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New Jersey 08405

FAA Composite Inspector Training Course to Enhance Proficiency and Improve Reliability

June 2019

Final Report

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16. Abstract

The FAA Composite Inspector Training Course is composed of a classroom lecture portion and a set of hands-on student exercises meant to teach and reinforce best practices for inspecting composite laminate structures. The information provided in each chapter in this report is intended to provide background to aid classroom lectures and student exercises associated with the FAA Composite Inspector Training Course. The Airworthiness Assurance NDI Validation Center (AANC), with input from industry and the Inspection Task Group of the Commercial Aircraft Composite Repair Committee chaired by the AANC, developed training materials for the Composite Inspector Training Course. Modules, including an introduction to composite materials, composite nondestructive inspection (NDI) theory and practice, special cases, and lessons learned, have been produced in addition to various hands-on NDI exercises. A set of proficiency specimens containing realistic composite structures and representative damage was designed to reinforce teaching points of the course and test inspectors' proficiency. Extensive details of the course modules, hand-on exercises, and the proficiency specimens are all presented in this report.

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LIST OF ACRONYMS

AANC	Airworthiness Assurance NDI Validation Center
NDI	Nondestructive inspection
OEM	Original equipment manufacturer(s)
PA UT	Phased array ultrasonic transmission
PE UT	Pulse echo ultrasonic transmission
POD	Probability of flaw detection
RF	Radio frequency
TCG	Time-corrected gain
TTU	Through-transmission ultrasonic
VARTM	Vacuum-assisted resin transfer molding

EXECUTIVE SUMMARY

Carbon fiber reinforced plastic is an essential material in the structural design and construction of modern commercial aircraft. This is evident in the cutting-edge Boeing 787 and Airbus A350, which are comprised of more than half solid laminate composites by weight. Inspection procedures, technology, and deployment for solid laminate composite structures differ markedly from traditional metallic structures. In addition, today's nondestructive inspection (NDI) personnel have little or no exposure to the demands of composite inspections. The airlines face a significant challenge in how to develop and deliver appropriate training to personnel performing NDI on these modern aircraft. To meet this need, the FAA initiated the development of a comprehensive composite laminate training course. The course, which was developed by the FAA Airworthiness Assurance Center at Sandia Labs, aims to provide the airlines with the information necessary to adequately train their inspectors.

This report provides the course content for the Composite Inspector Training Course. Development of the class that began in late 2014; it was completed and was sent to more than 20 airlines in the early part of 2017. The training package consists of six classroom training lecture modules and the drawings and specifications needed to fabricate the accompanying set of hands-on training aids, or NDI proficiency specimens. The classroom modules include an introduction to composite materials, composite NDI theory and best practices, special cases and lessons learned from in-service aircraft, and a guide on constructive use of the NDI proficiency specimens. In addition to the classroom content, the developers of the course designed and fabricated the set of solid laminate composite NDI proficiency specimens that contain realistic, challenging composite structures and representative damage. The specimen set is used to reinforce teaching points of the course and test/enhance inspector proficiency. Extensive details of the course modules, hand-on exercises, and the proficiency specimens are all presented in this report.

1. BACKGROUND AND INTRODUCTION

The FAA Composite Inspector Training Course is composed of a classroom lecture portion and a set of hands-on student exercises meant to teach and reinforce best practices for inspecting composite laminate structures. The information provided in each chapter in this report is intended to provide background to aid the actual classroom lectures and student exercises associated with the FAA Composite Inspector Training Course. The Airworthiness Assurance NDI Validation Center (AANC), with input from industry and the Inspection Task Group of the Commercial Aircraft Composite Repair Committee chaired by the AANC, developed training materials for the Composite Inspector Training Course. The first deployment of this course was conducted in collaboration with Delta® Air Lines in Atlanta, GA in July 2016 [1, 2].

Modules—including an introduction to composite materials, composite nondestructive inspection (NDI) theory and practice, special cases, and lessons learned—were produced in addition to various hands-on NDI exercises. A set of proficiency specimens containing realistic composite structures and representative damage was designed to reinforce teaching points of the course and "test" inspectors' proficiency. Appendix A provides a course guide titled "Composite Inspector Training Course Material Guide" that provides an overview of all of the materials developed as part of this effort. Extensive details of the course modules, hand-on exercises, and the proficiency specimens are all presented in this report.

Composites have many advantages for use as aircraft structural materials, including their high specific strength and stiffness, resistance to damage by fatigue loading, light weight, and resistance to corrosion. The aircraft industry continues to increase its use of composite materials, especially in principle structural elements. This expanded use, coupled with difficulties associated with damage tolerance analysis of composites, has placed greater emphasis on the application of accurate NDI methods. Carbon fiber-reinforced composite materials were once used mainly for secondary components, but they are now used to fabricate major structural elements in aircraft because of their extreme damage tolerance and high strength-to-weight ratio [3]. This is witnessed in projects such as the Boeing® 787, the Airbus® 380, and others in which carbon graphite composites are used for the wings, the empennage, the fuselage, bulkheads, and other primary structural elements. This increased usage in critical structural components has created a greater need for quality and NDI procedures [4, 5]. Accordingly, increased use of solid laminate composites is driving changes to airline NDI training requirements.

2. MODULE 1: INTRODUCTION TO THE COMPOSITE INSPECTOR TRAINING CLASS – MOTIVATION, BACKGROUND, AND COURSE CONTENT

Module 1 in the Composite Inspector Training Course lecture series addresses the reasons for offering a focused training class and discusses the benefits that students will receive from taking this class. Figures 1–48 encompass all of the lecture materials for "Module 1: Introduction, Motivation, Objectives, and Expected Outcome from the Class." If inspectors are provided with the motivation behind this class and shown how it can help them improve their performance when inspecting composite materials, they may be more receptive to the material and exercises presented in this class.

2.1 AIRLINE PERSPECTIVES ON TRAINING NEEDS FOR NDI OF COMPOSITES

Inspection procedures for solid laminate composite structures differ markedly from traditional metallic structures. Airlines and the aerospace sector face a challenge in how to develop and deliver appropriate training to personnel performing NDI on modern aircraft with an increased percentage of solid laminate composite construction. The Composite Inspector Training Course is intended to aid airlines in implementing a training program that allows inspection personnel to develop the skills to safely, effectively, and reliably inspect solid laminate composites. NDI classroom training is defined as an organized and documented program of activities designed to impart the knowledge and skills necessary to obtain the level of qualification sought, which consists of classroom and laboratory training. An individual maintains qualification through continued use of the skills, coupled with recurrent training.

Airlines maintain procedures that establish the minimum requirements for the training, examination, and qualification of personnel involved in the application of NDI methods to aircraft, engines, and components. The purpose of this class is to ensure that individuals performing NDI on composites are fully informed about procedures, techniques, and equipment, and are competent to perform the applicable duties. It is understood that the effectiveness of NDI relies on the capabilities, knowledge, and experience of the individuals responsible for proper application of the method and correct interpretation of test results. This procedure is established based on information contained in ATA Spec 105, NAS 410, and ASNT Recommended Practices No. SNT-TC-1A [6–8]. These documents apply to personnel using NDI methods to accept materials, repairs, products, systems, or components as directed by the applicable maintenance and inspection program. They also apply to those individuals who determine the technical adequacy of NDI techniques and to those providing technical training to NDI personnel.

At the forefront of NDI techniques is pulse-echo ultrasonics, which is the dominant inspection method in Boeing 787 part production [3, 9]. Within ultrasonic use, phased arrays are gaining popularity because of their increased inspection speeds for large-area NDI. Carbon fiber-reinforced polymer aircraft require in-service inspections, although to a lesser extent than more traditional aircraft designs. These in-service inspections cannot rely on robotic automated systems to achieve high scanning speeds. Data acquisition and interpretation are also sometimes made more difficult by complicated internal support structures that are often not well-defined to inspectors. Although inspection requirements vary somewhat, emphasis is still placed on rapid, efficient, and reliable inspections.

2.2 FAA ROLE AND BACKGROUND FOR CLASS

The mission of the FAA is to provide safe and efficient aviation systems. There are many ways in which the FAA seeks to accomplish its mission. The FAA develops safety regulations that set the minimum safety requirements for aviation and also conducts research and development to help it achieve its mission. The William J. Hughes Technical Center in Atlantic City, NJ is one of the nation’s premier aviation research, development, test, and evaluation facilities. The Technical Center serves as the FAA’s national scientific test base for research and development, test and evaluation, verification and validation in air traffic control, communications, navigation, airports, aircraft safety, and security.

The Maintenance and Inspection (M&I) Research Program is a research programs in the Structures and Propulsions Branch. The M&I program conducts research on topics related to maintaining the continued airworthiness of aircraft throughout their service lives. There are many aspects in maintaining aircraft with the overriding goal of ensuring that they remain fully compliant with regulations through the entirety of their commercial operation. As aircraft evolve, the FAA must remain proactive in its safety mission. As such, the FAA is constantly working with the industry, academia, and other government agencies to ensure the safe operation of today's aircraft. The main drivers of M&I research are evolving materials, technologies, and procedures. For example, aircraft have been predominately made from metal alloys such as aluminum and titanium. Today there are many aircraft composed of more than 50% composite materials by weight. The aerospace industry uses composites to take advantage of their high strength-to-weight ratios, superior fatigue lives, resistance to corrosion, and their tolerance to various types of damage. Much of the M&I research portfolio is based on composites' safety and includes evaluating both conventional and advanced nondestructive testing methods' effectiveness in detecting damage in composites, documenting best practices in industry repair methods, and providing training and education to the aviation maintenance workforce to familiarize them with new materials and technologies.

In the mid-2000s, the FAA teamed up with Sandia Labs AANC and more than 35 airlines and original equipment manufacturers (OEMs) from around the world to conduct two probability of flaw detection (POD) experiments: "Composite Honeycomb Flaw Detection Experiment" and the "Composite Laminate Flaw Detection Experiment." The motivation for these experiments was increasing the use of composites on commercial aircraft, and the program goals were to assess and improve damage detection performance in composite aircraft structures. Both programs quantified the performance of current inspection methods for honeycomb and solid laminate composite structures and then determined possible improvements through the use of more sophisticated inspection methods, optimized procedures, and enhanced inspector training. This research provided clear recommendations and best practices for inspecting composites. One recommendation was that the aviation industry should provide more composite-inspection training.

As a follow-up to the composite POD experiments, the M&I program decided to develop training material and proficiency specimens for the industry. Working with the industry, the FAA developed a curriculum that teaches airline inspectors the fundamentals of composite materials and gives them an opportunity to routinely inspect composites. The proficiency specimens used are a series of panels with simulated types of damage, which offer inspectors hands-on experience working with composites. This FAA final technical report presents the finalized curriculum and proficiency specimen exercises.

2.3 AANC EVALUATION OF INSPECTION PRACTICES FOR COMPOSITE STRUCTURES

The FAA Airworthiness Assurance Center, operated by Sandia Labs for the FAA, completed studies that produced quantitative assessments of conventional and advanced NDI techniques for detecting flaws in composite aircraft structures [10–12]. In addition to these quantitative assessments, several studies have been conducted to evaluate the detection and characterization capabilities of various nondestructive testing techniques applied to solid laminate aircraft structures [10, 13–14]. To assess the aviation industry's ability to detect composite damage, the AANC completed several structured POD experiments to evaluate the performance of NDI on composite honeycomb and solid laminate structures, relating damage threat to flaw detection stemming from impact, detection and quantification of weak bonds, assessment of composite porosity levels, and inspection of composite repairs.

These studies generated POD values for inspecting composite honeycomb laminate aircraft structures. The POD experiments were performed by airlines, third-party maintenance depots, and aircraft manufacturers to produce statistically valid POD curves representative of the industry. The collective reliability data allow the NDI community to determine how well current inspection techniques can reliably find flaws in composite structure and to determine the degree of improvements possible through the integration of more advanced NDI techniques and procedures. These studies also produced a series of recommendations for improving the performance of current inspection practices. A primary recommendation was to enhance inspectors' preparation and training by focusing on the unique challenges associated with composite laminate inspections.

Figures 29–46 (POD Studies to Quantify Flaw Detection in Composite Laminate Structures) **describe** the Sandia Labs Composite Laminate Flaw Detection POD Experiment. The intent of the composite POD experiments was to present the inspectors with uniform test panels to: 1) quantify their composite flaw detection performance using current inspection techniques (i.e., assess aircraft maintenance depots), and 2) to determine if better flaw detection can be produced using advanced NDI techniques. The study incorporated statistically relevant and realistic flaw profiles into the test specimens. A series of carbon composite specimens with statistically relevant flaw profiles (e.g., voids, disbonds, delaminations, and impact damage) were inspected using conventional, hand-held pulse-echo ultrasonic testing and resonance, and new NDI methods that have recently been introduced to improve sensitivity and repeatability of inspections. The primary factors affecting flaw detection in laminates were included in this study: material type, flaw profiles, presence of complex geometries like taper and substructure elements, presence of fasteners, secondarily bonded joints, and environmental conditions.

To acquire flaw detection data, POD test specimens representing actual composite structures found on today's aircraft and containing realistic types of damage were shipped to airlines, third-party maintenance depots, aircraft manufacturers, and NDI developer labs around the world. For the solid laminate POD experiment, more than 70 inspectors of different ages and experience levels performed the blind tests to produce statistically valid POD curves representative of the industry as a whole. Inspections were performed in typical maintenance hangars to provide representative environments with common impediments, such as poor lighting, noise, and distractions. All inspectors were provided with the appropriate inspection procedures from The Boeing Company and Airbus SAS maintenance manuals. In addition, each blind inspection process was preceded by proper equipment setup according to appropriate reference standards. Sample POD curves, which relate the probability of finding a flaw to the size of the flaw for a particular specimen construction, were charted for several different types of inspection devices. Results from these POD studies were used to identify paths to inspection improvement. They also provide excellent examples of how training can improve inspection performance. Several key results are included in this lecture module so inspectors can better understand the value of focused training.

Figure 36a shows one sample result for overall flaw detection in composite structures in the 12–20-ply thickness range (Thin Laminate Experiment) [11]. This figure shows the spread of all the individual inspector POD curves (dashed lines) compared with the cumulative $POD_{[90]}$ curve (solid line) for all 27 inspectors. These results were produced by considering all flaws in constant thickness and complex geometry regions of the composite test specimens. The spread shows 15 inspectors with a POD value of less than the cumulative $POD_{[90]}$ = 1.20” diameter flaw ($POD_{[90/95]}$ = 1.29”) and 12 inspectors with a POD value higher than the cumulative $POD_{[90]}$ value. The variation within the experiment ranged from a POD = 0.53” diameter flaw for the best-performing inspector to a POD = 2.17” diameter flaw for the worst-performing inspector. The standard deviation for the inspector POD data set was a 0.417” diameter flaw. The overall POD performance calculated when a 90% flaw detection was combined with a 95% confidence bound was $POD_{[90/95]}$ = 1.29” diameter flaw. For these experiments, POD values were calculated using a pass/fail analysis with a log normal model.

Figure 36b shows another sample result for overall flaw detection in composite structures in the 12–20-ply thickness range (Thick Laminate Experiment) [11]. This figure shows the spread of all the individual inspector POD curves (dashed lines) compared with the cumulative $POD_{[90]}$ curve (solid line) for all 30 inspectors who participated in the 20–32-Ply Thick Laminate Experiment. These results were produced by considering all flaws in constant thickness and complex geometry regions. The spread shows 19 inspectors with a POD value of less than the overall cumulative $POD_{[90]}$ = 0.77” diameter flaw ($POD_{[90/95]}$ = 0.82”) and 11 inspectors with a $POD_{[90]}$ value higher than the overall cumulative $POD_{[90]}$ value. The variation within the experiment ranged from a POD = 0.20” diameter flaw for the best-performing inspector to a POD = 1.70” diameter flaw for the worst-performing inspector. The standard deviation for the inspector POD data set was a 0.420” diameter flaw. The overall POD performance calculated when a 90% flaw detection was combined with a 95% confidence bound was $POD_{[90/95]}$ = 0.82” diameter flaw. It can be seen that the overall cumulative $POD_{[90/95]}$ for all flaws in the Thick Laminate Experiment (20–32-ply skins, plus substructure elements) was better (i.e., lower). This was mainly because of the construction method used for this set of test panels, which involved a co-cured substructure bond line that was less attenuative, and included less “noise” in the signals than the secondarily bonded substructure (film adhesive bonding) used in most of the Thin Laminate Experiment test specimens. Also, the test specimens for the Thick Laminate Experiment did not contain curvature, fasteners, sealed joints, or skin-over-honeycomb substructure. This eliminated some of the deployment, human factors, and signal interpretation challenges that were present in the Thin Laminate Experiment. Finally, it should be noted that the 20–32-ply specimen set included 12 ft^2 of inspection area, whereas the 12–20-ply specimen set included 34 ft^2 of inspection area. Therefore, inspector fatigue was less of an issue in the Thick Laminate Experiment.

The major item of concern in the NDI performance plots of figures 36a–36b is the amount of spread in the results across the array of inspectors. This indicates that an inspector carrying out a composite inspection job card could produce a good, average, or poor inspection result. The purpose of the Composite Inspector Training Class is to reduce the degree of variation from one inspector to another. By reinforcing key inspection practices, focusing on the unique challenges associated with composite inspections and providing more hands-on exposure to realistic composite inspections, this class is set up to eliminate the poor performance outliers and increase the level of reliability across the entire population of aircraft inspectors. Several more examples are included in this lecture package to reinforce this approach.

During the POD testing, the experiment monitors also recorded various methods that inspectors used to ensure inspection area coverage for the Composite Laminate POD Experiment. Some inspectors covered the inspection area with their UT transducers using a pure freehand approach (i.e., no guides or markings on the panels). Some inspectors divided the inspection surface into quadrants to reduce freehand coverage errors. Some inspectors used a series of tick marks, often placed at 0.5” or 1” intervals, to divide the inspection surface into a number of rows and columns. Some inspectors used flexible straight edges to guide their transducer movement. The different surface coverage techniques observed fall into four categories. The POD results produced by each of these inspection coverage methods were calculated separately and compared to quantify the benefits of deploying specific inspection coverage methods. The four different surface coverage techniques were combined and analyzed for the overall Solid Laminate POD Experiment. The POD results produced by each of these inspection coverage methods for the 12–20-ply and 20–32-ply specimen sets were combined and calculated separately. These are plotted in figure 38 along with the corresponding $POD_{[90/95]}$ values.

The best performing coverage method that produced the lowest (i.e., best) combined POD level was when inspectors made tick marks for spacing and used a straight edge on all panels throughout both experiments, achieving a $POD_{[90/95]}$ = 1.0” diameter flaw. This produced a 28% improvement compared with the $POD_{[90/95]}$ value of 1.29” diameter flaw. The second best performing coverage method was when inspectors used a straight edge on all panels throughout both experiments. This produced a $POD_{[90/95]}$ = 1.1” which is a 17% improvement compared with the overall, cumulative combined $POD_{[90/95]}$ value of 1.29” diameter flaw. The third best performing coverage method was when inspectors started the experiments using a straight edge but at some point during the experiments switched to freehand (13 inspectors). This method produced a $POD_{[90/95]}$ = 1.42” which is a decrease in performance of 10%. The poorest-performing coverage method was when inspectors used the freehand method on all panels throughout both experiments. This produced a $POD_{[90/95]}$ = 2.39” which is an 85% decrease in performance compared with the overall cumulative $POD_{[90/95]}$ value of a 1.29” diameter flaw. The data show inspectors need to use some sort of guide to ensure that they are achieving full coverage with their inspection.

Finally, figure 39 provides one last example depicting improvements that could be achieved with focused training. In this example, performance brackets were used to place inspectors into groups and then calculate the resulting $POD_{[90/95]}$ for each performance bracket. These performance brackets used the inspectors that fell into the 40, 60, and 80 percentile categories. The inspectors that fell into the 40 percentile group (12 inspectors, each having a $POD_{[90]}$ of less than 0.55”) produced a 42% improvement to $POD_{[90/95]}$ = 0.48” diameter flaw value compared with the overall cumulative $POD_{[90/95]}$ = 0.82” diameter flaw. The 60 percentile group (18 inspectors, each having a $POD_{[90]}$ of less than 0.75”) produced a 34% improvement with a $POD_{[90/95]}$ = 0.54” diameter flaw. The 80 percentile group (24 inspectors, each having a $POD_{[90]}$ less than 1.00”) showed a 20% improvement with a $POD_{[90/95]}$ = 0.66” diameter flaw. These performance brackets might be useful to airlines and maintenance and repair organizations, which can judge where their inspectors fall within the brackets and the resulting performance they will obtain from their inspectors. The results in figure 39 also reveal the degree of inspection improvements possible if inspectors can shift their performance from the higher (worse) performance brackets to the lower (better) performance brackets. This shift in performance can be brought about by improved or more extensive composite inspection training.

In general, these flaw detection performance experiments have established the current baseline for the aviation industry in terms of quantifying performance of NDI techniques for composite structures. They also are paving the way for improvements in industry inspections via optimized procedures and practices, including proper training for the inspectors. The superior capabilities of a host of advanced NDI techniques have been demonstrated, and these methods are ready to be applied to the next generation of complex composite structures on commercial aircraft about to enter service. This is a key point—adequate inspection capability is needed to instill confidence in the industry's ability to safely fly composites-intensive aircraft. Airlines and other aircraft maintenance facilities can implement good inspections with the right equipment using consistent standards, procedures and training.

3. RECOMMENDATIONS REGARDING ENHANCED TRAINING

The NDI performance assessment POD experiments described above produced a number of recommendations for improving inspection of composite structures [9, 11–12]. This included the use of guides to ensure proper surface area coverage, proper equipment calibration, recognition of critical aspects of the UT signals, and the proper deployment of transducers. Many of these issues can be mitigated by additional personnel training. Some of the training can be in the form of composite awareness training to instruct inspectors on composite materials, composite structure fabrication, and typical aircraft composite construction designs. Other forms of training can stress procedural aspects of the inspections such as the use of NDI deployment aids and the proper use of drawings to assist in signal interpretation.

In addition, Sandia Labs AANC conducted a survey of the aircraft maintenance industry to assess its needs and desires regarding training. The following are some recommendations for aviation industry teams and airline training departments to improve inspections [11]:

- Overall, the identified, potential measures to improve inspector's performance on composite inspections include increased training, apprenticeships, exposure to representative inspections, enhanced procedures, inspector teaming, and awareness training on inspection obstacles.

- A majority (86%) of the industry does not have additional special inspector qualification/certification to qualify personnel for conducting composite inspections. Most companies use the normal qualification program for general NDI inspection as qualification for composite inspection. Specialized certification for aircraft NDI professionals who inspect composite structures should be considered.

