

6. This is a general representation of the bonds described earlier under fuselage bond classification.

![Figure 51-42](image)

7. Clean and surface prep the disbond along the bondline, in accordance with 135A-926-994. All adhesive material needs to be removed down to the bare carbon. Prep the bare carbon material at the removed Epibond adhesive section for good surface adhesion of Epibond adhesive.

Grind the layer of Plain Weave enough to remove the area that has become dis-bonded.

![Figure 51-43](image)


**NOTE**

Efficient bonding of the substrate and Epibond adhesive depends upon clean, moisture free and good prep surface. Underlying carbon ply must NOT be damaged during surface prepping.
9. Secondarily bond a 4-ply reinforcement patch of Carbon Plain Weave (135A-925-997), using Epibond 1590 adhesive in accordance with 135A-926-994. The patch should be elliptical or rectangular (rounded edges) and should cover the disbond and +1in all around. The repair patch should be laid with orientation of ±45 / 0-90 / 0-90 / ±45.

![Figure 51-44](image)

**Figure 51-44**

10. Mechanically apply a distributed load on the back of the patch to ensure the Epibond adhesive joint is 0.015in (+0.025, -0.010) thick in accordance with 135A-926-994 until handling strength, 6hr per 135A-926-994.

![Figure 51-45](image)

**Figure 51-45**

11. Elevate temperatures post cured in accordance with 135A-926-994.

12. When the repair work is done from outside, the surface of, the fuselage is to be smoothed using aerodynamic filler during final painting in accordance with 135A-926-012.

13. Upon completion of the cure check the following aspects are inspected:
   - Pre cured patch must have good Material traceability (135A-925-997 & 135A-925-001) and must be processed in accordance with 135A-926-998 with good cure and good DMA
   - Check for a good SHORE D and DMA for the mixed batch of Epibond adhesive in accordance with 135A-925-985 & 135A-926-994.
   - The consolidation of the repair is good with no voids/discontinuities observed during tap testing of the repaired region.

*NOTE*

Flush the outer repair patch to blend it with the profile of the Liberty XL-2.
TYPE-1 OR 2 REPAIR TO BOTH SIDES OF ADHESIVE JOINT
LAMINATE TO LAMINATE

Perform this procedure to repair both sides of an adhesive joint

1. Original Configuration:

This is a general representation of the bonds described earlier under fuselage bond classification.

![Figure 51-46](image)

Grind the layer of Plain Weave enough to remove the area that has become dis-bonded.

![Figure 51-47](image)

2. Clean and prep the discrepancy along the bondline on both inner and outer surfaces, in accordance with 135A-926-994. All adhesive material needs to be removed down to the bare carbon. Also, prep the bare carbon material at the removed Epibond adhesive section.


4. Secondarily bond a 4-ply reinforcement patch of Carbon Plain Weave (135A-925-997), using Epibond 1590 adhesive in accordance with 135A-926-994. The patch should be elliptical or rectangular with rounded edges and should cover the disbond and 1.0 inches on all sides. In addition, it should be laid in the same schedule and orientation of the Bond Line strap i.e. ±45 / 0-90 / 0-90 / ±45.
5. Mechanically apply a distributed load on the back of the patch to ensure the Epibond adhesive joint is 0.015in (+0.025, -0.010) thick in accordance with 135A-926-994 until handling strength, 6hr per 135A-926-994.


7. The outer surface of the fuselage is to be smoothed using aerodynamic filler during final painting in accordance with 135A-926-012.

8. After completion of the cure, inspect the following aspects:
   - Pre cured patch must have good Material traceability (135A-925-997 & 135A-925-001) and must be processed in accordance with 135A-926-998 with good cure and good DMA
   - Check for a good SHORE D and DMA for the mixed batch of Epibond adhesive in accordance with 135A-925-985 & 135A-926-994.
   - The consolidation of the repair is good with no voids/discontinuities observed during tap testing of the repaired region.
TYPE-1 OR 2 REPAIR OF ADHESIVE JOINT
SANDWICH TO SANDWICH

The sandwich-to-sandwich joints in Liberty XL-2 can be observed in the area of Fin Spar where fin ribs are joined to the spar by means of adhesive and fin spar is joined to the upper fuselage by means of adhesive. Any discrepancy with the adhesive joint is hard to access in such areas. Hence, make a through circular hole on a non-discrepant surface near to the discrepant area so that this joint can be made accessible for work and inspection.