- Survey respondents requested additional guidance related to composite NDI training from the OEM, the FAA, and industry groups in the areas of specific instrument training, specific methods training, repair inspections, composite construction training, and reference standard fabrication and use. They requested that programs supporting the evolution of such training should be initiated and pursued in an industrywide approach.

- There is a general concern that the lack of routine exposure to composite inspections makes it difficult for the inspectors to maintain the necessary level of expertise. Furthermore, exposure to available flaw specimens is viewed as a way to keep inspectors ready for when an aircraft needs inspection because of damage. So, in addition to formal composite NDI training classes, aircraft inspectors should conduct routine practice inspections on representative composite structures that contain realistic damage. Such test specimens should be more complex and varied than the existing NDI Reference Standards and contain known, but nonuniformly spaced flaw profiles.

As a result, this effort was formed to produce more specialized training, above and beyond Level I, II, and III certification. The training was developed to specifically address composite inspections. It would help to have a class that focuses on the unique challenges and signal differences associated with composite inspections. For example, signal characteristics related to ply tapers, secondary bonds, and composite repairs could be discussed so that it is easier for inspectors to distinguish flaw signals from those generated by pristine structure.

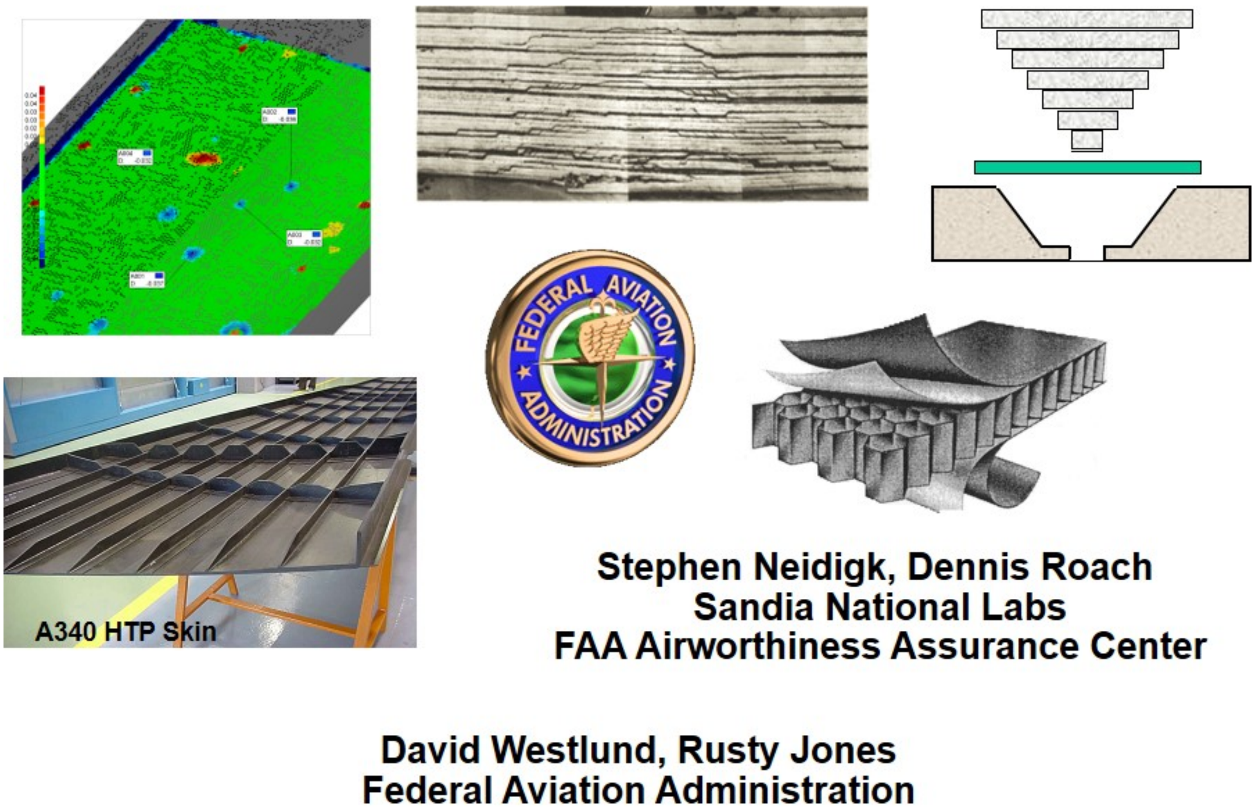


Figure 1. Composite NDI training class

Class Definition - Training Deployment

- **Output** – AIR with modified NDI training AC that references this AIR
- **Instructor** – recommendations on capabilities, background and experience to teach subject matter (approved by airline)
- **Target Class Length** – 3 days (1 1/2 day classroom, 1 1/2 day hands-on)
- **Format** – stand-alone course but assumption is min of Level I student
- **Apprentice Program** - create in-house competency to facilitate senior inspectors working with junior inspectors
- **Attendees** - Consider possible attendance by engineers or other maintenance personnel (aid for NDI that they specify)
- **Assessment of NDI Students** – qualification; certification or "endorsement" by airline
- **Use of Various NDI Devices** – consider possible courses run by NDI education companies (recommend use of std/common NDI equipment)

Figure 2. Overview of composite laminate NDI training class

Class Modules

1. Introduction, Motivation, Objectives & Expected Outcome from Class
2. Composite Awareness – Materials, Design, Fabrication and Use
3. Composite NDI – Theory and Practice
4. Special Cases - Challenges & Lessons Learned
5. NDI Proficiency Specimens
6. Composite NDI – Hands-On Exercises

Figure 3. Modules in composite laminate NDI training class

Class Definition – General Training Content

- Summary of typical structural configurations from NDI perspective - clearer schematics showing structural configuration
- Present NDI challenges and means to address them – geometry (e.g. taper, substructure, curvature), signal interpretation (many samples highlighting complex signal reflections, confounding presence of signal harmonics, rapid variations produced by changing/complex geometry), attenuative materials
- Field issues – repair and/or NDI common errors; human factors concerns
- Inspection cases – typical and unique (unexpected) demands; review of inspection processes and issues/problems from the field (examples from OEMs, airlines; "gaps" in NDI specifications)

Class Definition – General Training Content (cont.)

- NDI - theory & physics, methods & usage by application (alternatives; where & how to apply); deployment; reiterate proper use of procedures (use of laminate NDI procedure); training on navigation of existing documents
  - ➢ Single-element PE-UT
  - ➢ Phased array PE-UT (water box, rolling arrays, flexible) – unique calibration/element normalization
  - ➢ A-Scan, B-Scan, C-Scan & S-Scan
  - ➢ NDI deployment & process controls (QA)
  - ➢ Other methods – thermography, resonance, TTU (special case with access to both sides via yoke), FTIR, DAV
  - ➢ Resources – industry docs like ACs, AIRs, NDI Handbook, DOT reports, NDI support training, OEM support/guidance, in-house (airline) guidance

Class Definition – General Training Content (cont.)

- Use of NDI Proficiency Specimens - usage processes/modes for feedback & learning; optional use of actual aircraft structure to compliment NDI Proficiency Specimens (modeled after representative structures)
- Hands-on portion of class – designed exercises, equipment, selection of transducer, proper use of NDI Ref Stds; highlight lessons learned with lab exercises; use of TCG curves

Figure 4. Composite laminate NDI training class—Content

Class Definition - Use of Existing Guidance

- Related industry classes
- **Section 51**, Nondestructive Test Manuals
- **AC 25-29** – Development of a Nondestructive Inspection Program
- **AC-43-13-1B** - Acceptable Methods, Techniques, and Practices for Aircraft Inspection and Repair
- **AC 43-214** – Repairs and Alterations to Composite and Bonded Aircraft Structure
- **AC 65-31B** – Training, Qualification and Certification of Nondestructive Inspection Personnel
- **AC 65-33** – Development of Training/Qualification Programs for Composite Maintenance Technicians
- **AIR 4938** - Composite and Bonded Structure Technician/Specialist: Training Document
- **AIR 5279** - Composite and Bonded Structure Inspector: Training Document
- **AE-27** - Design of Durable, Repairable, and Maintainable Aircraft Composites
- **DOT/FAA/AR-08/54** – Guidelines for the Development of a Critical Composite Maintenance and Repair Issues Awareness Course

Class Definition - Use of Existing Guidance

Certification & Qualification of Inspectors

- **ATA-105** - Airlines for America (A4A) Specification 105, Training and Qualifying Personnel in Nondestructive Testing Methods
- **NAS-410** - Aerospace Industries Association, National Aerospace Standard, NAS Certification & Qualification of Nondestructive Test Personnel.
- **EN 4179** - Aerospace Series, Qualification and Approval of Personnel for Non-destructive Testing
- **SNT-TC-1A** – ASNT Recommended Practice, Personnel Qualifications and Certification in Nondestructive Testing
- **FAA Order 8900.1 Volume 6**, Chapter 11, Section 27 – Surveillance of a Nondestructive Test Program/Facility (Flight Standards policy and guidance concerning aviation safety inspector job tasks)
- **MIL-STD-410E** - Military Standard Nondestructive Testing Personnel Qualification and Certification
- **CAN/CGSB 48.9712.2006** - Canadian General Standards Board (ISO 9712:2005), Non-destructive Testing—Qualification and Certification of Personnel.

Figure 5. Existing guidance related to composite laminate NDI training class

There are many types of composite defects and damage that can arise in the field. Field damage can result from in-flight stress & fatigue loads, operating environment, dropped tools, service vehicles, aircraft-handling accidents, dropped parts, and other types of impacts. This course is intended to educate and refresh NDT inspectors in all aspects of composite laminate inspections while providing a foundational understanding of the fabrication processes for carbon fiber aircraft parts.

Figure 6. Composite NDI training class—Damage detection needs

- To produce a general understanding of composite materials and how carbon fiber parts are manufactured.
- To produce an in-depth understanding of the nondestructive testing methods used to inspect carbon fiber parts.
- To provide an understanding of the fundamental ultrasonic principles and the specific use of UT for composite NDI.
- To allow students to recognize critical aspects of A-Scan, B-Scan and C-Scan signals and to understand the analysis of ultrasonic pulse echo signals to include bond lines and substructures.
- To produce an overall inspection proficiency on composite aircraft parts made up of a variety of structural configurations to include: laminates, laminates with substructure, co-cured bond lines, secondary bond lines, co-bonded parts, tapered laminates, inspection over fasteners, effects of sealant in the inspection zone, and various types of damage to include disbonds, delaminations, porosity and impact damage.

- Course is used to facilitate airline inspector and maintenance personnel training so that they:
    1) See the proper approaches to their tasks (technique optimization, deployment issues and human factors concerns)
    2) Realize the value of their actions in realistic inspection, maintenance and repair scenarios
    3) Understand the relationship between their activities (e.g. inspection performance, repair process, coatings, damage assessment) and the structural integrity of the composite part.
- Note use of complimentary OEM training, guidelines and maintenance documents

Goal of training is to enhance aircraft safety & optimize aircraft utilization by improving NDI flaw detection performance in composite aircraft structure.

Figure 7. Composite NDI training class—learning objectives

- Extensive/increasing use of composites on commercial aircraft
- Solid Laminate Flaw Detection Experiment (Probability of Detection) produced a quantitative assessment of conventional and advanced NDI techniques for detecting flaws in composite laminate aircraft structures
- SLE also produced a series of recommendations for improving the performance of current inspection practices – key recommendation was to enhance an inspector's training
- Survey of NDI practices and training in aviation industry – overwhelming support for additional guidance and training
- Extensive portfolio of FAA & industry composite research initiatives highlighted the importance and challenges associated with inspection of composite structures: Industry-wide Composite NDI Reference Standards, NDI Assessment: Honeycomb Structures, NDI Assessment: Solid Laminate Structures, Composite Repairs and Bonding, Composite Porosity, Composite Impact , Assessment of Heat Damage in Composites
- Identified clear need for specific training, beyond Level I, II, III, that addresses composite inspection methods – focus on the unique challenges associated with composite laminate inspections; allow for additional and routine exposure to composite laminate inspections; reiterate/recommend helpful procedural aids

Figure 8. Composite NDI training class—Drivers

Figure 9. Motivation for composite NDI training class

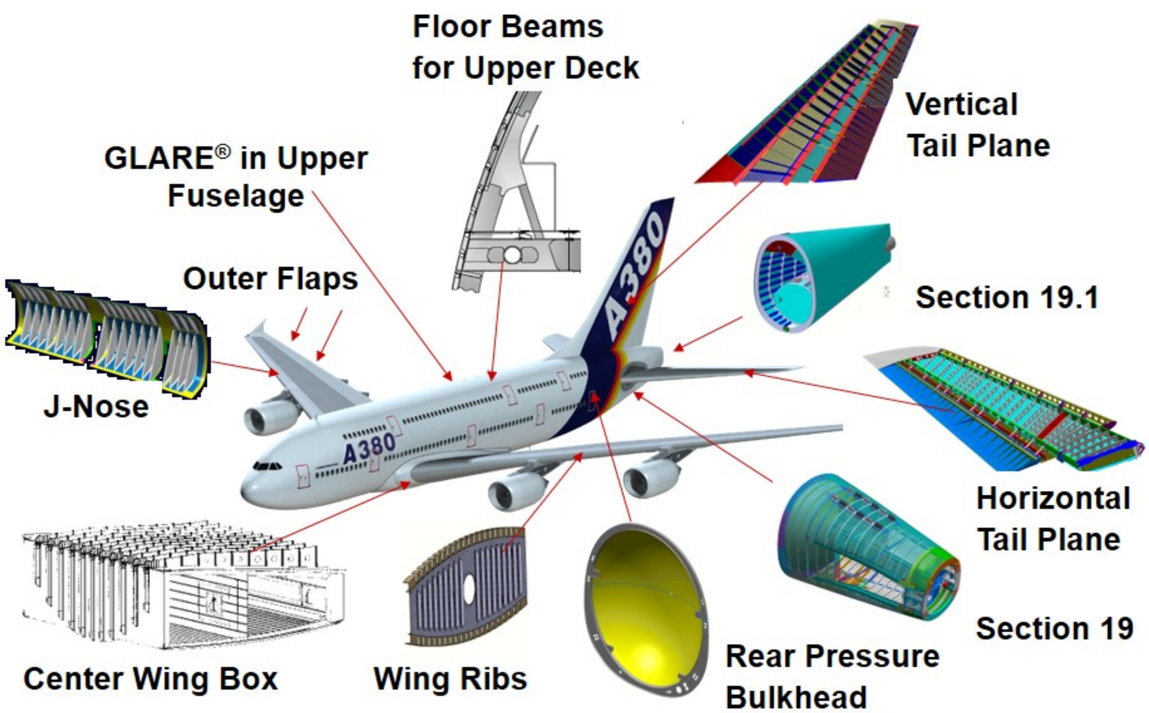


Figure 10. Sample composite structures on Airbus aircraft

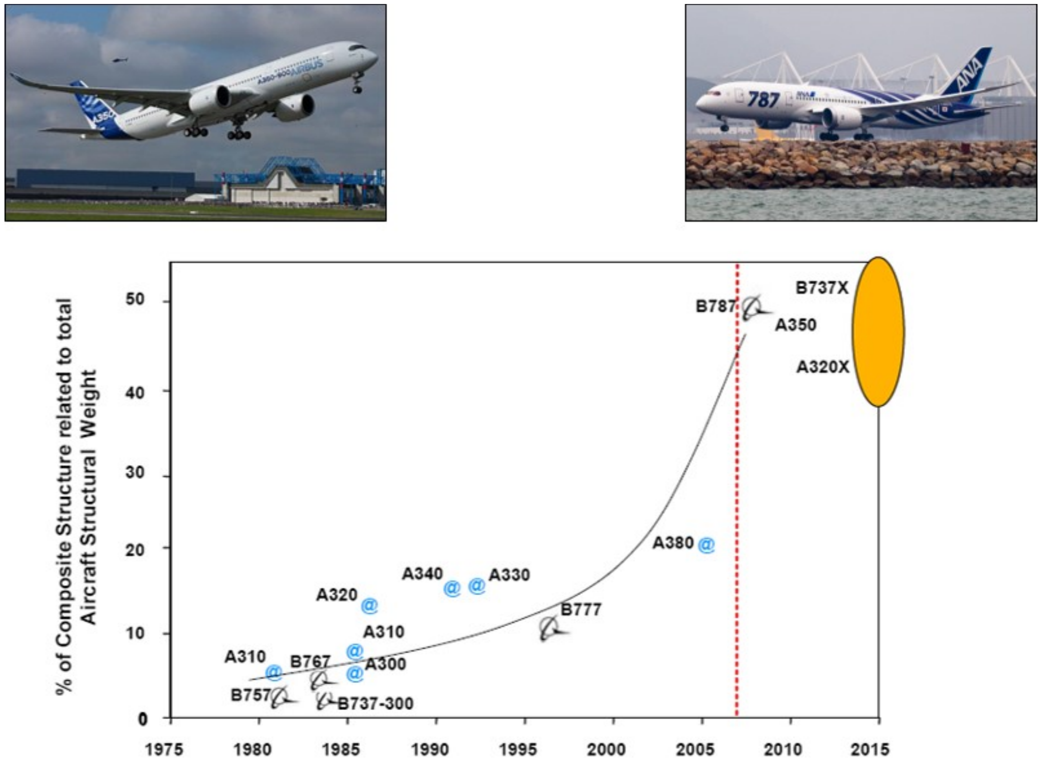


Figure 11. Expansion in use of composite materials in aircraft construction

Figure 12. Sample composite structures on Airbus aircraft

Figure 13. Prepreg—Automated tape laying

- Normal & abnormal flight loads
- Fluid contamination & ingress
- Surface coating removal; erosion
- Impact (in-flight, ground handling equipment)
  - hail, birds, lightning, runway debris, tire separation
- Heat & UV exposure
- Corrosion in metals associated with CFRP
- Maintenance errors

<u>Inspection Challenges</u>

- Delaminations & disbonds
- Hidden damage
- Small amounts of moisture
- Heat damage that affects resin matrix
- Weak bonds (manuf. or environment induced)

Figure 14. Types of damage in composite structure

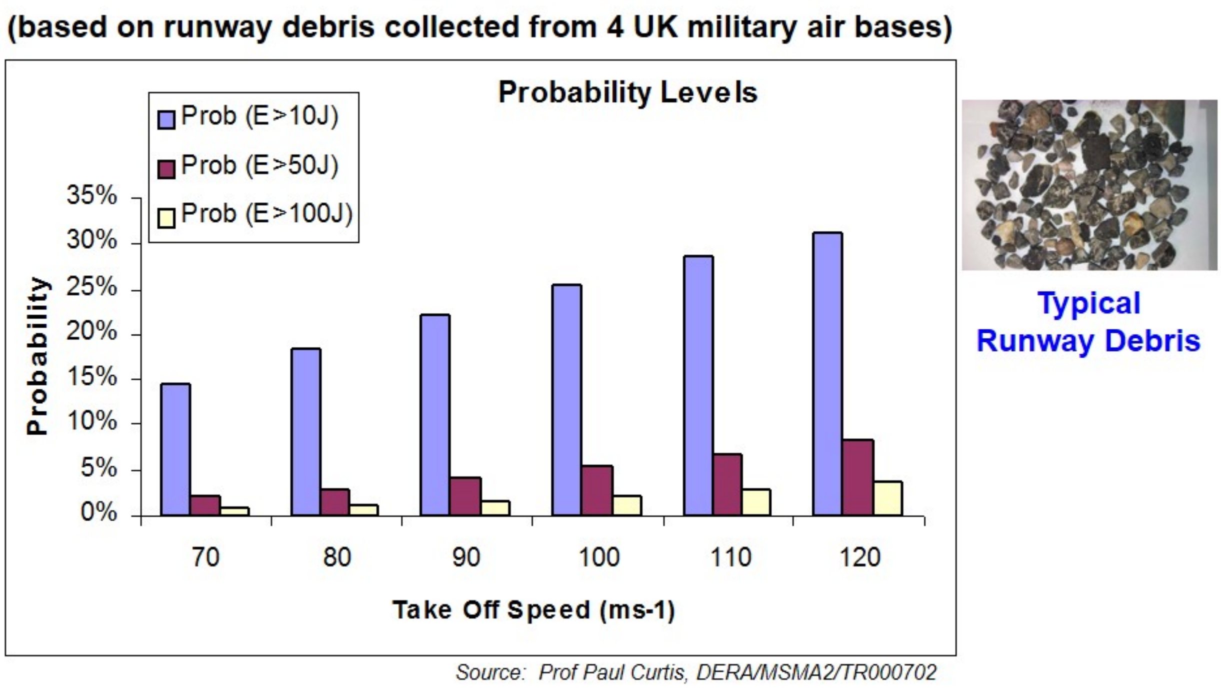


Figure 15. Probability of impact energy as a function of takeoff speed

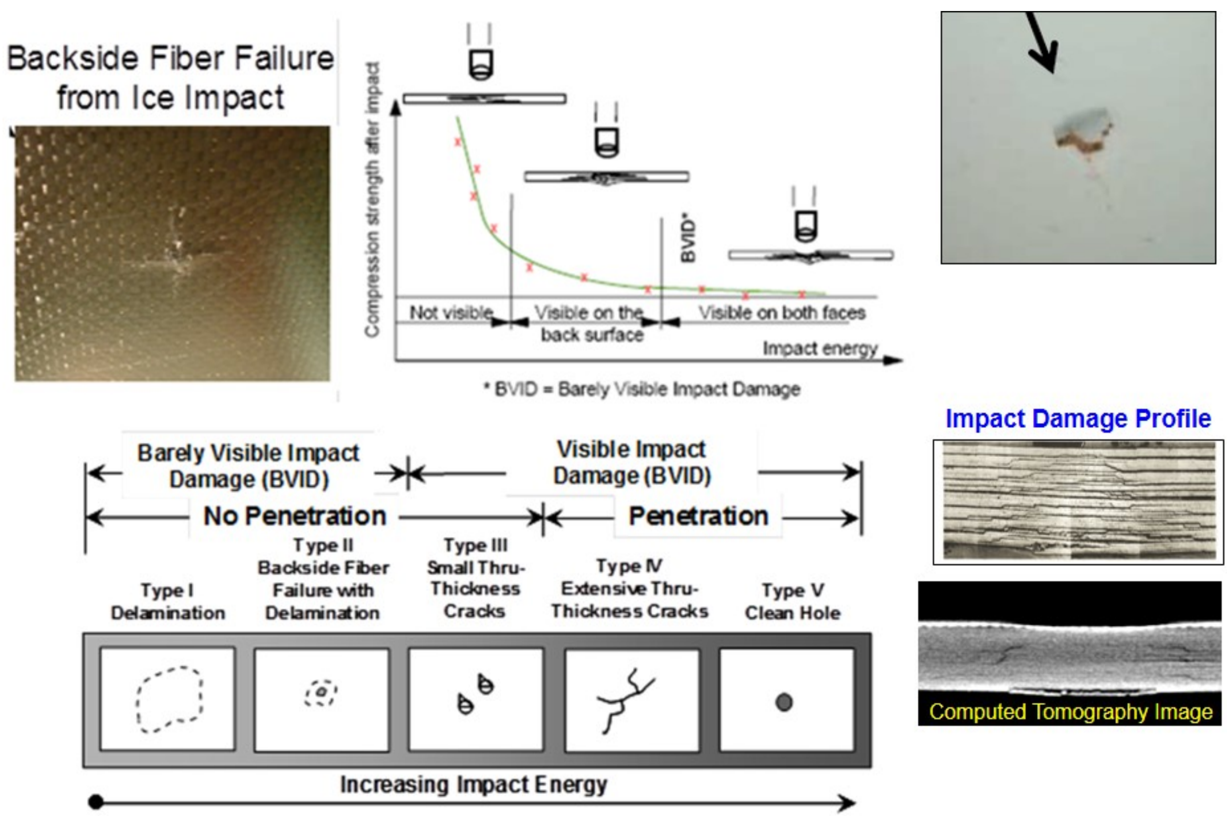


Figure 16. Effects of impact on composite structures

Category	Examples (not inclusive of all damage types)
Category 1: Allowable damage that may go undetected by scheduled or directed field inspection (or allowable mfg defects)	Barely visible impact damage (BVID), scratches, gouges, minor environmental damage, and allowable mfg. defects that retain ultimate load for life
Category 2: Damage detected by scheduled or directed field inspection @ specified intervals (repair scenario)	VID (ranging small to large), deep gouges, mfg. defects/mistakes, major *local* heat or environmental degradation that retain limit load until found
Category 3: Obvious damage detected within a few flights by operations focal (repair scenario)	Damage obvious to operations in a "walk-around" inspection or due to loss of form/fit/function that must retain limit load until found by operations
Category 4: Discrete source damage known by pilot to limit flight maneuvers (repair scenario)	Damage in flight from events that are obvious to pilot (rotor burst, bird-strike, lightning, exploding gear tires, severe in-flight hail)
Category 5: Severe damage created by anomalous ground or flight events (repair scenario)	Damage occurring due to rare service events or to an extent beyond that considered in design, which must be reported by operations for immediate action

Source: Larry Ilcewicz, FAA Senior Technical Specialist - Composites

Figure 17. Categories of damage and defects to consider for primary composite aircraft structures

Figure 18. Sources of damage in composite structures

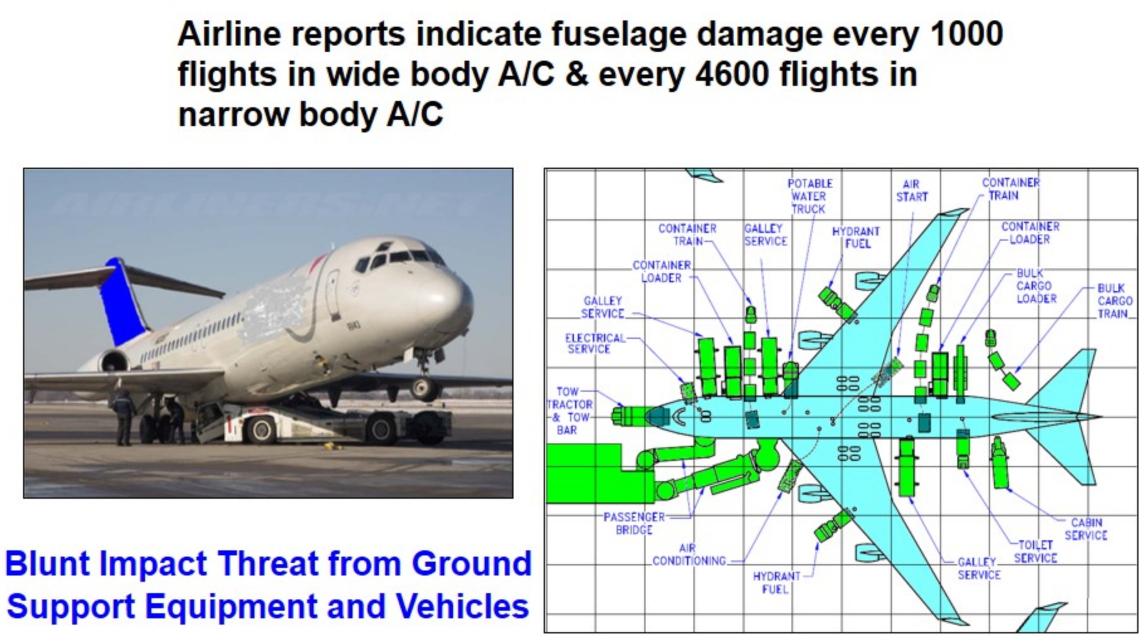


Figure 19. Ground-handling damage in composite structures

Figure 20. Unusual sources of damage in composite structures

Figure 21. Sources of damage in composite structures

Figure 22. Samples of composite laminate damage

Figure 23. Inspection challenge—Hidden impact damage

Figure 24. Composite laminate repairs

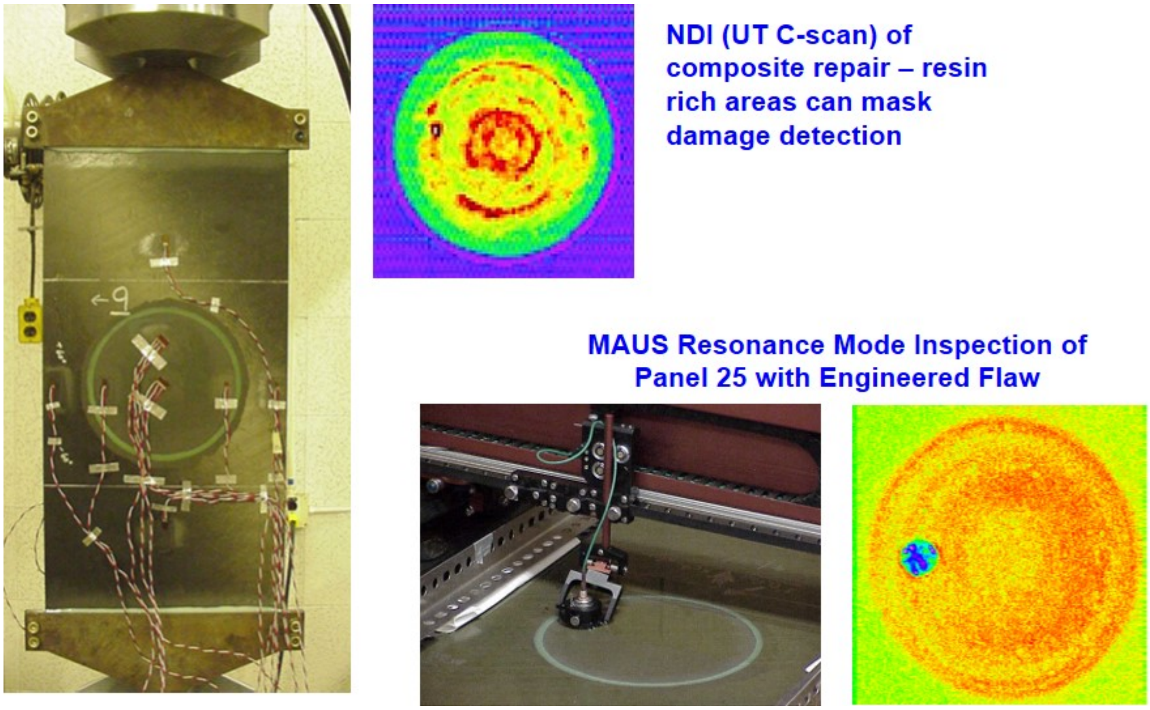


Figure 25. NDI of composite laminate repairs

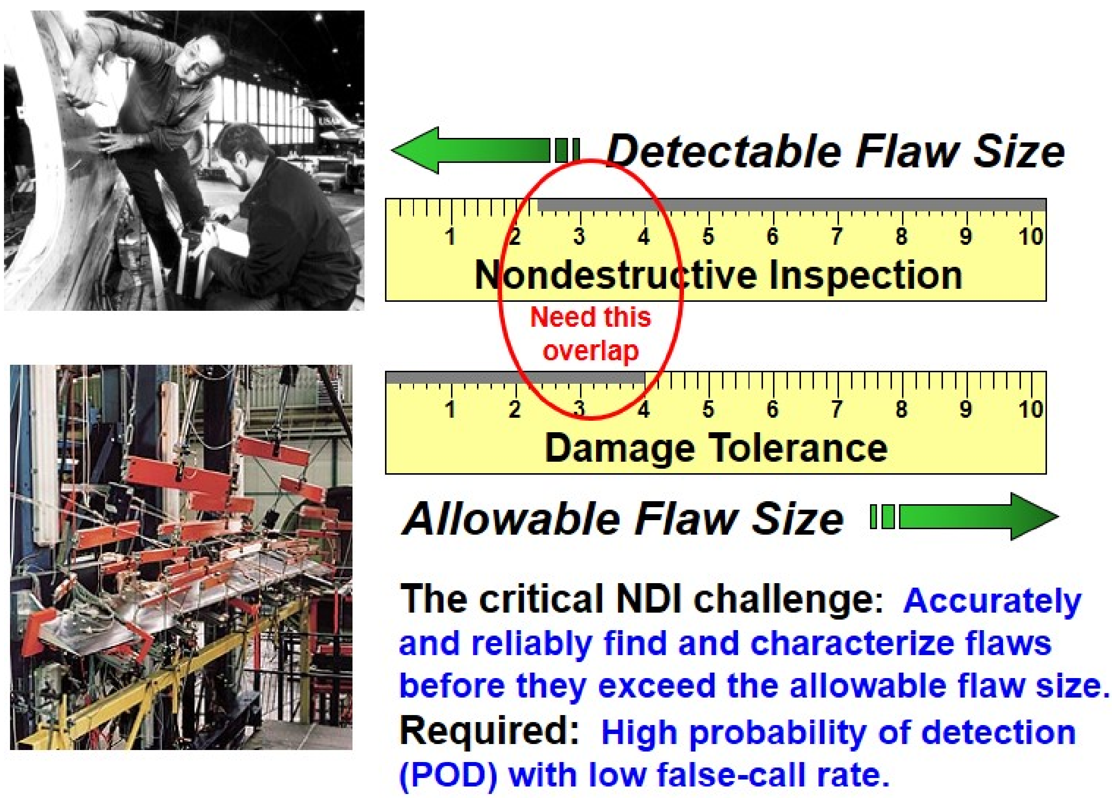


Figure 26. Bottom line for NDI—Required relationship between structural integrity and inspection sensitivity

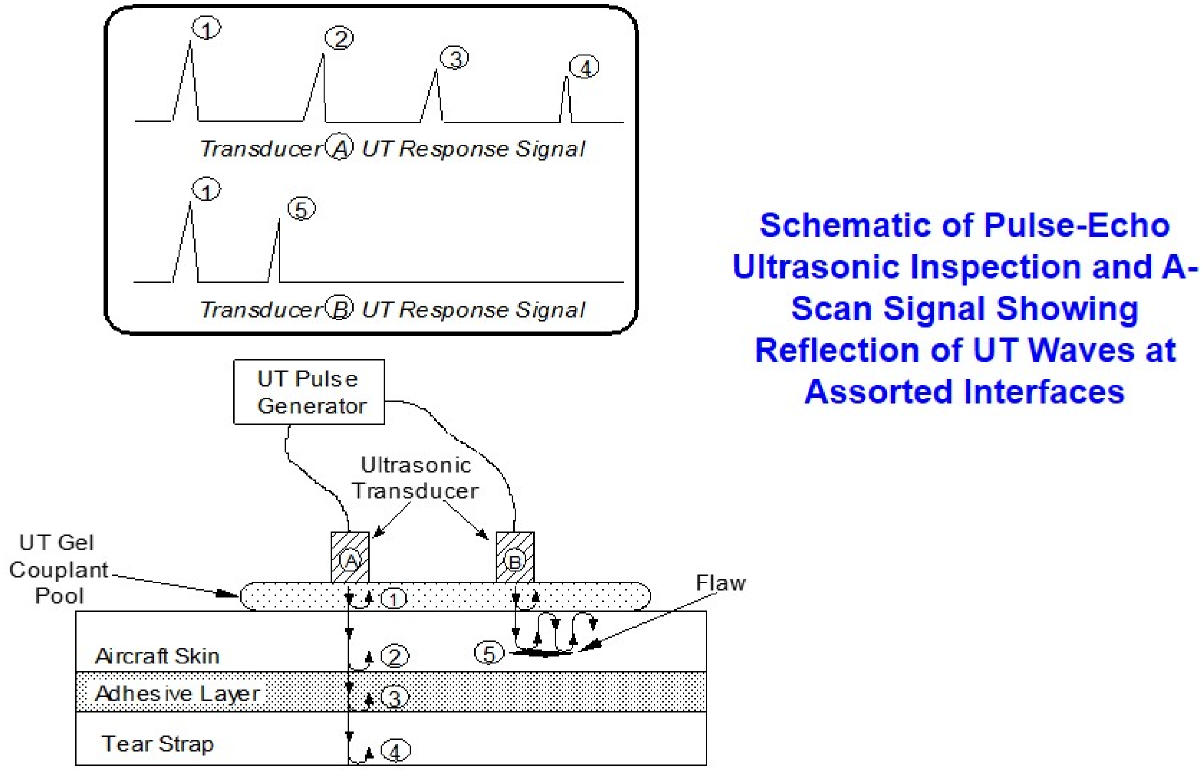


Figure 27. Basic pulse-echo ultrasonics

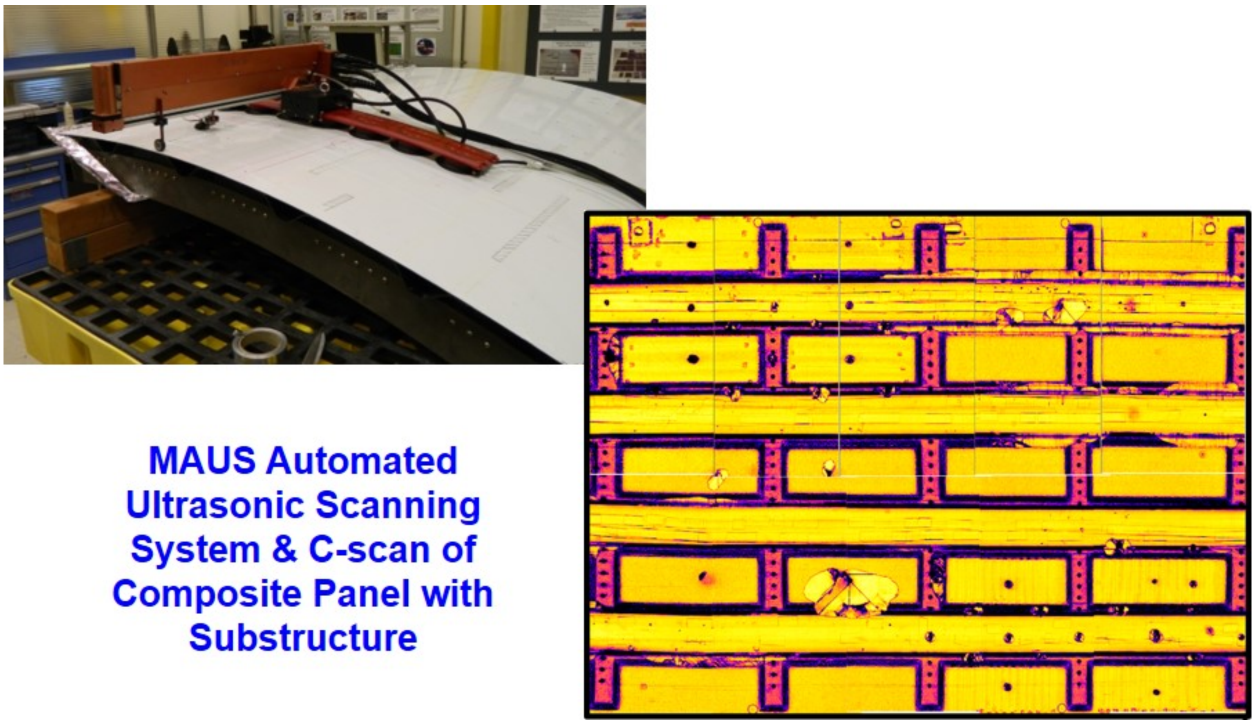


Figure 28. Movement into scanning pulse-echo ultrasonic equipment

Purpose

- Determine in-service flaw detection capabilities: 1) conventional NDT methods vs. 2) improvements through use of advanced NDT.
- Optimize laminate inspection procedures.
- Compare results from hand-held devices with results from scanning systems (focus on A-scan vs. C-scan and human factors issues in large area coverage).
- Provide additional information on laminate inspections for the "Composite Repair NDT/NDI Handbook" (ARP 5089).

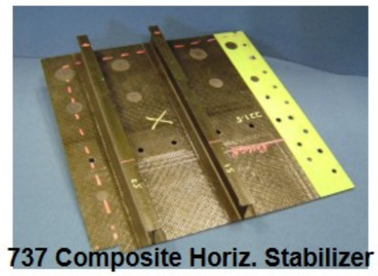


Figure 29. An experiment to assess flaw-detection performance in composite laminate structures

Approach

- Statistical design of flaws and other variables affecting NDI - range of types, sizes & depths of flaws
- Study factors influencing inspections including composite materials, flaw profiles, substructures, complex shapes, fasteners, secondary bonds, and environmental conditions
- POD and signal-to-noise data gathering
- NDI Ref. Stds. prepared to aid experiment

Expected Results - evaluate performance attributes

1) accuracy & sensitivity (hits, misses, false calls, sizing)
2) versatility, portability, complexity, inspection time (human factors)
3) produce guideline documents to improve inspections
4) introduce advanced NDI where warranted

Figure 30. Flaw detection in solid laminate composites

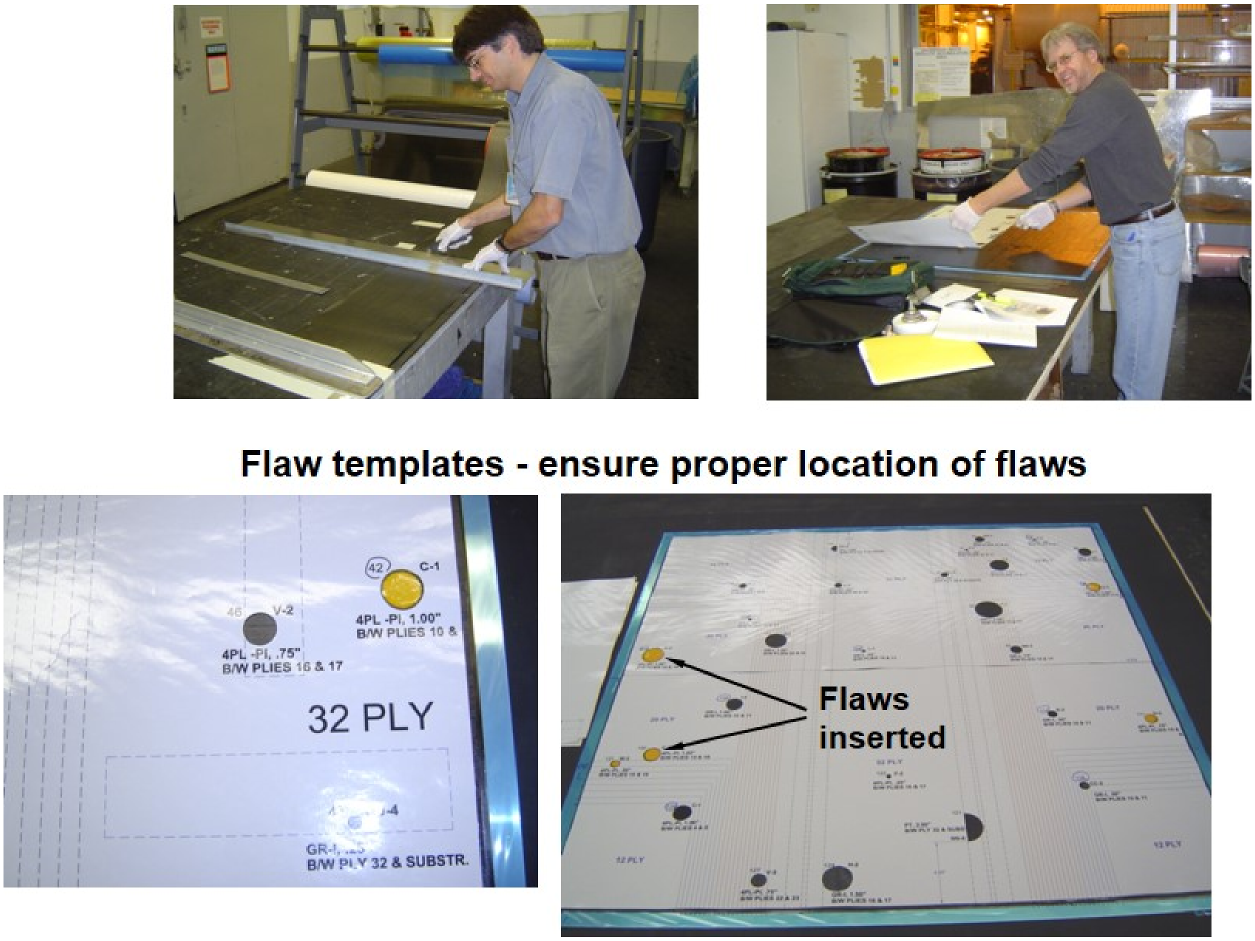


Figure 31. Thick laminate with complex taper—Fabrication

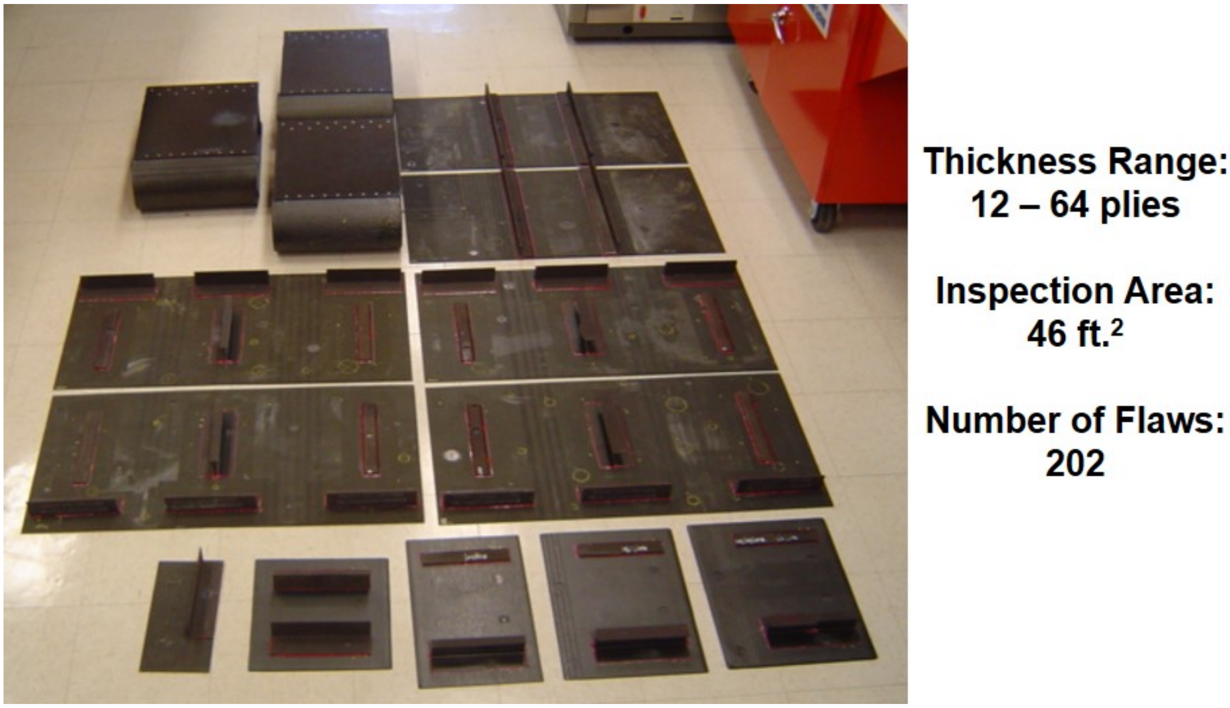


Figure 32. Specimen set—Flaw detection in solid laminate composites (back)