![Figure 51-48 Original Configuration:](image)

Record the number of plies removed. One (01) cured ply thickness of carbon plain weave is 0.0083in-0.0089in. Repair patch (cured or uncured) must replicate the same number of plies (n) removed plus one (n+1) from the discrepant/deviation section (For example if 2 plies of carbon PW are present prior to the 3mm /5 mm foam, then the patch repair must be n+1 or 3 plies)

The size of the Repair is to be such that the patch extends minimum 1” all around the deviant region.

The Plies need to be staggered such that each ply extends 0.5”-1” beyond the previous ply with the innermost ply of the patch being the smallest.

Shape and size of the repair foam core must be same as that of the core removed.

The Patches should be rectangular (rounded edge) or elliptical or circular in shape.

Perform this procedure to repair on the adhesive joint once the joint is made accessible.

1. Locate the effected area. If necessary, an access hole can be cut in the fuselage, only large enough to complete the task.
2. Clean and surface prep the discrepancy along the bondline, in accordance with 135A-926-994-G. All adhesive material needs to be removed down to the bare carbon. Also, prep the bare carbon material at the removed Epibond adhesive section.


4. Secondarily bond a 4-ply patch of Carbon Plain Weave (135A-925-997), using Epibond 1590 adhesive in accordance with 135A-926-994. The patch should be elliptical or rectangular (rounded edges) and should cover the disbond and +1.0 in all around. The repair patch should be laid in the orientation of ±45 / 0-90 / 0-90 / ±45.
5. Mechanically apply a distributed load on the back of the patch to ensure the Epibond adhesive joint is 0.015in (+0.025, -0.010) thick in accordance with 135A-926-994 until handling strength, 6hr per 135A-926-994.


7. After completion of the cure, inspect the following aspects:
   - Pre cured patch must have good Material traceability (135A-925-997 & 135A-925-001) and must be processed in accordance with 135A-926-998 with good cure and good DMA
   - Check for a good SHORE D and DMA for the mixed batch of Epibond adhesive in accordance with 135A-925-985 & 135A-926-994.
   - The consolidation of the repair is good with no voids/discontinuities observed during tap testing of the repaired region.

8. Once the adhesive repair is complete, repair any hole that used to access the area. Refer to the following procedure Type 1 or 2 Damage To Both Sides Of Sandwich Structure (Flat Surface)-Secondary Bonding on page 26 of this chapter.


10. The outer surface of the fuselage is to be smoothed using aerodynamic filler during final painting in accordance with 135A-926-012

11. After completion of the cure, inspect the following aspects:
   - Pre cured patch must have good Material traceability (135A-925-997 & 135A-925-001) and must be processed in accordance with 135A-926-998 with good cure and good DMA
   - Check for a good SHORE D and DMA for the mixed batch of Epibond adhesive in accordance with 135A-925-985 & 135A-926-994.
   - The consolidation of the repair is good with no voids/discontinuities observed during tap testing of the repaired region.

Based on a successful post repair inspection the part may be released to service.
**TYPE-3 DAMAGE REPAIR PROCEDURE**

Below are mentioned two minor damages defined under type-3 in other words foreign object deposition and porosity. Lack of adhesive also falls under this category. Foreign objects can be visually inspected, but tap testing is performed to confirm the porosity present in an adhesive.

**NOTE**

*The repair for single damage (foreign object or porosity) may also be performed as steps defined below.*

Perform this procedure to repair type 3 damage.

1. **Original Configuration:**

2. Remove the foreign object completely. Clean and surface prep the area along the discrepancy (around foreign object and porosity) in accordance with 135A-926-994. All adhesive material needs to be removed down to the bare carbon. Also, prep the bare carbon material at the removed Epibond adhesive (porosity) section.

![Figure 51-53](image)

Efficient bonding of the substrate and Epibond adhesive depends upon clean, moisture free and good prep surface. Underlying carbon ply must not be damaged during surface prepping.


4. Secondarily bond a 4-ply reinforcement patch of Carbon Plain Weave (135A-925-997), using Epibond 1590 adhesive in accordance with 135A-926-994. The patch should be elliptical or rectangular (rounded edges) and should cover the foreign object and porosity +1.0 in all directions. The repair patch should be laid in the following sequence i.e. ±45 / 0-90 / 0-90 / ±45.
5. Mechanically apply a distributed load on the back of the patch to ensure the Epibond adhesive joint is 0.015in (+0.025, -0.010) thick in accordance with 135A-926-994 until handling strength, 6hr per 135A-926-994.