Figure 33. Specimen set—Flaw detection in solid laminate composites (front)

Figure 34. Sample participants in the solid laminate experiment

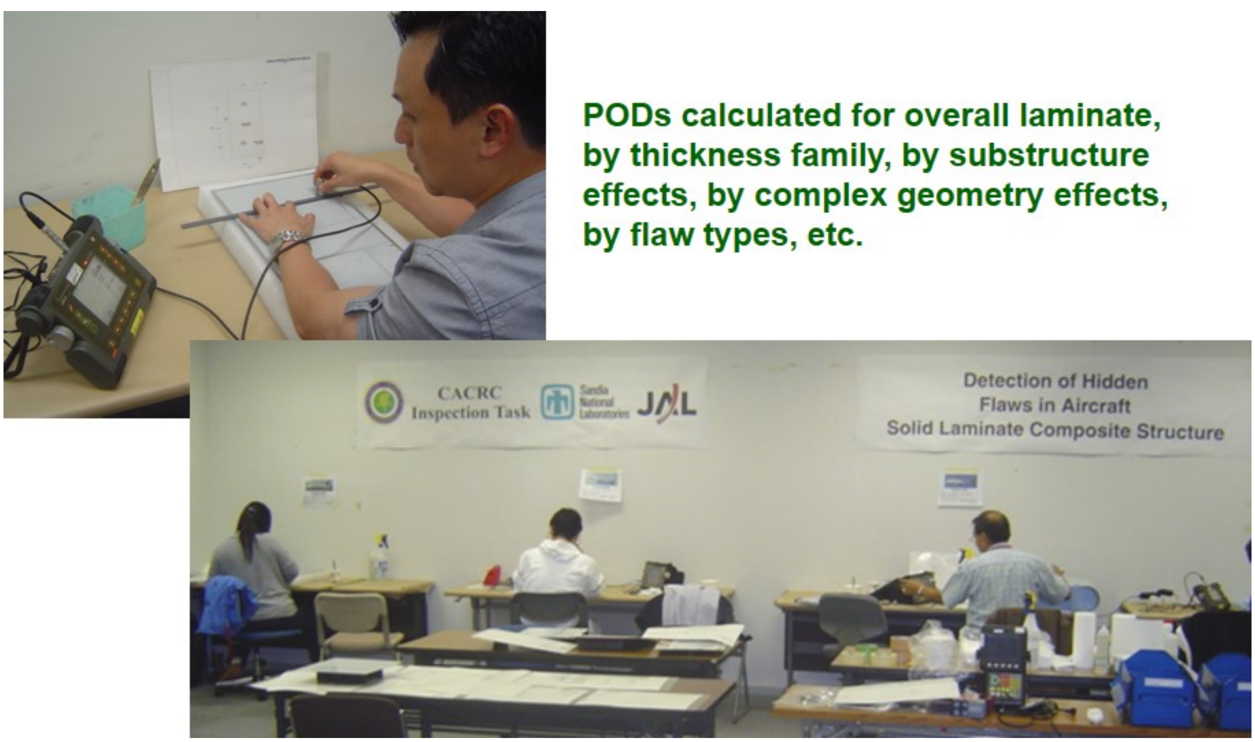
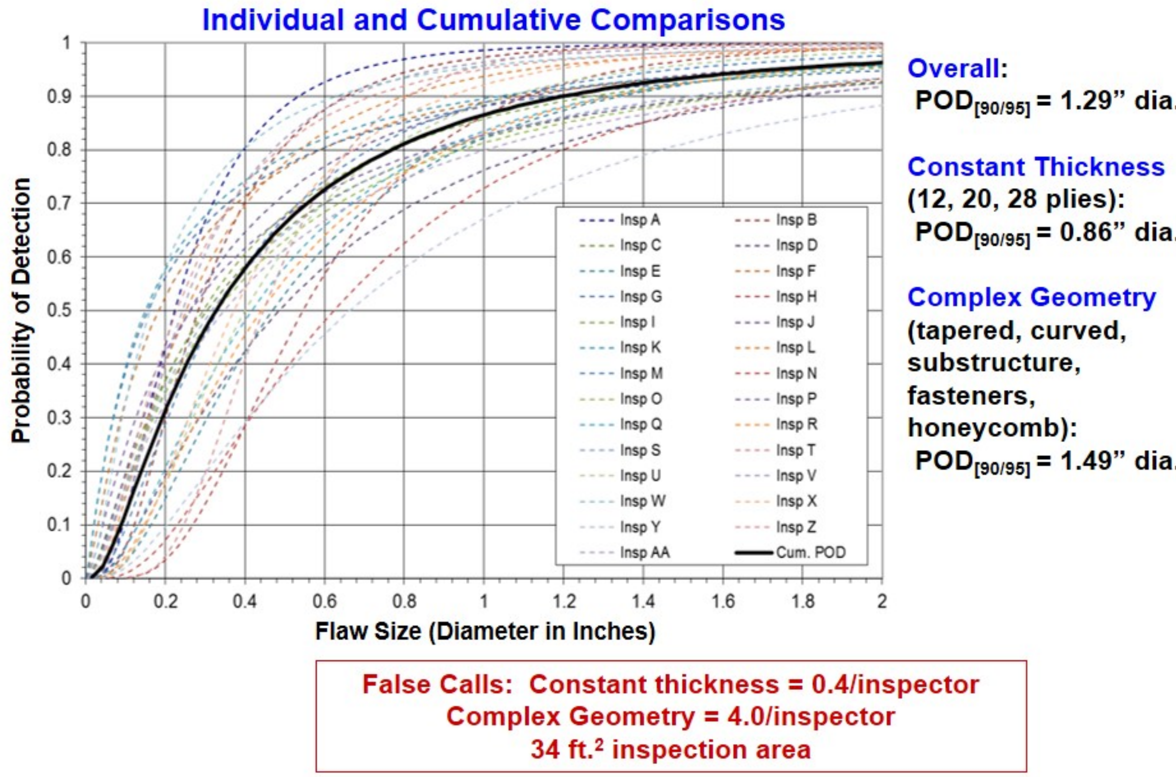


Figure 35. Implementation of solid laminate flaw-detection experiment

(a)



(b)

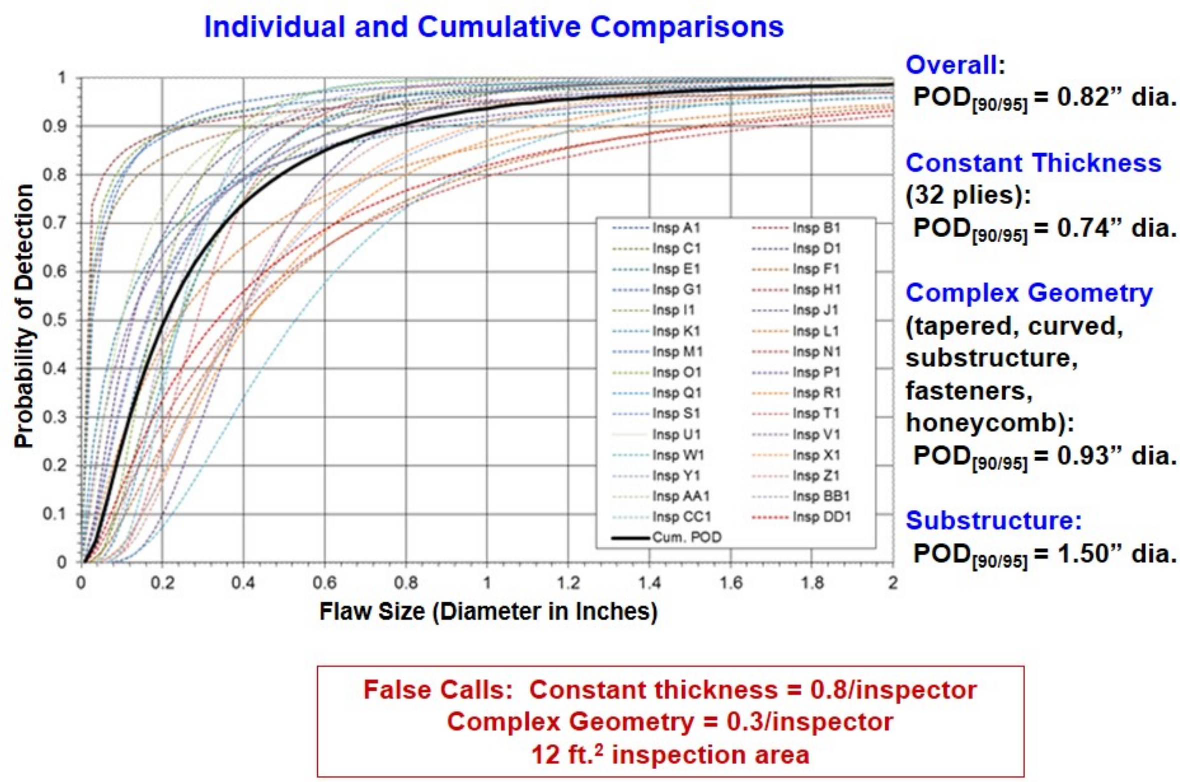


Figure 36. Probability of damage detection curves for 32-ply solid laminate family

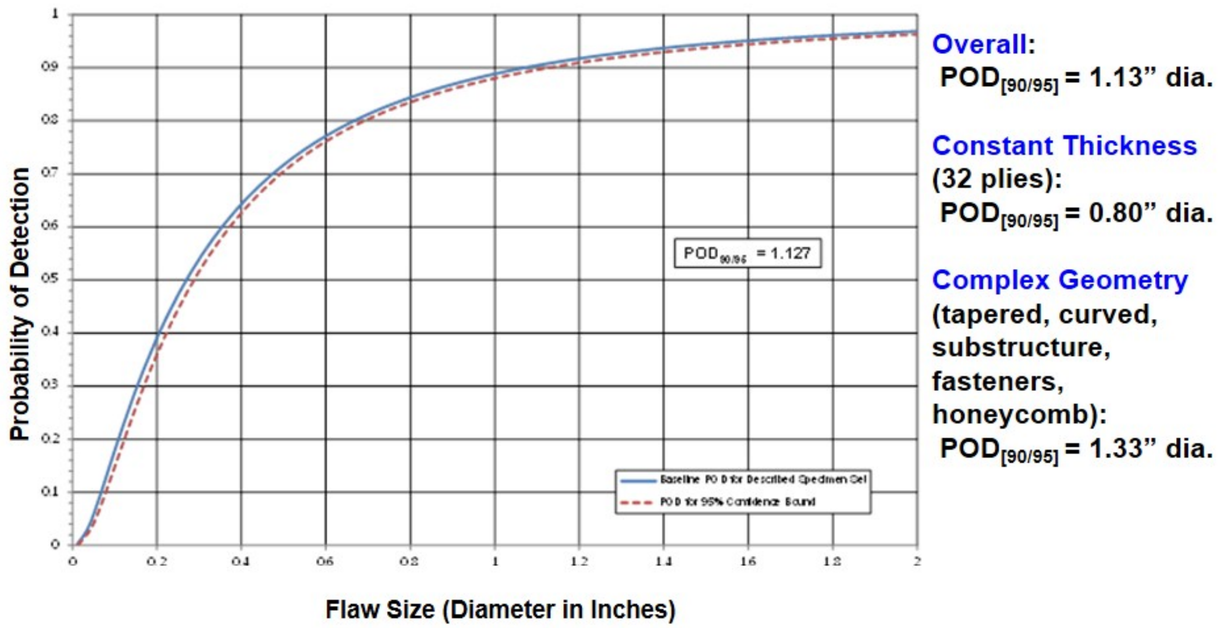


Figure 37. Overall performance of pulse-echo UT for flaw detection in composite laminates

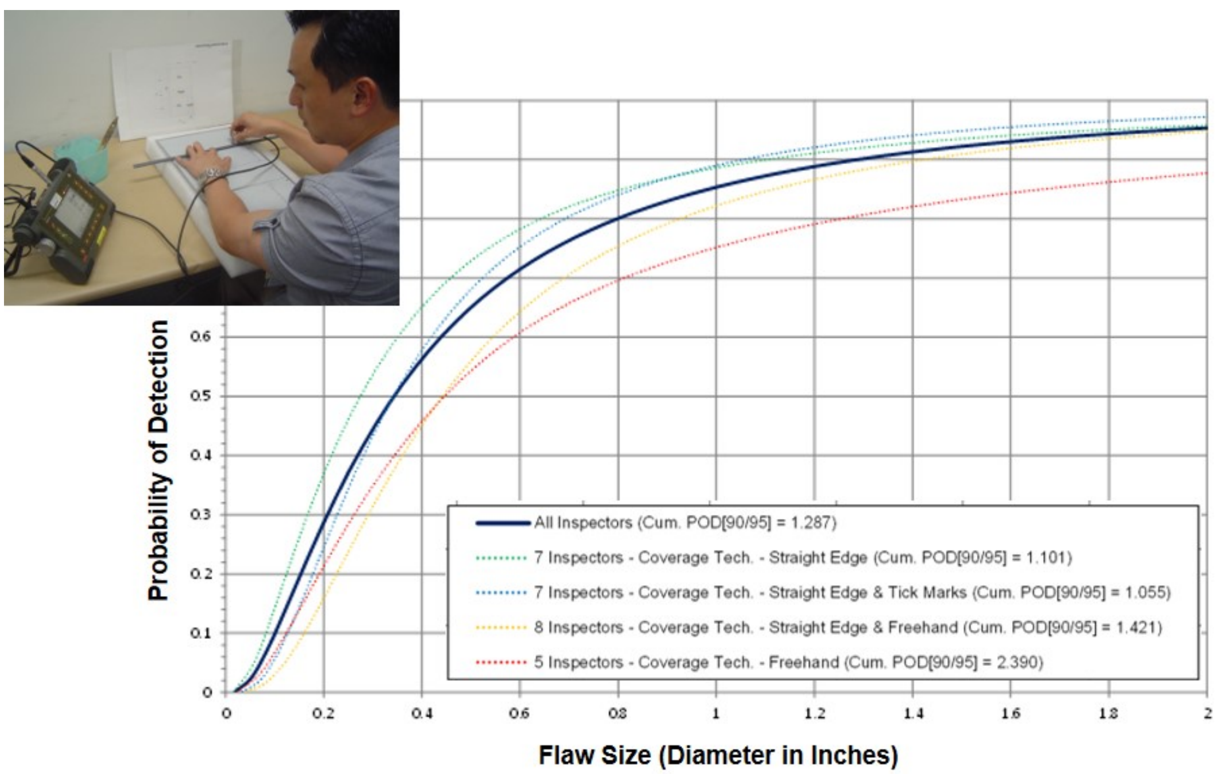


Figure 38. POD improvements from use of methods to ensure proper coverage

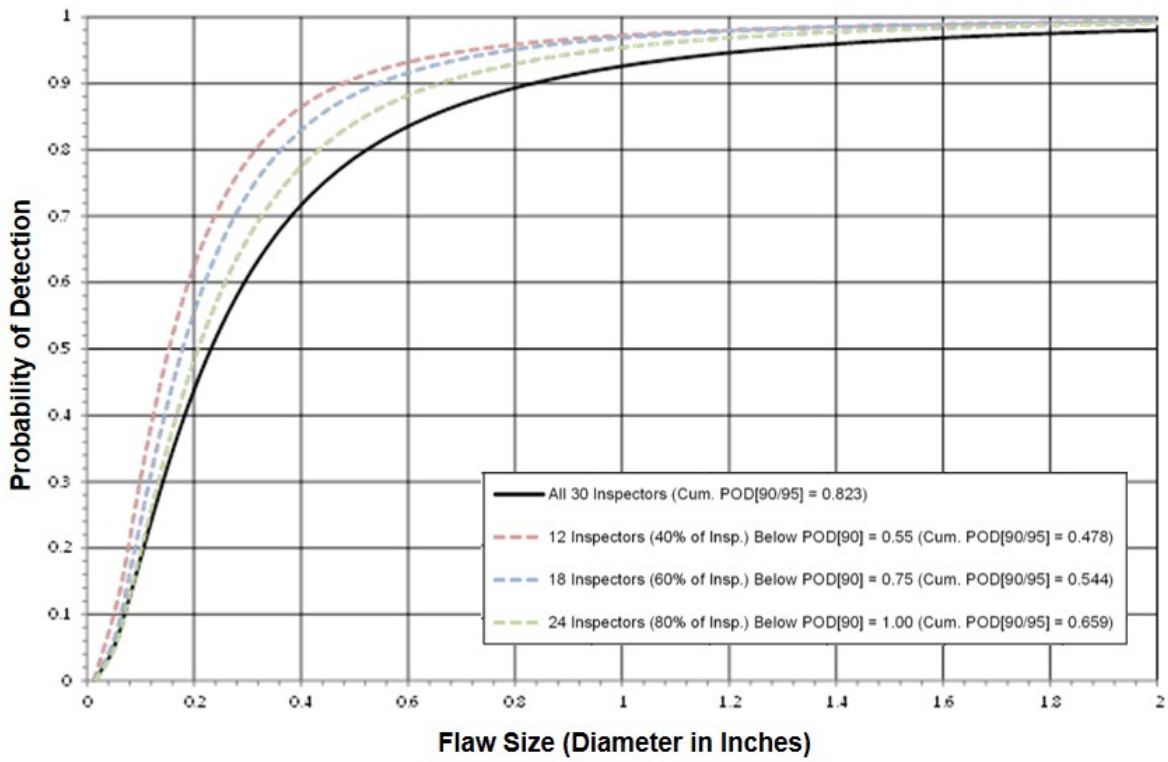


Figure 39. Desire to transition inspectors from "average" to "good" to "outstanding"

- Pulsed Thermography – Thermal Wave Imaging
- Phased Array UT – Olympus Omniscan
- Phased Array UT – Toshiba Matrix Eye
- Phased Array UT – Sonatest RapidScan
- Phased Array UT – Boeing MAUS V
- Phased Array UT – GE RotoArray & Phasor
- Microwave – Evisive
- Digital Acoustic Video – Imperium Acoustocam
- Ultrasonic Video - DolphiCam
- Line Infrared – Mistras
- Shearography – LTI
- Shearography – Dantec
- Backscatter X-ray – Scannex
- Acousto Ultrasonics – Physical Acoustic Corp. T-SCOUT
- Locked-In Infrared – MovieTherm
- Laser UT – iPhoton

Figure 40. Solid laminate experiment—Advanced NDI testing evaluations

Figure 41. Wide area and C-scan inspection methods

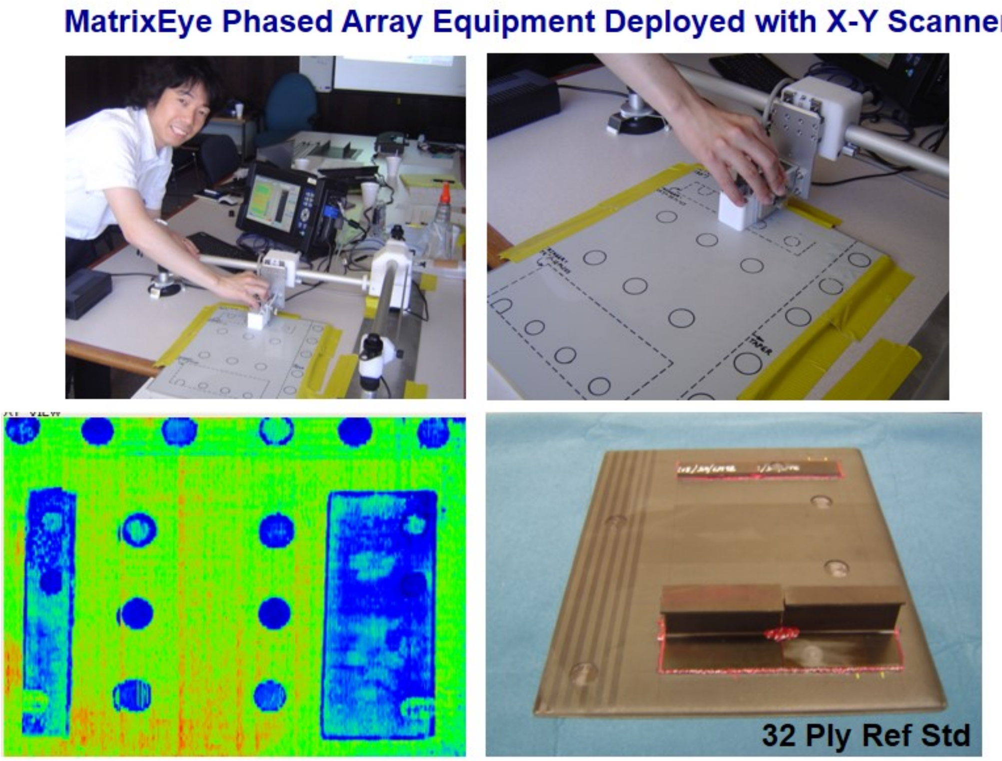


Figure 42. Initial inspections on feedback panels

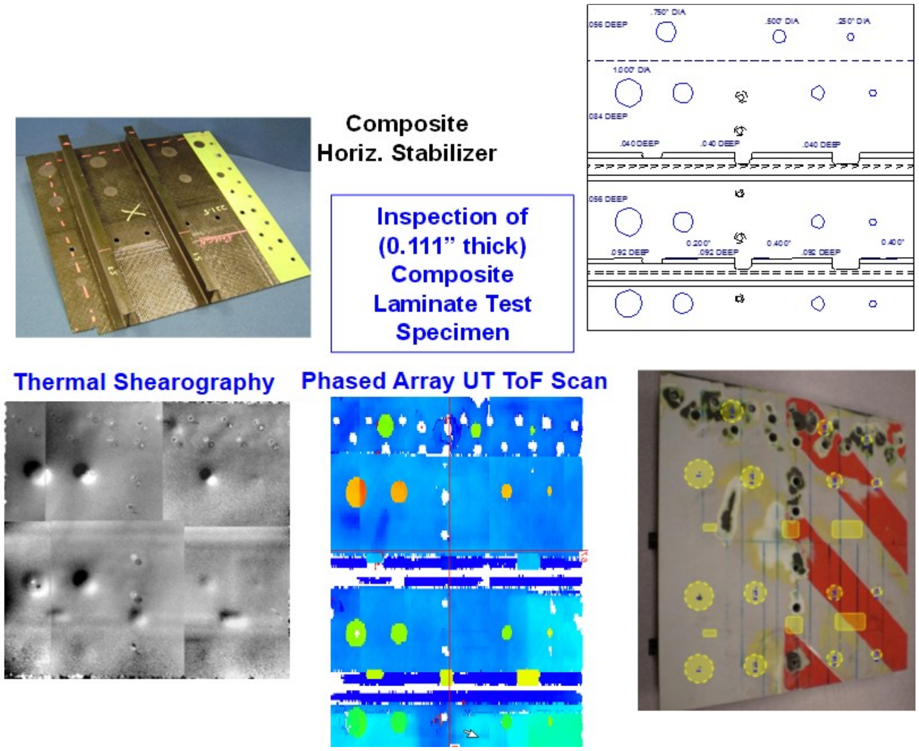
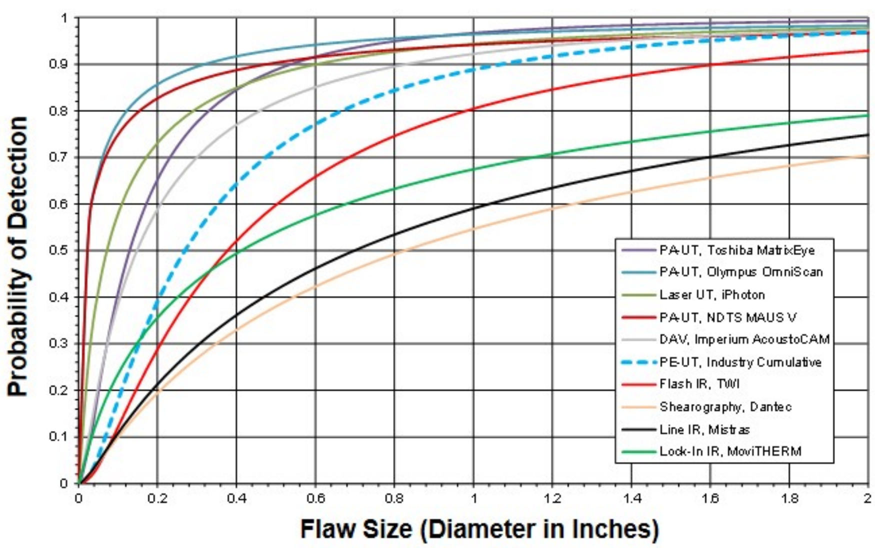


Figure 43. Preliminary NDI assessments of advanced NDI methods



	PA-UT, Toshiba MatrixEye	PA-UT, Olympus OmniScan	Laser UT, iPhoton	PA-UT - NDTS MAUS V	DAV, Imperium AcoustoCAM	PE-UT, Industry	Flash IR - TWI	Shearography - Dantec	Line IR - Mistras	Lock-In IR, MoviTHERM
POD[90/95]	0.69	0.72	0.85	0.88	1.12	1.13	2.3	>3	>3	>3
False Calls	1	0	0	0	0	5.5	0	1	NA	26

Figure 44. Sample of POD results for composite flaw detection performance of advanced NDI

- Results can be used by OEMs and airlines to: 1) define detectability for various flaws/damage, 2) guide NDI deployment & training; used by FAA to produce guidance documents.
- Flaw Detection - Conventional, manually-deployed PE-UT: $POD_{[90/95]}$ = 1.12" dia.; skin flaw detection is higher, flaw detection in substructure is more challenging ($POD_{[90/95]}$ = 1.34" dia.)
- False Call rates were extremely low: 1 false call per 17 $ft.^2$ (flaws ≥ 0.25 $in.^2$)
- Optimum inspection rates = 2 $ft.^2$ /hour
- NDI Performance Obstacles – attenuation, complex signal reflections, confounding presence of signal harmonics, rapid variations produced by changing/complex geometry, optimum deployment and difficulty with inspecting large areas

Figure 45. Conclusions—Inspection of solid laminate structures

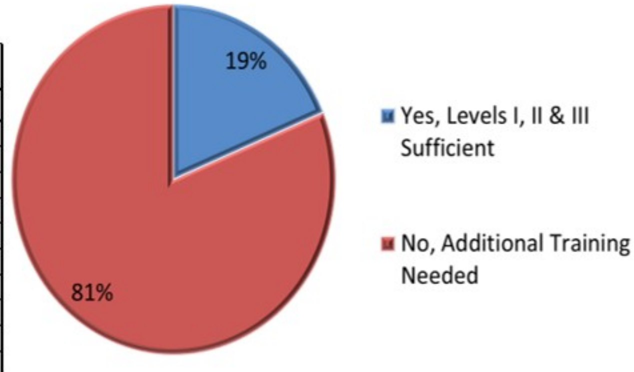
- Increased exposure to representative composite inspections – common industry NDI Proficiency Specimens
- Increased, focused composite NDI training
- Use of NDI and composite shop apprenticeships (OJT, awareness training, formal/uniform use of this tool)
- Enhanced NDI procedures – deployment, signal interpretation, clearer schematics showing structural configuration
- Use of inspection coverage aids should be required
- Divide large area inspections into a number of smaller regions
- Reiteration of best practices & use of NDI apprenticeships
- Guidance on addressing complex geometry challenges
- Prepare additional industry guidance to address training, use of NDI Reference and Proficiency specimens, procedures, composite construction awareness – produced by joint effort of OEMs, Airlines, FAA & industry groups

Figure 46. Recommendations—How to move inspections from "average" to "good" to "outstanding"

Question 16 - In your opinion, do Level I, II, and III training/qualifications provide the necessary expertise for both metal and composite NDI or should additional training take place for composite inspections?

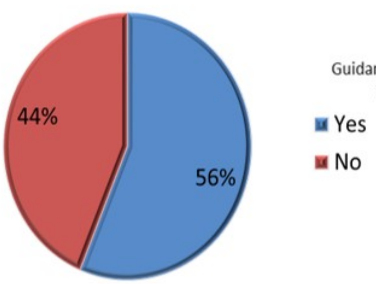
Composite NDI Training Survey Participants

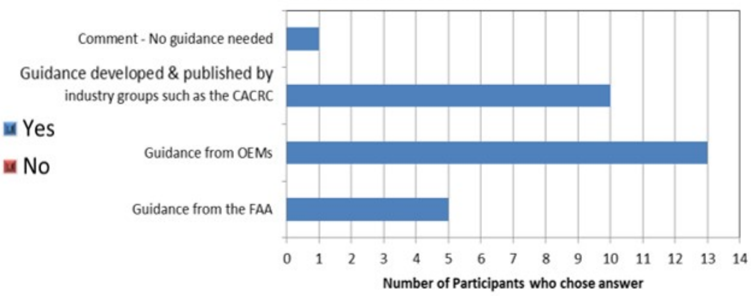
Company	Completed Survey
AAR-ASI (Indy)	Yes
American Airlines (Tulsa)	Yes
Aviation Technical Services, Inc (Seattle)	Yes
Delta Air Lines (Atlanta)	Yes
Delta Air Lines (MN)	Yes
FedEx (Indy)	Yes
FedEx (Los Angeles)	Yes
Goodrich Aerostructures (Chula Vista)	Yes
Kalitta Air LLC (Michigan)	Yes
Rohr Aero Services LLC (Alabama)	Yes
Southwest Airlines (TX)	Yes
Timco (Georgia)	Yes
United Airlines (Houston)	Yes
United Airlines (San Fran.)	Yes
UPS (KY)	Yes
US Airways (PA)	Yes



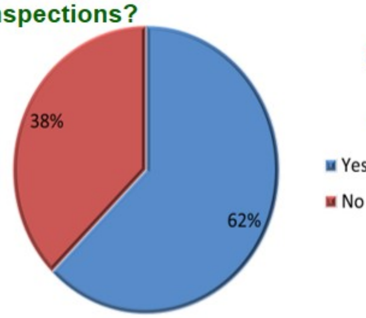
Only 25% of responders currently have special composite NDI training in place

Question 21 - In what areas is additional guidance needed to help ensure comprehensive composite training programs for the aviation industry?





Question 15 - If experience level is a factor in determining qualification to perform certain inspections, do you use some sort of apprentice program to expose newer inspectors to such inspections?



Question 5 – Do inspectors also receive general composite training to understand composite materials, plies, lay-ups, scarfed repairs, composite design, composite processing, etc.?

Figure 47. Survey of industry composite NDI training

Figure 48. Solid laminate flaw detection training insights

4. MODULE 2: COMPOSITE AWARENESS—MATERIALS, DESIGN, FABRICATION, AND USE

Module 2 in the Composite Inspector Training Course lecture series addresses composite awareness including an overview on composite materials, composite structural design, component fabrication methods, and general composite material use. Figures 49–107 encompass all of the lecture materials for "Module 2: Composite Awareness." It is believed that if inspectors are provided with some background on composite materials and how they are designed and fabricated, it will help them better understand how their inspection system interfaces with these structures.

Composites are combinations of two or more distinct materials present as separate phases and combined to form desired structures and achieve specific structural properties. They take advantage of the desirable properties of each component. The properties of the composite material are superior to the properties of the individual materials from which it is constructed. An advanced composite material is made of a fibrous material embedded in a resin matrix, generally laminated with fibers oriented in alternating directions to give the material strength and stiffness. The primary advantages of composite materials are their high strength, relatively low weight, ability to tailor designs to meet specific anisotropy needs, excellent damage tolerance, and resistance to corrosion.

The manufacturing technique used to fabricate a composite structure is dependent upon material performance requirements, structure configuration, and production rates. The use of composite materials is becoming more important in the construction of aerospace structures, especially for primary structures. Aircraft parts made from composite materials, such as fairings, spoilers, and flight controls, were developed during the 1960s mainly for their weight savings over aluminum parts. These were mostly composite honeycomb construction. Recently, aircraft have been introduced with all solid laminate composite fuselage and wing structures. The repair of these advanced composite materials requires in-depth knowledge of composite structures, materials, and tooling.

An isotropic material (e.g., aluminum, titanium) has uniform properties in all directions. The measured properties of an isotropic material are independent of the axis of testing. Conversely, a fiber is the primary load-carrying element of the composite material, and the composite material is only strong and stiff in the direction of the fibers. Unidirectional composites have predominant mechanical properties in one direction and are said to be anisotropic, having mechanical and/or physical properties that vary with direction relative to natural reference axes inherent in the material. Fibers can be manufactured from many different materials and the primary aviation components are produced from fiberglass, Kevlar®, and carbon graphite fibers. A matrix, usually some form of a resin material, supports the fibers and bonds them together in the composite material. The matrix transfers any applied loads to the fibers, keeps the fibers in their position and chosen orientation, gives the composite environmental resistance, and determines the maximum service temperature of a composite [15].

Components made from fiber-reinforced composites can be designed so that the fiber orientation produces optimum mechanical properties. Structural properties, such as stiffness, dimensional stability, and strength of a composite laminate, depend on the stacking sequence (orientation) of the plies. This module provides students with some background on the various materials that are used to construct composite aircraft parts. It introduces students to the following aspects of composite material makeup, design, and construction: 1) composite usage including advantages and limitations, 2) materials, 3) design using composites, 4) construction and array of fabrication processes, 5) composite repairs, and 6) types of defects and damage found in composites. These major topics are further broken down into their sub-elements as follows:

- Drivers for composite usage
- Composite honeycomb versus composite solid laminate structures
- Various fibers and matrix materials
- Use of pre-impregnated tape or wet lay-up dry fiber materials
- Fiber orientation – uniaxial fibers and woven fabric fibers
- Design – tailoring composite parts to produce needed properties; damage tolerance; composite joining methods
- Ply lay-ups to produce composite laminates
- Composite part manufacture using various heat and pressure methods
- Vacuum bagging, autoclave vessels and Vacuum Assisted Resin Transfer Molding
- Sample composite structures in aircraft
- Defect and damage types and damage severity categories – voids, porosity, resin rich/starved, fiber waviness, inclusions, fluid ingress, erosion, impact, heat exposure, weak bonds or disbonds, fatigue, and fiber fracture, lightning damage
- Composite safety
- Types of composite repairs and post-repair inspections
- Introduction to different composite inspection methods

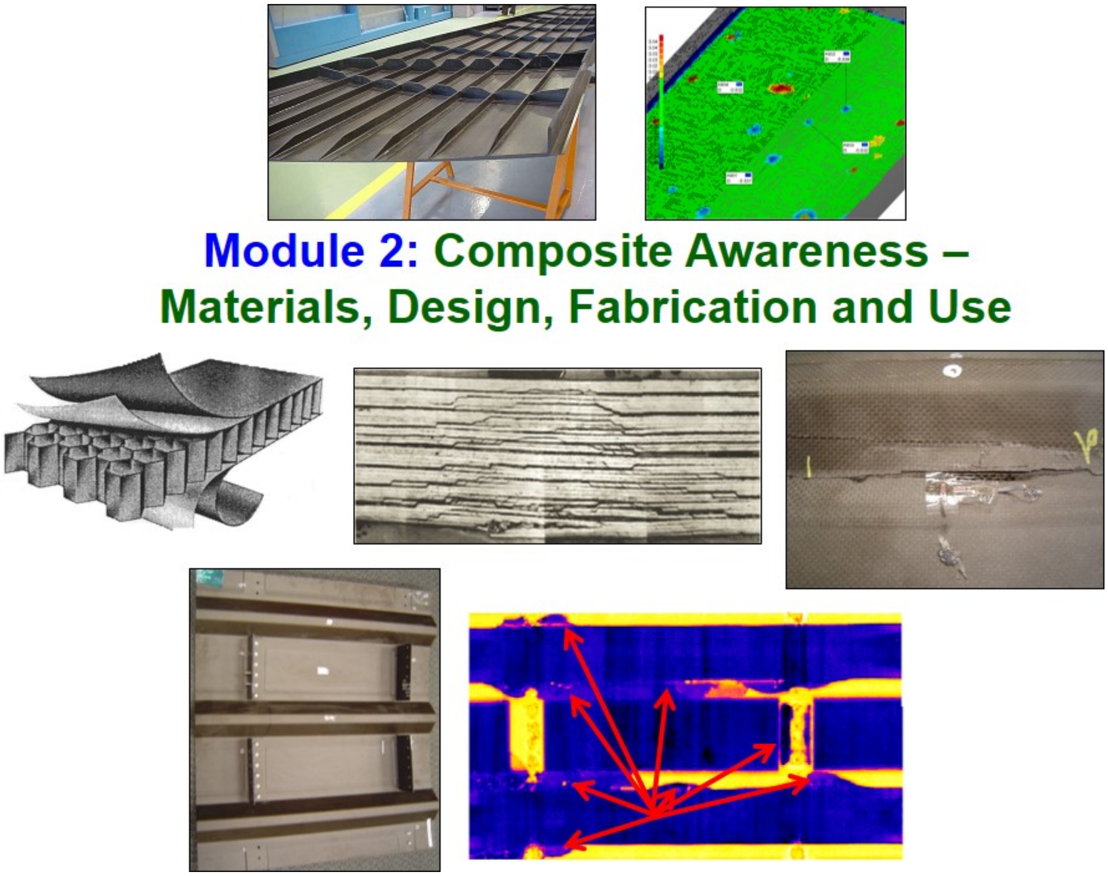


Figure 49. Composite laminate NDI training class

Class Definition - Composite Awareness

- **Advantages and Limitations of Composites – usage and applications; engineering properties**
- **Materials – carbon, fiberglass, resin, uniaxial, weave, pre-preg, wet lay-up; peel plies; thermal set vs thermal plastic; safety in handling; time sensitivity**
- **Design & Construction – ply orientations, honeycomb, laminate, skin/substructure, ply taper, co-cured and secondarily-bonded adhesive joints, pad-ups, use of sealants & fasteners: lightning protection & corrosion protection schemes, complex geometry (radius, spar caps); noise protection (tiles)**
- **Fab Processes & Tooling – lay-up process, vacuum bag, Autoclave; bonding processes**
- **Composite Repairs**

Class Definition - Composite Awareness

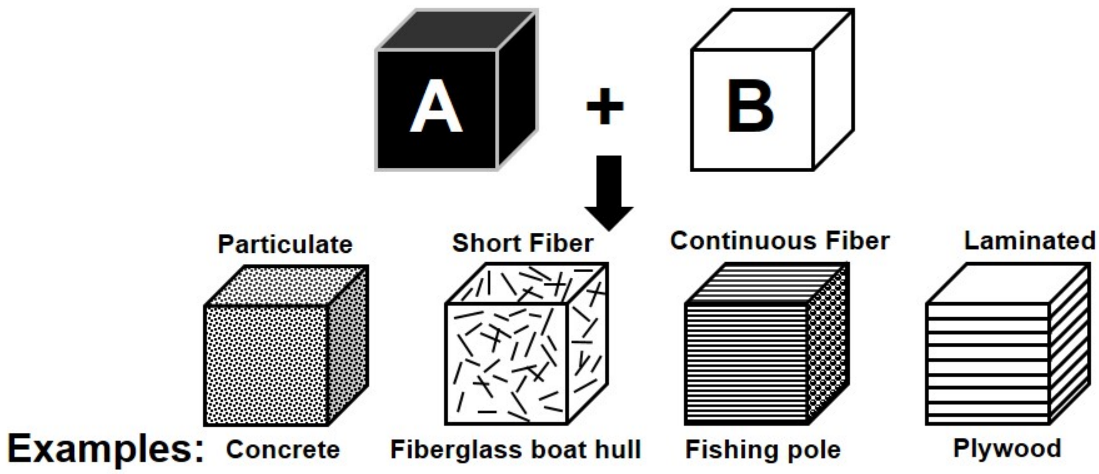
- **Composite Defect Types - disbonds, delaminations, impact, microcracking, moisture ingress, porosity, adhesive voids, resin rich/starved, fiber waviness, FOD inclusions**
- **Manufacturing of Composites**

- Autoclave
  1. Description of Autoclave process
  2. Vacuum Bagging Process
  3. Autoclave Curing Process
  4. Ramp Up and Down's
  5. Advantages
  6. Disadvantages
- Other Devices for Curing Composites
  1. Heated Platen Process
  2. Industrial Ovens
  3. Heat Blankets
  4. Double Vacuum Bagging

- Co-Cured – simultaneous cure of all green components
- Co-bonded (one green, one pre-cured, no added adhesive layer)
- Secondarily Bonded- (two separate cured items, bonded with an adhesive layer)

Figure 50. Composite laminate NDI training class—Class definition for composite awareness module

- Most broadly - a composite is a combination of two materials, a reinforcing material that provides strength and stiffness and a stabilizing matrix that distributes the load and protects the reinforcement.
- Two (or more) distinct phases remain present in the resulting material



Many composite aircraft parts are both continuous fiber and laminated

- Natural composites: wood, bone and mollusk shells
- Early manmade composites: adobe, concrete and plywood
- Composites are typically categorized by what type of matrix material they use:
  - Polymer matrix composites (PMCs) – aircraft parts, wind turbines, boat hulls, automotive parts, sporting equipment, bathtubs, pipes
  - Metal matrix composites (MMCs) – cutting tools, automotive engine parts, high end structural parts
  - Ceramic matrix composites (CMCs) – heat shields, gas turbine parts, brake disks, tank armor
- These materials remain separate in the conglomeration and contribute their individual properties to produce a new material that is stronger than each could be separately.
- The inherent flexible nature of these constituents allow for easy molding and forming of complex geometries.

Figure 51. What are composites?

- **Focus on continuous fiber reinforced polymer matrix composites because these are what are typically used in large aircraft structures.**
- **The single most important difference is that unlike metals, composite structures (with a few exceptions) have mechanical properties that vary depending on direction (anisotropic).**

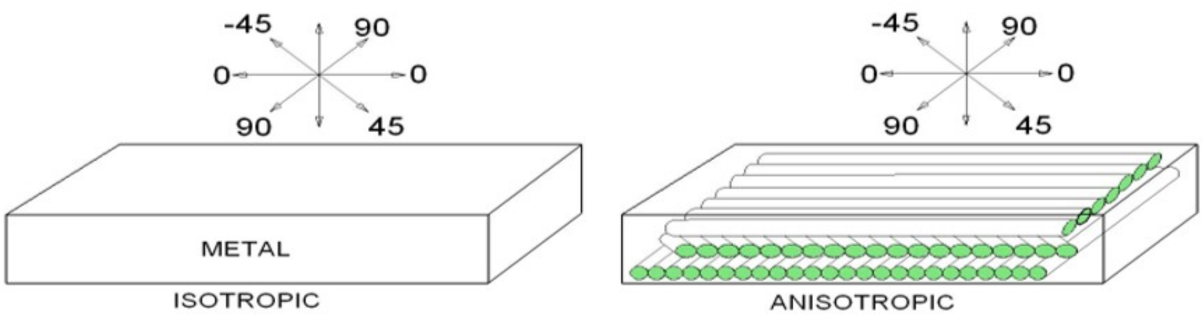


Figure 51. What are composites? (cont.)

- High strength and stiffness for their weight (specific strength and stiffness)
- Allows for optimized structural performance (anisotropic).
- Improved fatigue and corrosion performance over metals
- Forming of complex geometries is possible
- Multiple assembly parts can be integrated into one composite structure
- Additional materials can be added to improve properties such as RF transparency and thermal, chemical, and environmental resistance.