6. Elevated temperature post cure the repair in accordance with 135A-926-994.

7. Upon completion of the cure check the following aspects are inspected:
   - Pre cured patch must have good Material traceability (135A-925-997 & 135A-925-001) and must be processed in accordance with 135A-926-998 with good cure and good DMA
   - Check for a good SHORE D and DMA for the mixed batch of Epibond adhesive in accordance with 135A-925-985 & 135A-926-994.
   - The consolidation of the repair is good with no voids/discontinuities observed during tap testing of the repaired region.

Based on a successful post repair inspection the part may be released to service.
Section 51-20 Minor Structural Metallic Repairs

Minor repairs to the aluminum flying surfaces and controls may be carried out using standard materials and techniques in accordance with FAA Advisory Circular 43.13-1B. These repairs should be limited to patching of small holes < 0.1in or tears in aluminum skins, replacement of moving parts such as hinges, etc. Any repairs involving damage to underlying structural components of flying surfaces (significant skin damage, any damage to underlying structures such as spars, ribs, stringers, etc.) should be referred to Liberty Aerospace Inc Customer Support.

The wings and flying surfaces of the Liberty XL-2 is composed of structural aluminum alloys. These materials are riveted or mechanically fastened to form structural stringer stiffened structures.

NOTE

Major repairs on aileron, flap, horizontal stabilizer, tab and rudder surfaces is by replacement. Removal and replacement or mass balanced flight controls requires flight control rebalancing in accordance with Section 51-60 Control Surfaces Balancing on page 63 of this chapter. Major repairs to the fixed, non-moving, portion of the wing surface must be coordinated with Liberty Aerospace Inc Customer Support.

The chassis and engine mount frame of the Liberty XL-2 is composed of carbon steel tubing materials. These materials are welded to form structural trusses.

Section 20-01 General Corrosion Inspection And Metal Component Protection:

All metal parts of the airplane have been inspected and corrosion protected prior to leaving the manufacturing facility. Aluminum components have been treated with Alodine® EC²™ Electro-Ceramic Coating, a corrosion inhibitor, a paint primer, and paint. Steel components have been treated with Zinc Chromate a rust inhibitor, a paint primer, and paint. Any repair to metal components should be made by qualified personnel in an environment where once paint, primer, and the corrosion/rust inhibitor have been removed the exposed metal will not incur prolonged exposure to the elements to incur more damage.

NOTE

When applying Alodine or Zinc Chromate, care should be taken, as these chemicals are hazardous to humans and the environment in liquid form.
**METAL COMPONENT INSPECTION FOR CORROSION OR RUST:**

Metal components of the airplane include the rolling chassis, engine mount frame, and wing and tail plane surfaces. The rolling chassis and the engine mount frame are constructed of steel, and the wing and tail plane surfaces are aluminum. These components should be inspected for signs of rust or corrosion.

1. Visually inspect all painted metal surfaces for cracking, pitting, or corrosion or rust.
2. Corrosion or rust will appear as a bulge or lifting of the painted surface.
3. Corrosion will appear as a white or light grey chalky powder.
4. Rust will have a reddish brown appearance and appear grainy.
5. Repair rust or corrosion as stated below.
STRUCTURAL ALUMINUM REPAIRS:

Perform this procedure to effect repairs to the aluminum structure.

All precautions should be taken as to local requirements for the handling of chemicals and the disposal of chemically soaked rags, wiping cloths, or materials used in the preparation and repair of the surface.

1. Identify the area to be repaired, isolating it from surrounding areas.
2. Remove paint down to a bare metal surface.
3. Remove any foreign materials.
4. Re-treat the exposed metal surface with Alodine in accordance with the product manufacturer handling procedure.
5. Re-prime the surface for painting.
6. Paint the surface and blend to match the surrounding area.

NOTE

Structural repair for aluminum fuel tanks is mention in chapter 28.
STRUCTURAL STEEL REPAIRS

Perform this procedure to effect repairs to the steel structure.

All precautions should be taken as to local requirements for the handling of chemicals and the disposal of chemically soaked rags, wiping cloths, or materials used in the preparation and repair of the surface.