In 1963, the Polaris A3 rocket was developed using composites - this added strength, reduced weight, increased the range & reduced the cost by 60%

- The use of composite materials in aerospace started on military aircraft and has grown on commercial planes over the decades.
  - 1% of the weight of 747 in 1969.
  - 11% of the weight of 777 in 1995
- Boeing 787
  - 50% of the 787's weight
  - Holes drilled into fuselage during assembly: 1 million holes for a 747 vs fewer than 10,000 (1%) for the 787
- Airbus A350 XWB
  - 53% of the A350's weight
  - Uses conventional panel design vs 787's single piece barrel

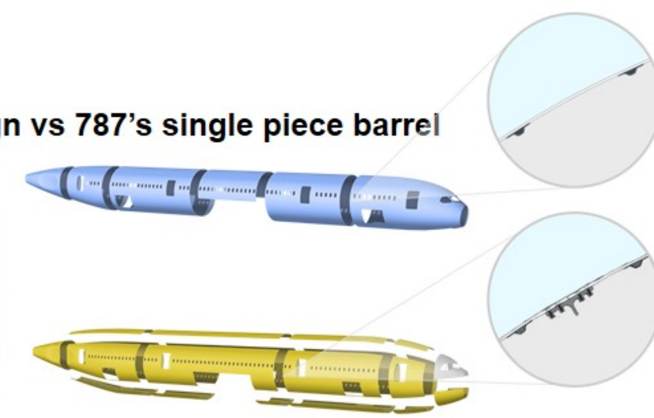


Figure 52. Why composites?

- **Silicon Dioxide (SiO_2) based fibers**
  - E-glass: most common type of fiber for use in composites, but also the lowest performance; high strength, low stiffness, and low cost. Contains various metal oxides and other mineral additives
  - S-glass: much higher strength than E Glass.
  - Quartz: high strength at high temperatures, excellent electromagnetic properties useful for fabricating parts such as aircraft radomes; expensive to produce. Nearly pure SiO_2
- **Carbon/Graphite: high strength and high stiffness; expensive to produce**
- **Boron: stronger than carbon fiber; excellent compressive properties**
- **Aramid: very tough and abrasion resistant, good temperature resistance, prone to water absorption**
  - Kevlar is a sub type (and brand name) with higher stiffness and strength
- **Ultra-high-molecular-weight polyethylene (UHMWPE). Dyneema and Spectra brand fibers: very light weight, less prone to water absorption than Aramid but lower max temperature**

- **Fiber reinforcements play a major role in defining the strength of a composite material.**
- **The fiber comprises most of the volume and is responsible for carrying the tensile, flexure, and compressive loads.**
- **Fibers have higher strength (due to fewer defects) and greater flexibility than bulk materials as a result of their small size.**
- **Most of the common aerospace fibers are solid and range in size from 7 to 150 microns. By comparison, a human hair is roughly 70 microns in diameter.**
- **Fiber length is another consideration. Most high performance composite materials are made of extremely long unbroken fibers. In use, this continuous fiber property contributes to the toughness and load bearing capacity of the component.**
- **Fibers must be appropriately prepared for the resin system they will use to ensure a good bond between them and make full use of their strength**

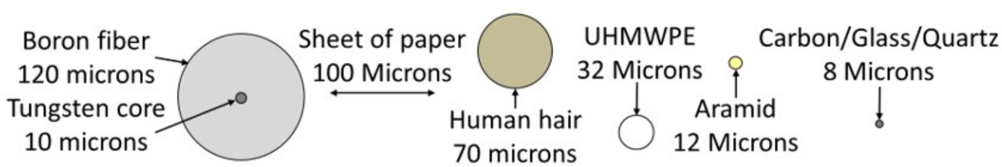
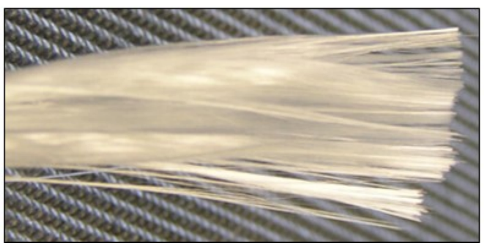


Figure 53. Fiber-reinforcement types

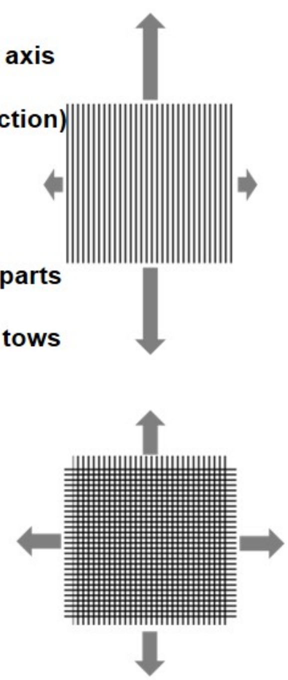
- **Filament**: This is the building block of fiber reinforced elements. They are produced in a drawing or spinning process. They are continuous and can be as long as 1000 ft in length. They are generally too small in diameter to be used alone as a structural element, so they are bundled to produce materials for composites.
- **Yarn**: a collection of individual filaments (less than 10,000 in total) twisted together to form a useable element suitable for weaving textiles.
- **Tow**: a collection of individual filaments that are not twisted.
  - Generally come in 1,000 (1K) 3,000 (3K) 6,000 (6K) and 12,000 (12K) filament counts.
  - Lower filament counts (1K to 6K) are used to make cloth or textiles.
  - 12K and up are generally reserved for filament winding process and manufacture of unidirectional (Uni) Tape.



Fiberglass tow with filaments visible

Figure 53. Fiber-reinforcement types (cont.)

- Unidirectional
  - All fibers are oriented in the same direction along the longest axis (longitudinally)
  - Maximum structural properties in one direction (the fiber direction)
  - Can be difficult to drape when laying up a part
- Cloth types
  - Two-directional woven fabric of unidirectional tows
  - Provides structural properties in two directions
  - Good ability to drape and ease of processing when laying up parts
- Hybrids
  - Blend of two directional woven cloth knitted to unidirectional tows can provide the best of both worlds

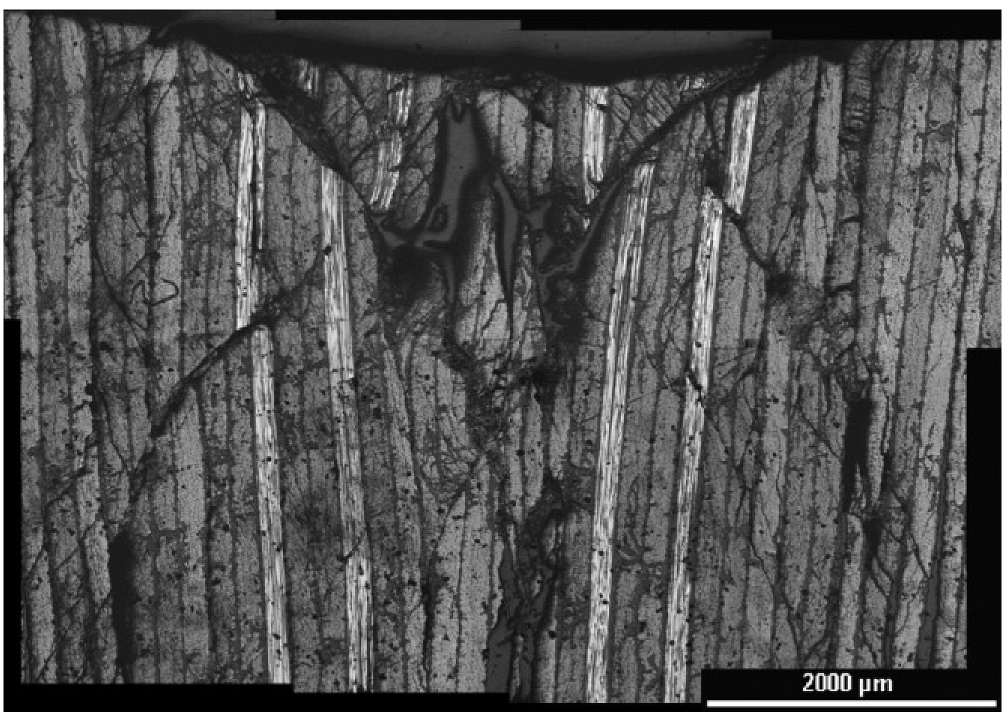


Percentage of fiberglass reinforcement increases strength in direction of fiber orientation		
Unidirectional Fiber Orientation		**Reinforcement types:** Continuous strand roving **Processes:** Continuous pultrusion, compression molding
Bidrectional Fiber Orientation		**Reinforcement types:** Continuous strand roving **Processes:** Filament winding, compression molding **Reinforcement types:** Woven fabrics, woven roving **Processes:** Hand lay-up
Multidirectional FIber Orientation		**Reinforcement types:** Chopped strands, continuous, chopped strand mat tri axial fabric **Processes:** Compression and injection modling, spray-up pressure bag, preform

Figure 54. Fiber reinforcement—Fiber formats

- Fibers are often processed into planar forms as a first step towards manufacturing composites. Often described by areal weight (e.g. oz/yd^2)
- Fiber volume fraction plays a critical role in determining strength of a composite
  - Want just enough matrix to support fibers, and transfer load and no more. This is on the order of 50%-65% for high performance aerospace composites.
  - Minimize voids
- Unidirectional composite can easily be 50x weaker and 20x less stiff transverse to the fibers rather than parallel.
- Because of this, composites are sensitive to fiber misalignment, a small shift in angle results in a significant reduction in stiffness and strength:
  - A unidirectional laminate loaded in tension is:
    - 15% less stiff and 47% weaker 5° off axis
    - 40% less stiff and 73% weaker 10° off axis
  - This is why laminates almost always use a variety of fiber orientations

Figure 55. Fiber-reinforcement methods



Different ply directions are clearly visible as are resin rich regions between plies. Photo is of impact damage on the edge of a composite plate.

Figure 56. Sample fiber reinforcement

- Filaments are stabilized at a micro level by the matrix material. The choice of matrix contributes to the part's mechanical properties.
- Some considerations:
  - Adhesive strength to take advantage of higher strength fibers (we need the resin to bond well to the filaments)
  - Compressive strength - because it stabilizes the fiber and keeps them in the proper orientation/position so they can carry the intended loads rather than buckling
  - Toughness – reinforcements generally have poor toughness so resins with high toughness can be employed to increase overall composite toughness
  - Heat resistance – typically related to the glass transition temperature (T_g) of the matrix
  - Environmental resistance (chemical, moisture, fatigue, *etc.*)
- Two broad categories: thermoset and thermoplastic
  - Thermosets are more common in aerospace
  - Thermoplastics have a couple advantages (toughness and some processing characteristics) that make them attractive in some circumstances

Figure 57. Resin matrix—Purpose

- **Thermosets – cure by forming crosslinks between molecules, heating eventually results in decomposition**
  - Polyester
    - Inexpensive, easier processing, good toughness, low mechanical properties, high shrinkage
  - Vinylester
    - Very high chemical and environmental resistance, better mechanical properties than polyester
  - Epoxy
    - High strength, excellent adhesion, brittle, sensitive to mixing ratios
  - Polyurethanes
    - High toughness
  - Phenolic
    - High service temperature, useful for fire resistance/low smoke generation, brittle, difficult processing
  - Bismaleimide (BMI)
    - High service temperature, high elongation at failure, high cost
  - Polyimide
    - Very high service temperature, brittle, very high cost
  - Cyanate ester
    - Low dielectric constant (often used in radomes)

- **Thermoplastic – Polymer chains interact through intermolecular forces, can be repeatedly softened and hardened with heating and cooling (i.e. no chemical curing)**
  - **Most everyday plastics are thermoplastics including: polyethylene (PE), polypropylene (PP), polystyrene (PS), polyvinylchloride (PVC), Polyamide (PA/Nylon), Poly(methyl methacrylate) (PMMA/Acrylic), polycarbonate (PC)**
  - **For use in composites common thermoplastics include: Polyetheretherketone (PEEK), Polyethylenimine (PEI), polyethersulfone (PES) and polyphenylene sulfide (PPS)**
  - **Advantages: Generally have high toughness, better chemical resistance, allow for fast cycle times (no need to wait for cure) and unlimited shelf life**
  - **Disadvantages: additional difficulties in processing (high viscosity, lack of tack and drape), high cost, low creep resistance, not as good in compression**

Figure 58. Resin matrix—Types

- **Ply (aka Lamina)** – A single layer of material used to build up the laminate
- **Laminate** – A thicker material assembled from multiple plies
- **Isotropic** – Properties (mechanical, thermal, electrical, etc.) the same in all directions
- **Anisotropic** – Different properties in different directions
- **Orthotropic** – Different properties in each of the three principal directions
- **Transversely Isotropic** – Same properties in one plane and different properties normal to this plane
- **Quasi-Isotropic Layup** – Isotropic properties in-plane. This can be accomplished by using 0º, 90º +45º and –45º or 0º, 60º and 120º oriented plies.

Figure 59. Composite terms

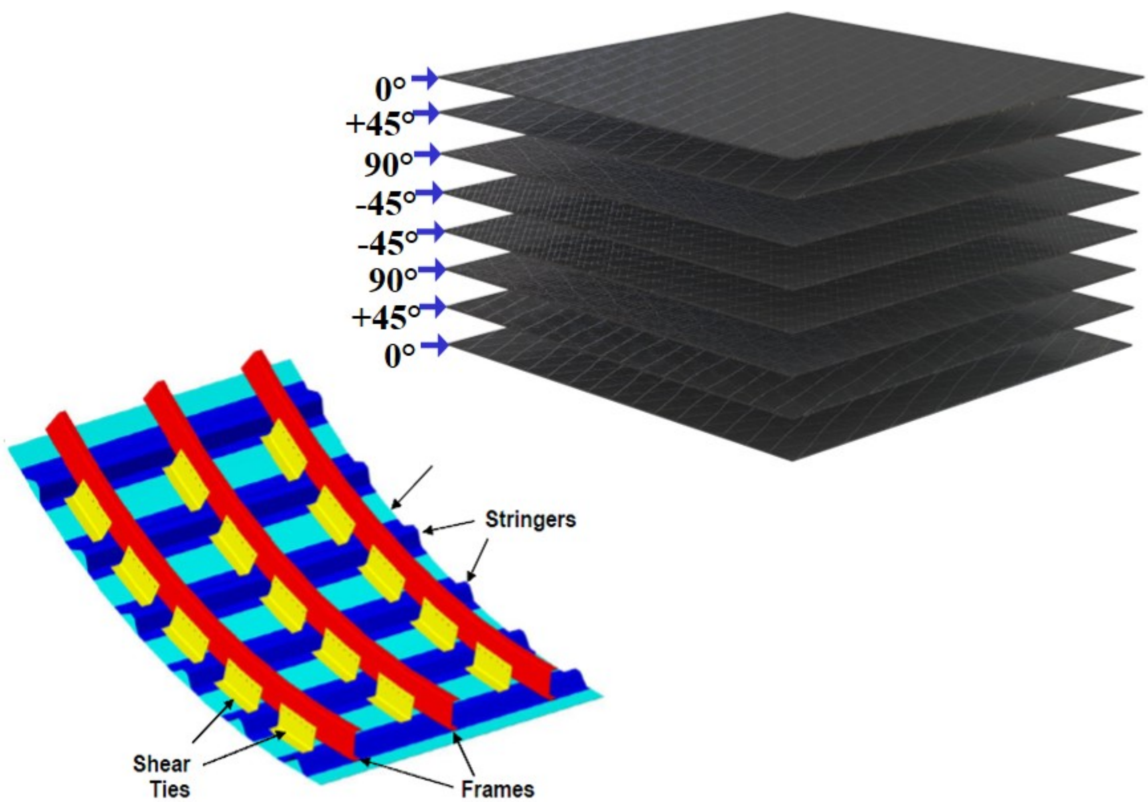


Figure 60. Ply lay-up to produce a laminate

- Way to describe the layup of the laminate
- Start with the first ply placed on the tool
- Each ply is indicated by the angle its axis should be rotated from the part zero axis (described in the manufacturing drawing). This is followed by a slash or comma
- A fabric ply is indicated with an "F"
- A plus/minus unidirectional ply means a positive and then a negative while a plus/minus fabric ply means it can be placed in either orientation
- A subscript number indicates multiple plies at the same angle
- An "S" after the closing bracket indicates a symmetric laminate with the plies mirrored on the other side of the mid-plane. A bar over the center ply is used for laminates with an odd number of plies.
- A "T" indicates a "Total" laminate. Used as an alternative to "S".
- Parentheses indicate repeated blocks

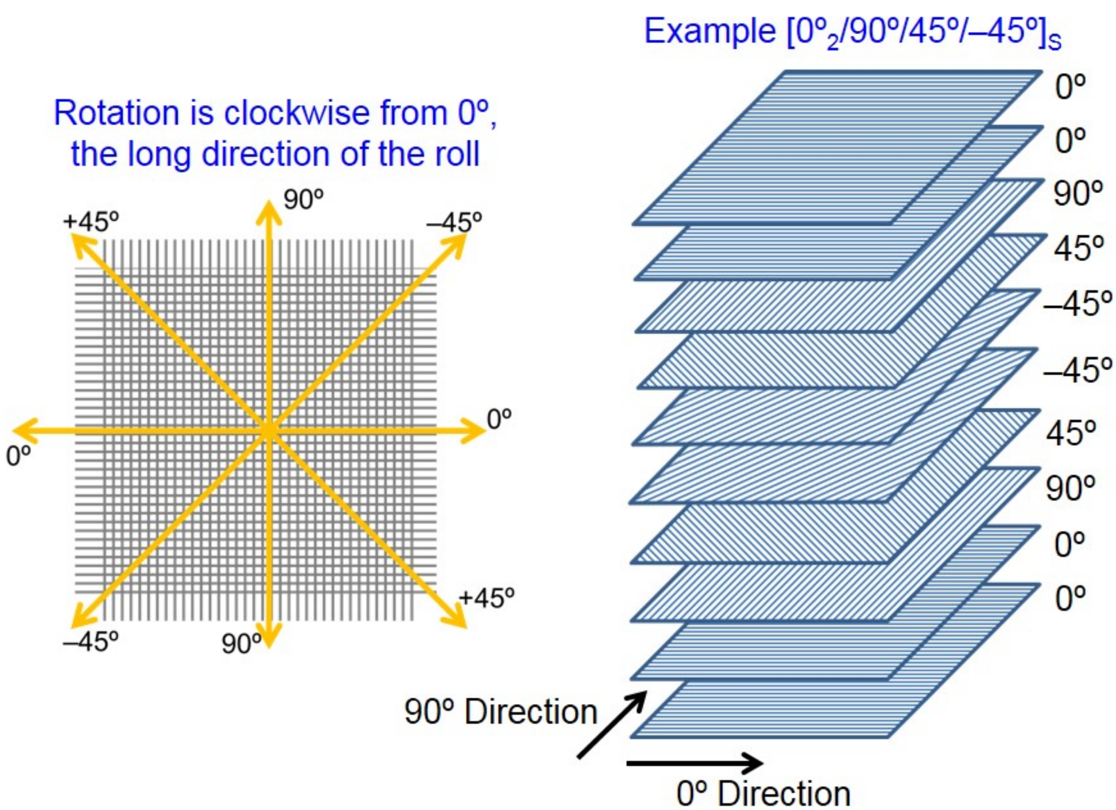
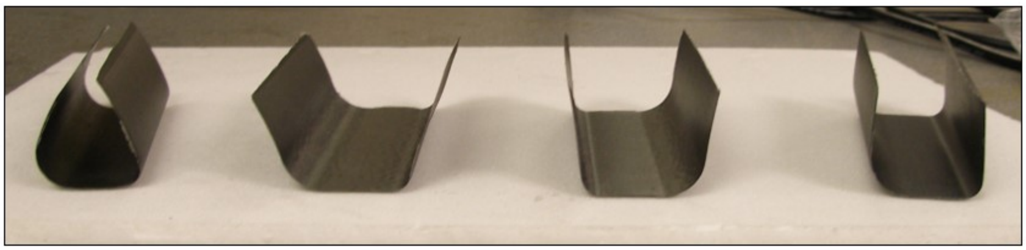


Figure 61. Laminate notation

- **Symmetric** - A laminate is symmetric when the plies on one side of the mid-plane are a mirror image of those on the other side. This avoids curving. Example: $[0°_2/90°/45°/45°]_S$
- **Balanced** - A laminate is balanced when each angled ply has a corresponding negatively angled ply. This avoids twisting. Example: $[0°_2/90°/45°/–45°]_2$
- **Balanced symmetric** lay-ups help to avoid thermal twisting and warping of parts as they experience temperature changes (from autoclave/oven during cure or while in use). Example: $[0°_2/90°/45°/–45°]_S$



These four parts all came off the same mold with different asymmetric 4 ply laminates. A symmetric laminate would produce a U-shaped part.

Figure 62. Laminate terms

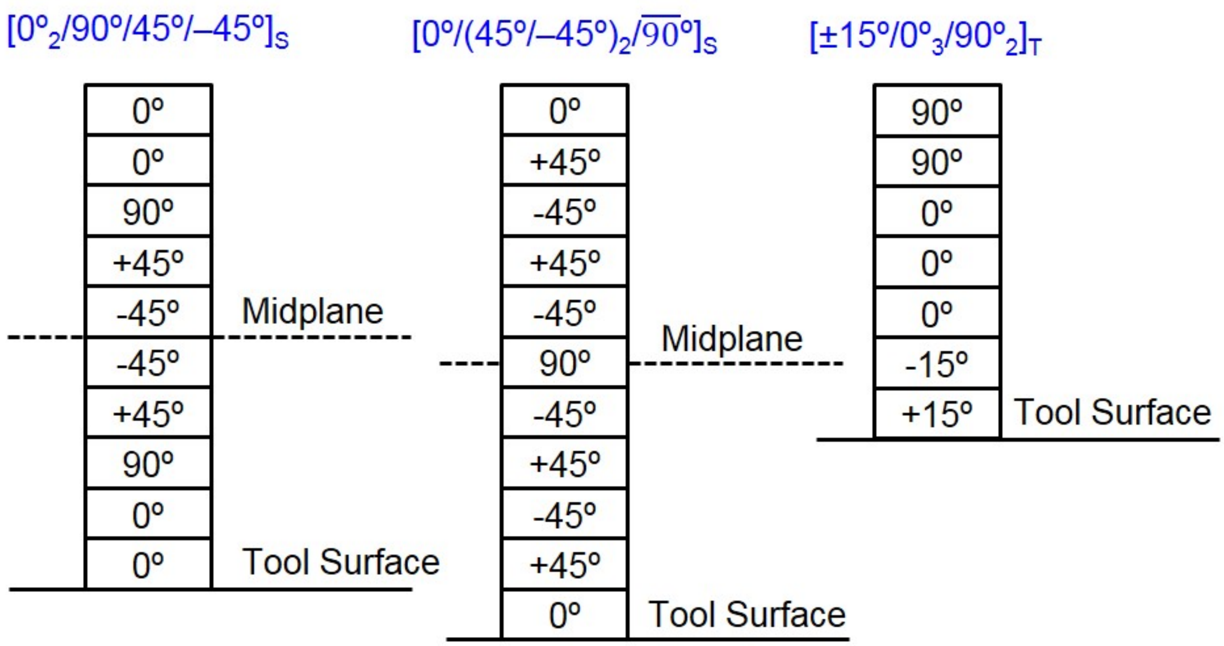


Figure 63. Laminate notation examples

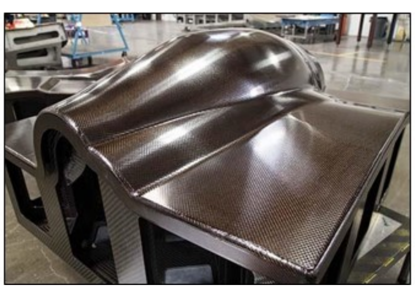
- Tooling
- Prepreg vs wet layup
- Vacuum Bagging
- Autoclave processing
- VARTM
- Automated methods

Laminate:
Vacuum-Bagged for Cure

Baron Autoclave

Figure 64. Composite manufacturing

- Used to control the shape of the part
- Tooling is most typically made from composites or metals
  - Metal more often used for long runs or tight tolerances
- Single sided tools result in only one side of the finished part with a controlled geometry and a fine finish
  - Male tools have laminates laid up on their exterior surface and produce parts with a smooth interior (e.g. bath tub)
  - Female tools have laminates laid up on their interior surface and produce parts with a smooth exterior (e.g. boat hulls)
- Matched molds control geometry and finish on both sides



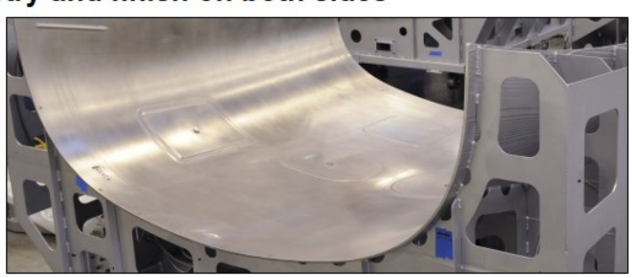


Figure 65. Manufacturing—Tools and molds

There are two main methods for applying resin

Pre-preg *(fiber impregnated with resin)*

- Advantages
  - Comes coated in resin
  - Correct resin to fiber weight ratio
  - Cleaner processing
- Disadvantages
  - More expensive
  - Storage (low temperature)
  - Cured at elevated temperature
  - Special equipment required
  - Limited shelf life

Wet Layup or Resin Transfer Molding

- Advantages
  - Cheaper
  - Easy to store
  - Curable at room or elevated temperature
- Disadvantages
  - Inconsistent resin to fiber ratio
  - Messy (coats equipment)
  - Hazardous clean up process

Figure 66. Resin matrix—Application

- Fabric (or unidirectional material) pre-impregnated with resin
- Process
  - Dry fibers – fabric or uni
  - "B-staged" (i.e. partially cured) resin matrix is infused using specialized equipment
  - Material placed onto rolls and into cold storage (0° C) to prevent further curing of resin
  - When ready to use, the material is removed from freezer and thawed to room temp (in a sealed bag to avoid moisture ingress)
    - Prepreg materials have an "out life" (room temperature) ranging from weeks to months

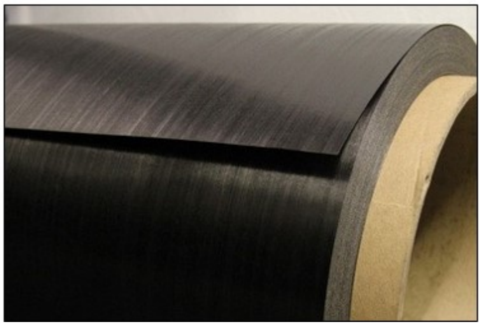
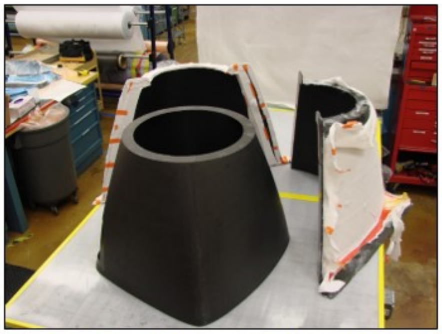


Figure 67. Manufacturing with prepreg materials

- Prepreg removed from cold storage and thawed
- Plies of prepreg are cut out from a roll by hand or using a CNC process. Shape is based on engineering drawings previously generated.
- Plies are laid onto tool one by one in the desired position and orientation
- Debulking to consolidate
- Final vacuum bagging and curing

Roll of Prepreg Material



Aft Cowl Demolded and Removed from Female Tool

Figure 68. Manufacturing with prepreg process

- Most basic method
- Mold is prepared with a release agent
- Liquid resin is applied to dry fabric
- Plies placed onto the tool one by one
- Part is then vacuum bagged to eliminate excess resin and provide compaction
- Can be cured at ambient conditions or placed in an oven.
- Low cost but slow and difficult to maintain consistent results.

Figure 69. Manufacturing with wet lay-up materials

- Vacuum bagging provides more manufacturing control to produce more consistent parts
  - Provides compaction to reduce voids and give better control of final shape
  - Provides a place for excess resin to go
  - Removes volatiles during cure

- Basic layers in order placed (i.e. going from laminate outward)
  - Peel ply or release film – provides consistent surface finish for post-cure bonding or painting and allows layers above to release from cured part
    - Peel ply is a woven material that allows excess resin to pass through
    - Release film is not penetrable but comes in perforated forms to allow excess resin to pass through
  - Bleeder – absorbs excess resin from the laminate
  - Release film to keep bleeder and breather apart
  - Breather – allows air passage to distribute vacuum evenly within the bag
  - Vacuum bag – air impemeable layer applies vacuum pressure evenly

Figure 70. Manufacturing—Vacuum bagging method

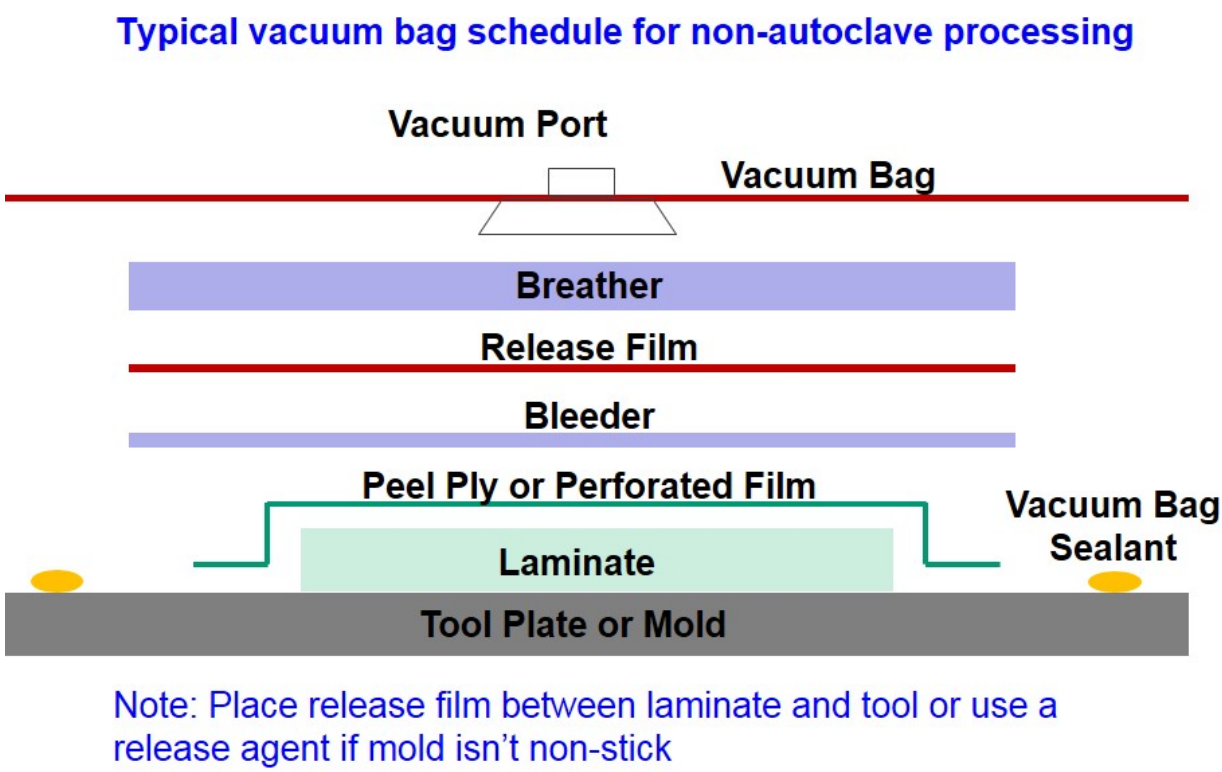


Figure 71. Manufacturing—Vacuum bagging lay-up

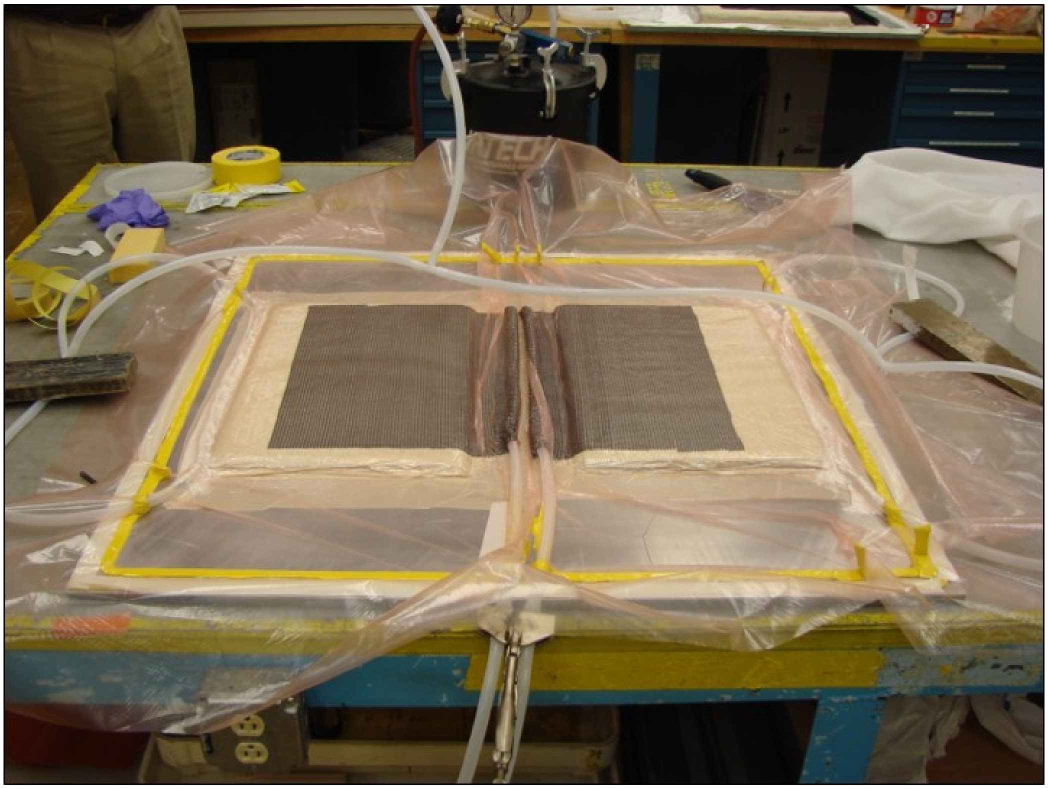
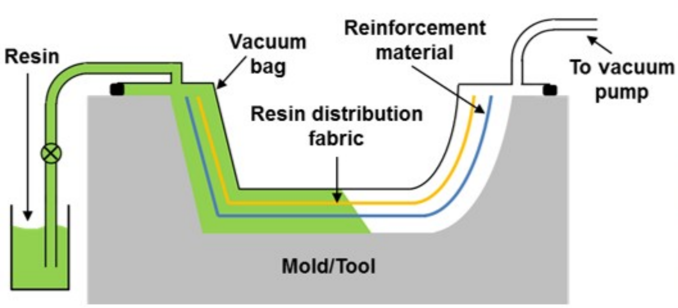


Figure 72. Manufacturing—Vacuum bagging sample

VARTM (Vacuum Assisted Resin Transfer Molding)

- **Dry fabric laid out in mold**
- **Part is vacuum bagged, resin distribution fabric can be added to improve resin flow**
- **Additional tube is added opposite vacuum pump that is placed in a resin reservoir**
- **As vacuum is applied liquid resin is drawn into part, wetting the fabric**
- **Part is allowed to cure**



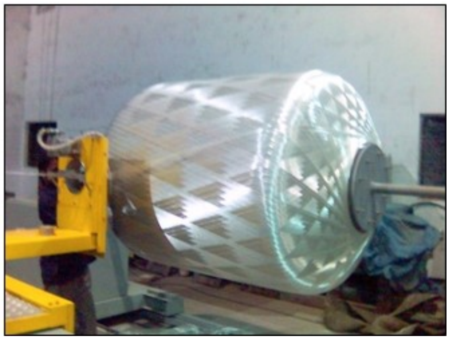
VARTM Shot - Resin Impregnating Laminate Buildup

Figure 73. Manufacturing with vacuum-assisted resin transfer molding (VARTM) process

Figure 74. Manufacturing—VARTM process sample

- **Pultrusion**
  - Continuous fibers are pulled through a resin bath and then into a heated die where the resin is cured
  - Limited to axial reinforcement and uniform cross-section

- **Filament Winding**
  - Filament tow is run though a resin bath (or prepreg is used) and then wound around a rotating mandrel
  - Commonly used to manufacture composite pressure vessels



- **Automated Tape Laying/Automated Fiber Placement (ATL/AFP)**
  - CNC head places layers of prepreg tows or tape onto a tool

- **3D Woven**
  - Woven/knitted fiber preform is created
  - Can create very complex shapes with thought thickness reinforcement
  - Infused with resin and processed through a variety of methods

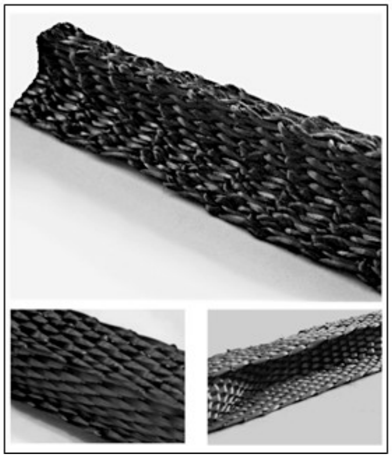


Figure 75. Manufacturing—automated methods

Figure 76. Prepreg—Automated tape laying systems

- Vacuum bagged part cured inside a heated pressure vessel
- Temperature and pressure schedule are dictated by manufacturer
- Thermocouples used to measure part temperature and control autoclave temperature
- Advantages
  - ➢ Allows more pressure to be applied to part (vacuum bagging is limited to 1 atm out of autoclave) which decreases voids and increases fiber volume fraction
  - ➢ Increased temperatures leads to better resin flow and faster cure
- Disadvantages
  - ➢ Equipment intensive (requires an autoclave large enough to fit desired parts) = expensive
  - ➢ Cooling after a high temp cure can cause warping

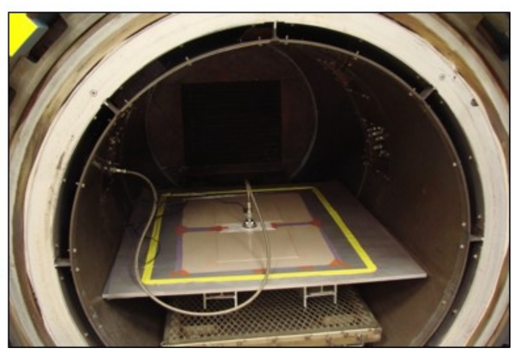
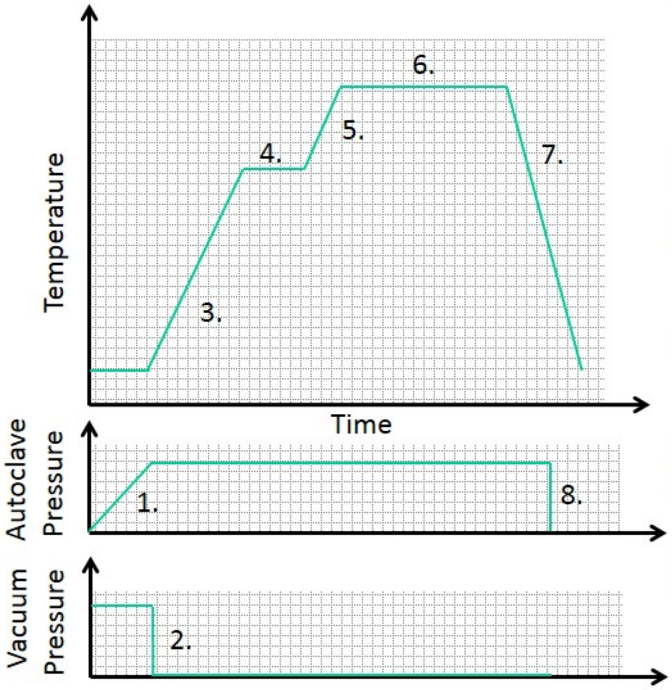


Figure 77. Manufacturing—Autoclave curing



1. Pressurization – Generally pressurized to around 100psi (50psi for sandwich structures) with an inert gas (e.g. nitrogen) to preventing combustion or charring
2. Release Vacuum – Typically done when autoclave reaches full pressure
3. Temp Ramp Up – Rate varies and is dictated by resin manufacturer but 2-10°F/min typical
4. Temp Dwell – Not always required but can help resin flow before it gels and help thick parts heat evenly
5. Temp Ramp Up – Temp to final cure
6. Temp Hold – 250 to 400°F (120 to 200°C) for 1-3 hours typical
7. Temp Cool Down – 5°F/min typical
8. Depressurization – Release pressure when temperature reaches that directed by manufacturer

Thermoplastic composites may require higher temperatures and pressures due to higher viscosity.

Figure 78. Manufacturing—Thermoset autoclave cure cycle

- **Other ways to apply heat and/or pressure which allows for proper cure, assists in resin distribution/excess elimination, control of final shape**
  - **Matched molds**
  - **Heated platen process**
  - **Industrial ovens**
  - **Heat blankets**
  - **Double vacuum bagging**

Figure 79. Manufacturing—Other curing methods

- Resins can have noxious fumes. Avoid direct contact with eyes and skin.
- Vapors can be flammability hazard.
- Resin cure is exothermic and large quantities of curing resin can be a fire hazard.
- Most fibers are non-toxic but avoid direct contact with eyes and skin. Avoid breathing dust if chopping or grinding.
- Splintering

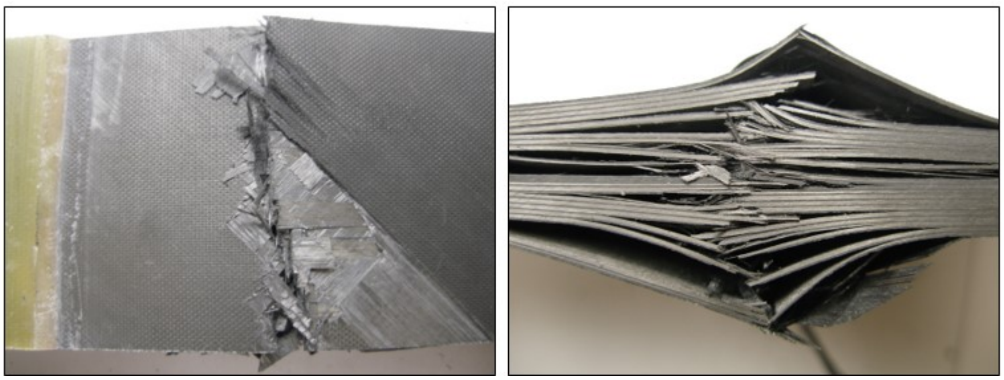
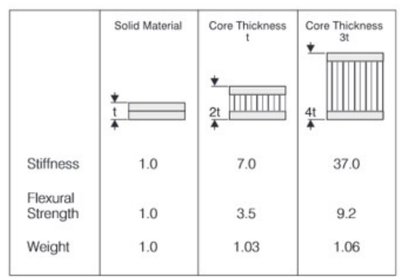


Figure 80. Composite safety



	Solid Material	Core Thickness t	Core Thickness 3t
Stiffness	1.0	7.0	37.0
Flexural Strength	1.0	3.5	9.2
Weight	1.0	1.03	1.06

- Laminate (monolithic) multi-ply structures
- Honeycomb (sandwich) structures
  - Lightweight core with high strength skins
  - Makes more efficient use of material;
  - Huge increase in stiffness and strength for small increases in weight
  - Common core materials – honeycomb (Nomex or aluminum), foam or wood
  - Multiple materials with vastly different properties and new failure modes add additional complication for NDI