1. Identify the area to be repaired, isolating it from surrounding areas.
2. Remove paint down to a bare metal surface.
3. Remove any foreign materials.
4. Re-treat the exposed metal surface with Zinc Chromate in accordance with the product manufacturer handling procedure.
5. Re-prime the surface for painting.
6. Paint the surface and blend to match the surrounding area.
7. The repair methods prescribed by FAA Advisory Circular 43.13-1B must be used to prevent and repair corrosion.

If tube wall thickness is reduced in diameter by 0.005 inches due to corrosion, or is visibly marred by corrosion or damage, or leads to free-play (movement) between adjacent mating components (for example: tail-plane surface to torque tube interface, or torque tube to fuselage bearings), Liberty Customer support should be contacted such that specific structural repair procedures can be provided.
Section 51-60 Control Surfaces Balancing

This section details the information on balancing the aileron and rudder control surfaces. If the balance on the horizontal stabilizer is suspect, contact Liberty Aerospace, Inc. Customer Service.

Section 60-01 Mass Balance Vertical Clamp Jig

The aileron and rudder mass balance procedure call for a vertical clamp jig. This jig is fabricated locally as needed for these procedures. The jig is made from a 2X4 mounted to the side of a table or bench. Mounting plates attach to the 2X4, and either the aileron or rudder attach to the plates. Once attach, the aileron or rudder should be able to move freely on its hinge. Figure 51-55 shows the mass balance vertical test jig.

Section 60-02 Balancing Procedures

This section details the procedures to balance the aileron, rudder and horizontal stabilizer control surfaces.
AILERON BALANCING

Perform this procedure to check and correct the balance of the aileron. The vertical clamp jig referred to in this procedure is any method that will allow the aileron to swing freely.

NOTE

This procedure can not be done with the aileron mounted on the airplane.

1. If the aileron is mounted to the airplane, remove the aileron as shown in Chapter 27 – Control Surfaces.
2. Thoroughly clean all surfaces of the aileron.
3. Hold the two hinges in a vertical clamp jig.
4. Lubricate the hinges so that they are free to operate.
5. Operate the aileron in the vertical clamp jig to verify that the hinges are in the same plane and operate about the collinear axis without binding or galling through their full range of movement.
6. The weight of the mass balance is engineered to be heavy on the balance arm, nose heavy (NH).
7. Locate a load cell (such as a Dillon/Quality Plus, Inc. Force Gauge CFG-50N) under one of the mass balance, 5.0 in from the hinge line. The load recorded on the load cell to bring the lower surface of the aileron horizontal must be between 0 and 0.05 lb. For details, see Figure 51-56.
8. If the load recorded is higher than 0.05 lb, weight must be subtracted by sanding from both mass balances.

9. In case the aileron is trailing edge heavy, check the surfaces (internal and external) of the aileron for foreign material or other matter. Also, if the aileron has been painted (since leaving manufacturing) remove all finishes (back to the original paint on the aileron) and clean. Repeat step 7 above.

10. Re-attach the surface to the wing, rig assembly per Chapter 27 – Control Surfaces.
RUDDER BALANCING

Perform this procedure to check and inspect the mass balance on the rudder. The vertical clamp jig referred to in this procedure is any method that will allow the rudder to swing freely.

1. If the rudder surface is attached to the vertical stabilizer, remove the rudder from the vertical stabilizer. See Chapter 27 – Control Surfaces.

2. Thoroughly clean all surfaces of the rudder.

3. Attach the test article to the vertical clamp jig.

4. Lubricate the hinge so that it is free to operate.

5. Operate the rudder in the vertical clamp jig to verify that the hinges are in the same plane operate about the collinear axis without binding or galling through its full range of movement.

6. To measure balance, use a load cell positioned beneath the rudder surface root, adjacent to the drive, at a point 16.0 in aft of, and perpendicular to, the hinge center. This point lies almost coincident with the location of the rudder skin to rudder cap joint.

7. Record the load required to hold rudder level.

8. If a load of 1.50 to 1.56 lb is noted, proceed to step 14 below.

Figure 51-57 Location of Load Cell on Rudder

6. To measure balance, use a load cell positioned beneath the rudder surface root, adjacent to the drive, at a point 16.0 in aft of, and perpendicular to, the hinge center. This point lies almost coincident with the location of the rudder skin to rudder cap joint.

7. Record the load required to hold rudder level.

8. If a load of 1.50 to 1.56 lb is noted, proceed to step 14 below.
The intent is to verify that the rudder hinge moment lies between 24 in-lbs and 25 in-lbs.

9. If the load is outside of the range of 1.5 to 1.56 lbs., check all surfaces (internal and external) for foreign material or other matter. Re-clean the surfaces of the rudder and go back to step 6 above.

10. If the upward load is more than 1.56lb, it will require the removal of the rudder horn from the rudder assembly. Remove the rudder horn from the rudder assembly by de-riveting. Gain access to the mass balance inside the rudder horn and add mass by means of adding washers (NAS 1149FO 332P or equivalent), as needed, to the mass balance mounting screws. If washer stack exceeds safe thread length of screws, increase screw length as required. Install the rudder horn temporarily to the rudder assembly, and repeat step 6 above.

11. If the upward load is less than 1.5 lb, it will require the removal of the rudder horn from the rudder assembly. Remove the rudder horn from the rudder assembly by de-riveting. Gain access to the mass balance inside the rudder horn and remove washers, as needed for required weight, from mass balance mounting screws. It may be necessary to remove/shave off a portion of the mass balance (from the aft side). Fine adjustment may be made on the assembled rudder by drilling the Mass Balance – Horn (135A-30-643) thru the 0.201in tooling hole using a #7-drill bit. If removed from the rudder, install the rudder horn temporarily to the rudder assembly, and repeat step 6 above.

12. After completing step 9 or 11, permanently attach the rudder horn to the rudder assembly using rivets.

13. After assembling the rudder horn and rudder together (permanently), repeat steps 3 through 11 until rudder is made level within the stated tolerance.

14. Attach the rudder to the vertical stabilizer.

15. Check the rigging; see Chapter 27 – Control Surfaces.

This completes the Rudder Balancing procedure.
HORIZONTAL STABILIZER BALANCING

At this current time, Liberty Aerospace, Inc. has not defined a field procedure to check the balance on the horizontal stabilizer. If the balance on the horizontal stabilizer is suspect, contact Liberty Aerospace, Inc. Customer Service.
Section 51-80 Ground and Bonding

This section details the information on grounding and bonding on the airplane that are specific to the Liberty XL-2 airplane.

Section 80-01 Ground Studs

Ground studs are permanently installed on the airplane metallic structure and shall be treated as permanent bonds. See Figure 51-58.

![Ground Studs Diagram](image)

Figure 51-58 Ground Studs

Section 80-02 Bulkhead Connectors

Bulkhead connectors that are used for termination of cable shields shall be bonded to the airplane structure with a maximum resistance of 0.003 ohm. See Figure 51-59.

![Bulkhead Connectors Diagram](image)

Figure 51-59 Bulkhead Connectors
Section 80-03  Metallic Pipes, Tubes and Hoses

Metallic pipes, tubes, hoses and etc. that carry fluids in motion shall be bonded to structure as shown in Figure 51-60.

![Figure 51-60 Bonding to a Metal Pipe/Tube/Hose](image)

Section 80-04  Carbon Fiber Composite to Aluminum Bonds

Aluminum components cannot be placed in direct contact with Carbon Fiber Composites (CFC) components due to corrosion that occurs with such contact. Direct contact between aluminum and CFC components shall be prevented by the use of insulation (i.e. fiberglass, Mylar, paint, etc.) between these components. Bonding between these materials shall be performed by use of Corrosion Resistant (CRES) steel or titanium fasteners between the components. See Figure 51-61 for details.

![Figure 51-61 Composite to Metal Bonding](image)

Section 80-05  Antenna Bonds

Antenna mounting plates shall be mounted to the carbon fiber composite skins to enhance the ground plane and for structural support. Refer to Figure 51-62 for antenna installation.
Section 80-06  Inspection Procedures for Grounding and Bonding

This section details the procedures for checking and inspecting the airplane’s bonding and grounding systems.

Figure 51-62 Antenna Bounding
GROUND AND BONDING INSPECTION

Perform this procedure to check and inspect the airplane’s ground and bonding system. Although a standard digital voltmeter will suffice, Liberty Aerospace, Inc. recommends the use of a calibrated milliohm meter.

1. Remove the belly panel and engine cowlings and inspection covers. See Chapter 53 – Fuselage.

2. Open inspection covers in the wings. See Chapter 57 – Wings.

3. Refer to Figure 51-63 for Test points.

4. Refer to Table 51-9 for testing points. Use copies of the table during the inspection process.

5. Test each checkpoint to the nearest chassis ground point.

6. Indicate the resistance reading in the table.

Figure 51-63 Diagram of the Ground Checkpoints for Testing

4. Refer to Table 51-9 for testing points. Use copies of the table during the inspection process.

5. Test each checkpoint to the nearest chassis ground point.

6. Indicate the resistance reading in the table.
<table>
<thead>
<tr>
<th>Checkpoint</th>
<th>Location</th>
<th>Maximum Value</th>
<th>Value Received</th>
<th>Pass?</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>Starboard MPC Ground Bolt</td>
<td>&lt; 0.5 Ω</td>
<td>Ω</td>
<td></td>
</tr>
<tr>
<td>2</td>
<td>Port MPC Ground Bolt</td>
<td>&lt; 0.5 Ω</td>
<td>Ω</td>
<td></td>
</tr>
<tr>
<td>3</td>
<td>Stainless Steel Firewall</td>
<td>&lt; 0.5 Ω</td>
<td>Ω</td>
<td></td>
</tr>
<tr>
<td>4</td>
<td>Chassis Grounding Tab</td>
<td>&lt; 0.5 Ω</td>
<td>Ω</td>
<td></td>
</tr>
<tr>
<td>5</td>
<td>Engine Mount Ground Tab</td>
<td>&lt; 1.0 Ω</td>
<td>Ω</td>
<td></td>
</tr>
<tr>
<td>6</td>
<td>Engine Block</td>
<td>&lt; 1.0 Ω</td>
<td>Ω</td>
<td></td>
</tr>
<tr>
<td>7</td>
<td>Exhaust Pipe</td>
<td>&lt; 1.0 Ω</td>
<td>Ω</td>
<td></td>
</tr>
<tr>
<td>8</td>
<td>Starboard and Port Wing Tie-downs</td>
<td>&lt; 1.0 Ω</td>
<td>Ω</td>
<td></td>
</tr>
<tr>
<td>10</td>
<td>Fuel Line Between Gascolator and Fuel Pump</td>
<td>&lt; 1.0 Ω</td>
<td>Ω</td>
<td></td>
</tr>
<tr>
<td>11A</td>
<td>Aluminum Panel: Circuit Breakers</td>
<td>&lt; 1.0 Ω</td>
<td>Ω</td>
<td></td>
</tr>
<tr>
<td>11B</td>
<td>Aluminum Panel: Avionics</td>
<td>&lt; 1.0 Ω</td>
<td>Ω</td>
<td></td>
</tr>
<tr>
<td>11C</td>
<td>Aluminum Panel: Instruments</td>
<td>&lt; 1.0 Ω</td>
<td>Ω</td>
<td></td>
</tr>
<tr>
<td>11D</td>
<td>Aluminum Panel: Console</td>
<td>&lt; 1.0 Ω</td>
<td>Ω</td>
<td></td>
</tr>
<tr>
<td>12</td>
<td>Fuel Tank</td>
<td>&lt; 1.0 Ω</td>
<td>Ω</td>
<td></td>
</tr>
<tr>
<td>13</td>
<td>Fuel Line Between Tank and Check Valve</td>
<td>&lt; 1.0 Ω</td>
<td>Ω</td>
<td></td>
</tr>
<tr>
<td>14</td>
<td>Verify resistance between refueling inlet and composite fuselage and between refueling inlet and chassis</td>
<td>&lt; 5.0 Ω</td>
<td>Ω</td>
<td></td>
</tr>
<tr>
<td>15</td>
<td>Fuel Vent Line on Fuel Tank</td>
<td>&lt; 1.0 Ω</td>
<td>Ω</td>
<td></td>
</tr>
<tr>
<td>16</td>
<td>Camloc Fastener on Upper Cowl</td>
<td>&lt; 5.0 Ω</td>
<td>Ω</td>
<td></td>
</tr>
<tr>
<td>17</td>
<td>VOR Antenna Cover Mounting Screw</td>
<td>&lt; 5.0 Ω</td>
<td>Ω</td>
<td></td>
</tr>
<tr>
<td>18</td>
<td>Verify resistance between rudder pedal control circuit and chassis across braided connectors</td>
<td>&lt; 0.10 Ω</td>
<td>Ω</td>
<td></td>
</tr>
<tr>
<td>19</td>
<td>Verify resistance between aileron and elevator control circuit and chassis across braided connectors</td>
<td>&lt; 0.10 Ω</td>
<td>Ω</td>
<td></td>
</tr>
</tbody>
</table>

Table 51-9 Table of Ground Checkpoints for Testing