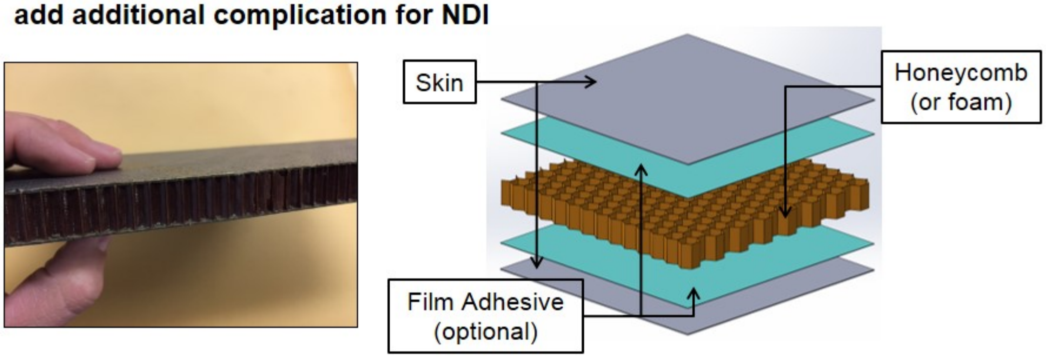


Figure 81. Common composite structures

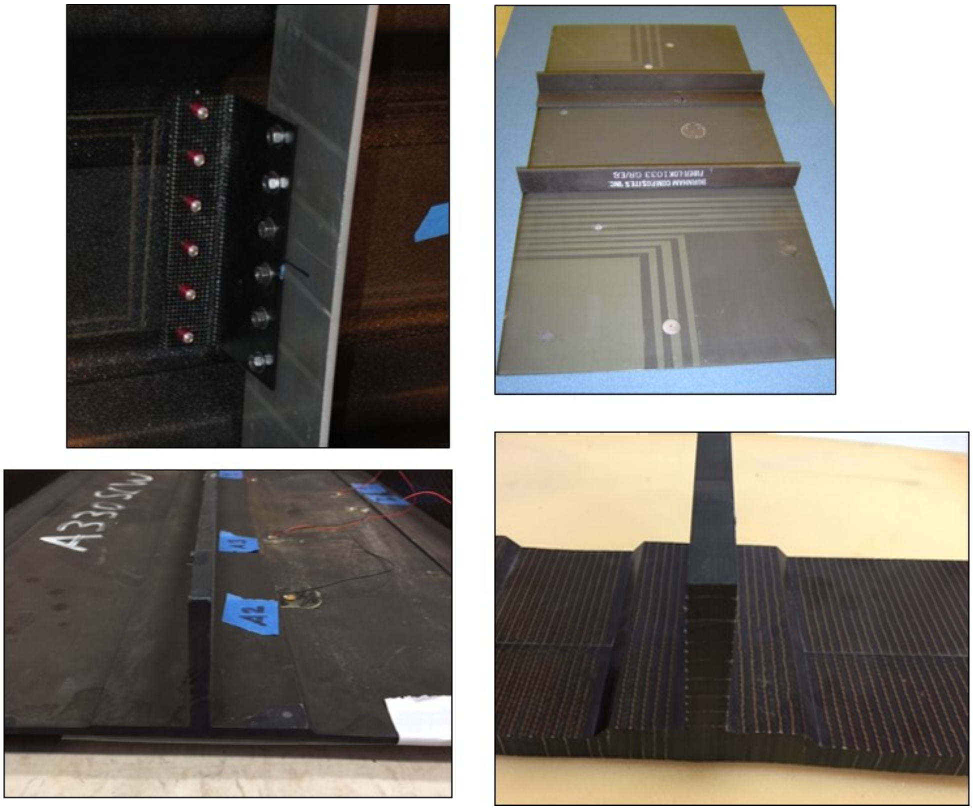


Figure 82. Common composite structures—Stiffeners

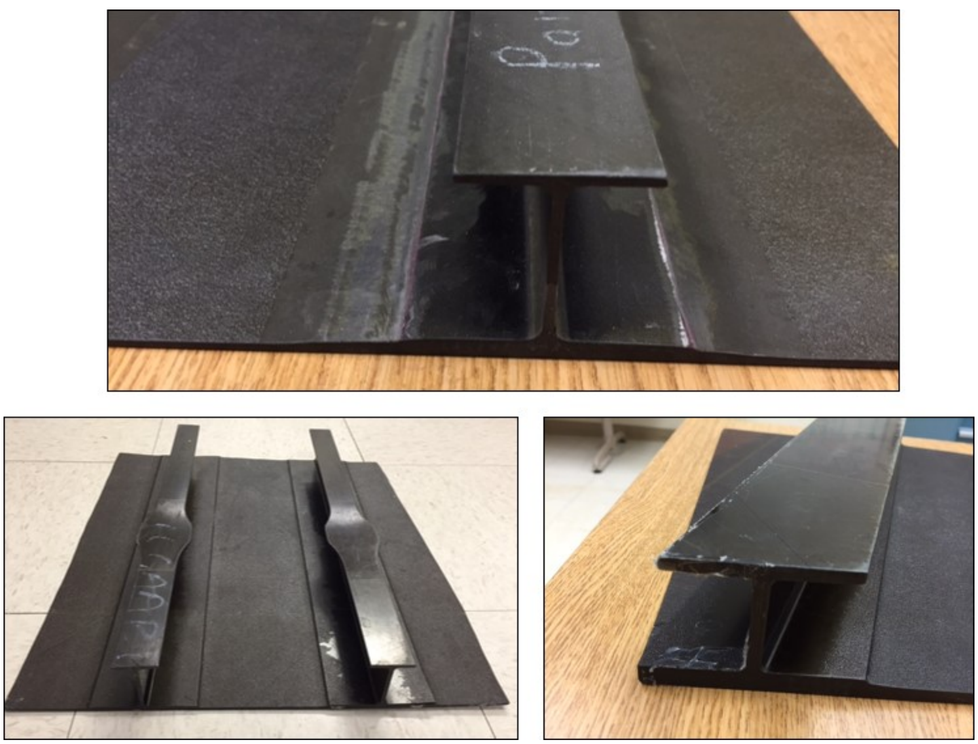


Figure 83. Common composite structures—I-beam

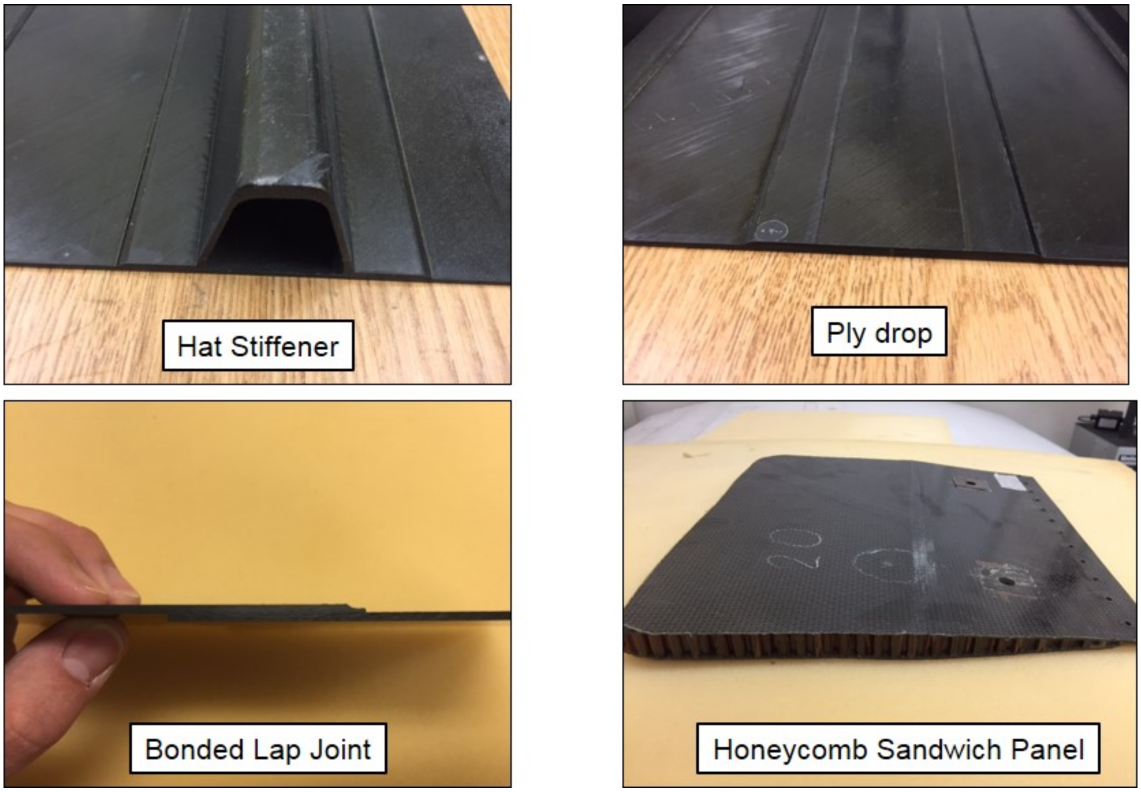


Figure 84. Common composite structures—Joints and reinforcements

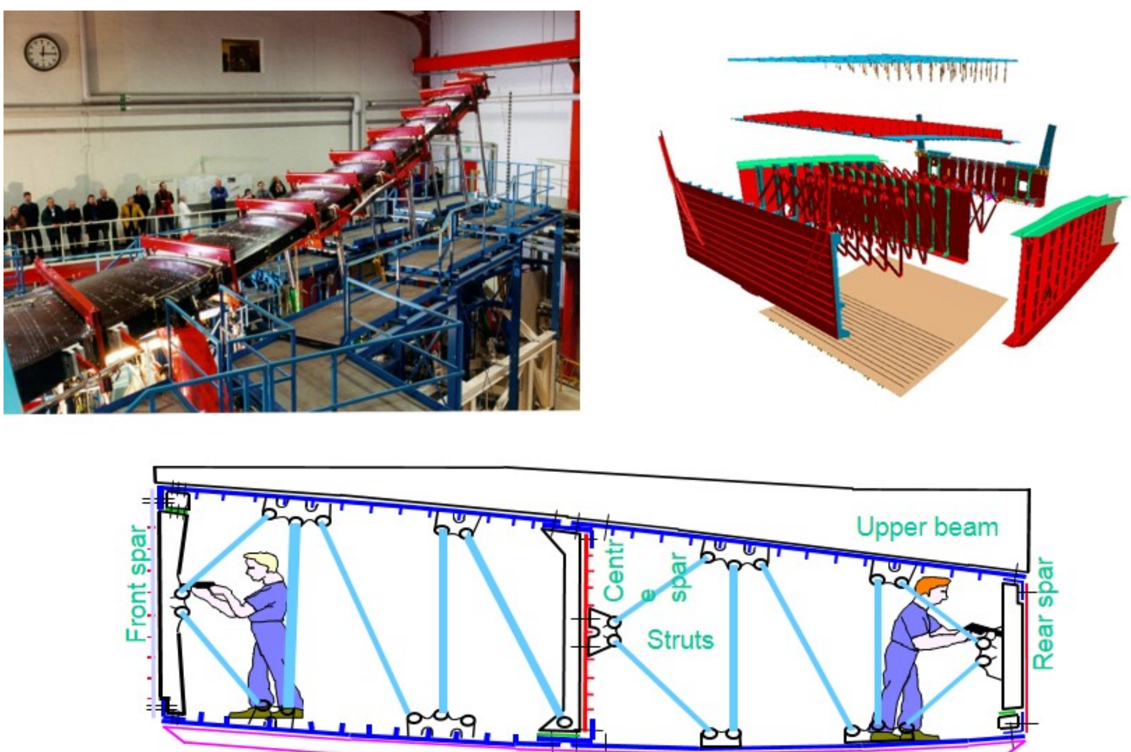
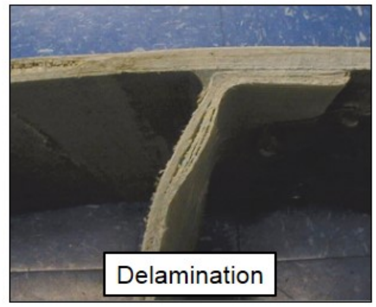
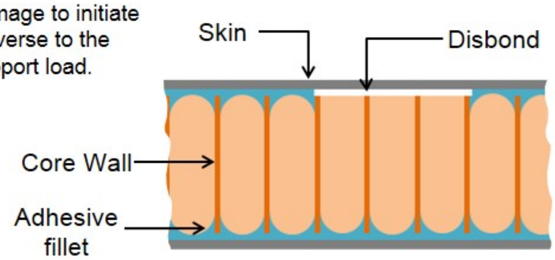


Figure 85. A-wing technology—Center wing box

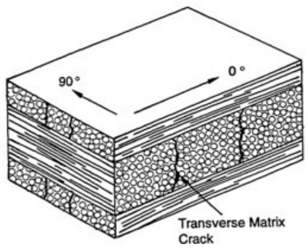
- **Composite Defect Types**
  - **Delaminations** - Region where plies aren't bonded together. Leads to buckling under compressive loads. Growth controlled by matrix toughness.
  - **Disbonds** – regions where the skins aren't bonded to the core. In composites with honeycomb cores the matrix material or an adhesive should form a fillet between skin and honeycomb.
  - **Moisture ingress** – of particular concern for honeycomb sandwich panels since they have a large amount of open volume
  - **Matrix microcracking** – Typically the first damage to initiate in composites. It occurs in plies oriented transverse to the applied load since they have less ability to support load.



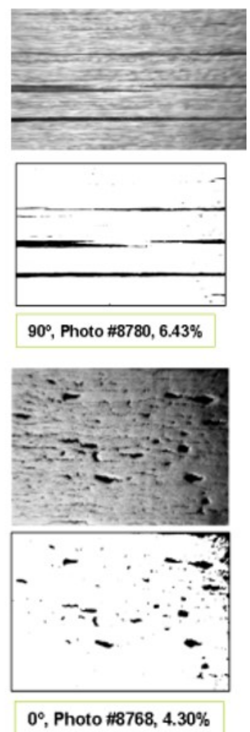


Honeycomb Core Disbond

Microcracking



- **Composite Defect Types**
  - **Porosity/voids** – Dissolved gases and absorbed moisture are common in uncured resin and will form voids if resin pressure isn't maintained above volatile vapor pressure until gel. Larger voids can occur due to poor compaction during cure.
  - **Resin rich/starved** – easiest to get when doing wet hand lay up or resin transfer/infusion methods, but can also occur due to excessive resin bleed with prepreg.
  - **Fiber waviness** – caused by excess fabric bunching, often in corners. Finished product has reduced strength in compression.
  - **FOD inclusions** – Allowing undesired material in-between composite plies. (e.g. backing material, dust, grease, mold release agent) to create a region where plies are poorly bonded together; can lead to delaminations .



Porosity

Fiber Waviness

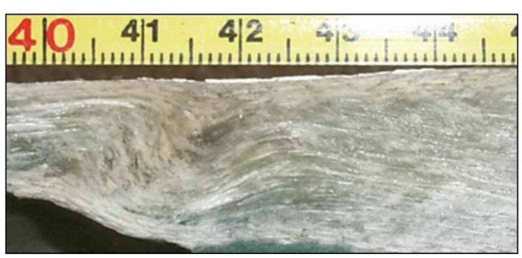


Figure 86. Composite defects

- Normal & abnormal flight loads
- Fluid contamination & ingress
- Surface coating removal; erosion
- Impact (in-flight, ground handling equipment)
  - hail, birds, lightning, runway debris, tire separation
- Heat & UV exposure
- Corrosion in metals associated with CFRP
- Maintenance errors

Inspection Challenges

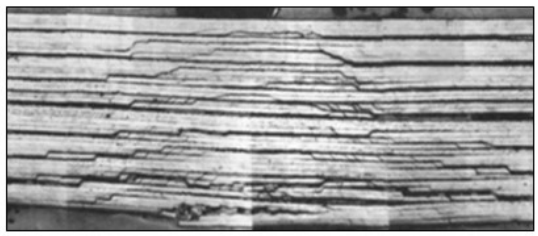
- Delaminations & disbonds
- Hidden damage
- Small amounts of moisture
- Heat damage that affects resin matrix
- Weak bonds (manuf. or environment induced)

Figure 87. Types of damage in composite structures

Causes of Damage

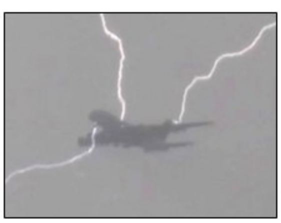
- **Impact** – of particular concern with composites due to their poor through thickness strength.
  - Barely visible impact damage (BVID) can occur where surface detection is difficult but there is significant damage (cracking and delaminations) under the surface.
- **Erosion** – Composites are less resistant to erosion in general than are metallics so they need to be properly protected.
- **Fatigue** – Again behavior is complex due to anisotropy
  - Different failure modes depending on applied stress and layup
  - Better in tension than compression (damage leads to instability in compression)
  - On axis: fiber fracture/Interfacial debonding at high stresses, matrix cracking at intermediate stresses and interface shear failure/fiber debonding at low stresses
  - Exacerbated by delaminations, propagating damage from fasteners holes, stress concentrations from holes and fabric weaves.

BVID Impact Damage



Causes of Damage

- **Corrosion** – Composites in isolation have excellent resistance to corrosion however…
  - Galvanic corrosion – conductive carbon can foster corrosion in contacting metals
  - An insulating layer (often fiberglass) can protect from galvanic corrosion
  - Titanium is a good choice when contact is required
- **Lightning Damage** – vaporizes matrix material
  - Carbon fiber is 100X less conductive than aluminum and matrix materials are one million times less
  - Aircraft require conductive mesh or foil to protect them from lightning strikes
- **Other Environmental** - polymer matrix materials (and polymer reinforcing fibers e.g. Kevlar) are particularly susceptible to environmental degradation
  - They absorb moisture which swells and softens matrix as well as attacking the fiber/matrix interface reducing strength
  - UV sensitivity – polymers are often sensitive to UV light and need protection to prevent degradation
  - Some polymers are also sensitive to certain solvents



Lightning Damage

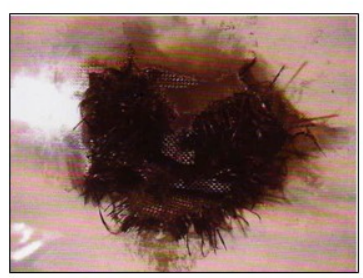


Figure 88. Composite damage descriptions

Figure 89. Categories of damage and defects to consider for primary composite aircraft structures

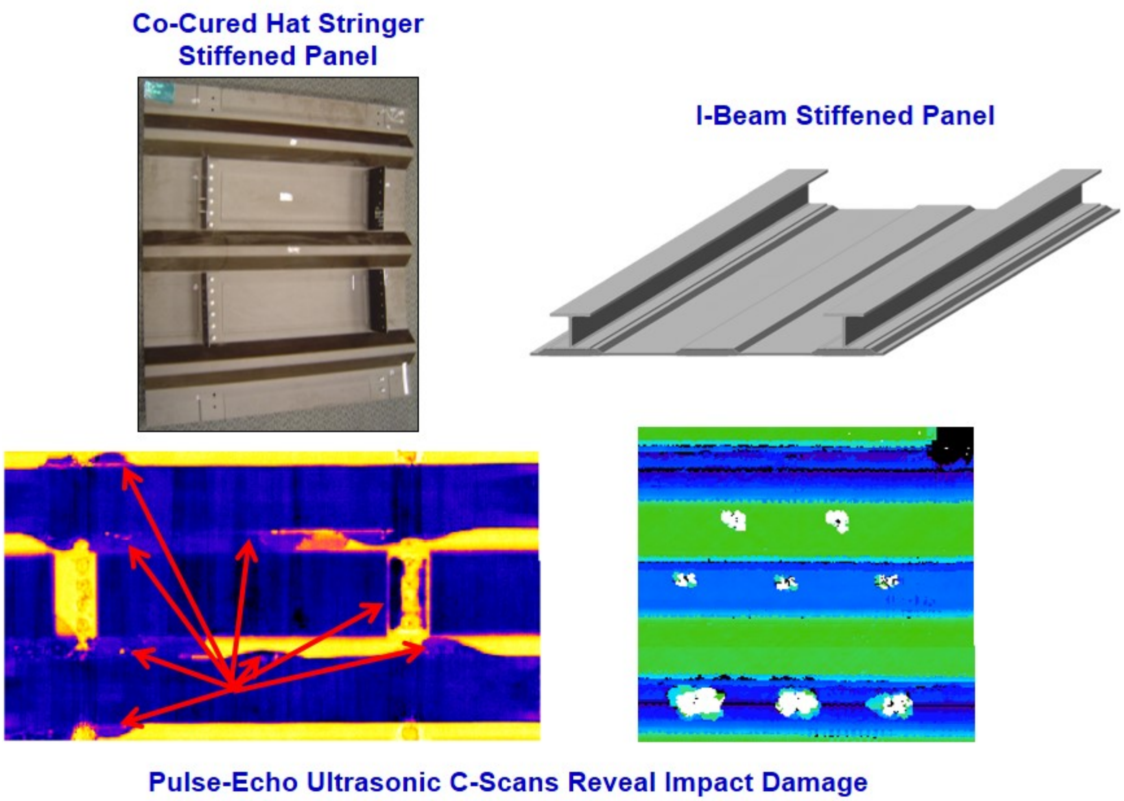


Figure 90. Inspecting for composite damage

- **Damage resistance** – ability to resist the creation of damage
- **Damage tolerance** – response associated with a damage state
- Fibrous nature of composites mean they generally have a redundant load path. The load from a single broken fiber can shift to the surrounding fibers
- Reinforcements deflect cracks slowing their growth
- High strain at failure fibers (e.g. Aramid and glass) are excellent at absorbing energy and can be added to carbon fiber composites to increase impact resistance.
- **Excellent in fatigue** – Stress concentrations from notches, holes, impact, etc. tend to form damage zones in fatigue that reduce stress concentrations resulting in smaller changes in strength when compared to static strength

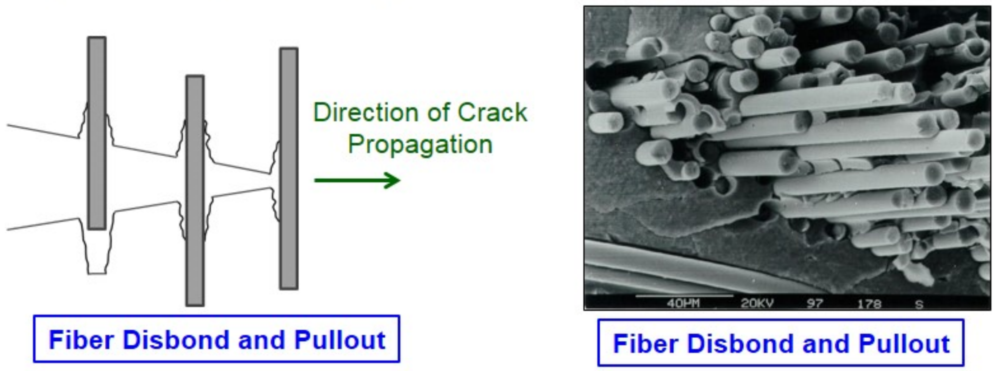


Figure 91. Composite damage resistance and tolerance

Fasteners used to mechanically join two similar or dissimilar materials together in a fashion that allows them to transmit stress.

Advantages:

- **Easy to disassemble**
- **No thickness limitations**
- **Simple joint configuration**
- **Manufacturing and inspection straightforward**
- **No surface preparation**
- **Environmentally insensitive**
- **Residual stress is generally not a problem**

Disadvantages:

- **Significant stress concentration**
- **Hole formation can damage composite**
- **Composites have poor bearing properties**
- **Added weight of mechanical fasteners**
- **Lightning tends to attach to metallic fasteners in poorly conductive composites.**
- **Less corrosion resistant**

- **Anisotropy of composites leads to failure modes that wouldn't occur with isotropic materials**
- **Composites have poor through-thickness strength (because this is a matrix dominated property and typical matrix materials aren't particularly strong)**

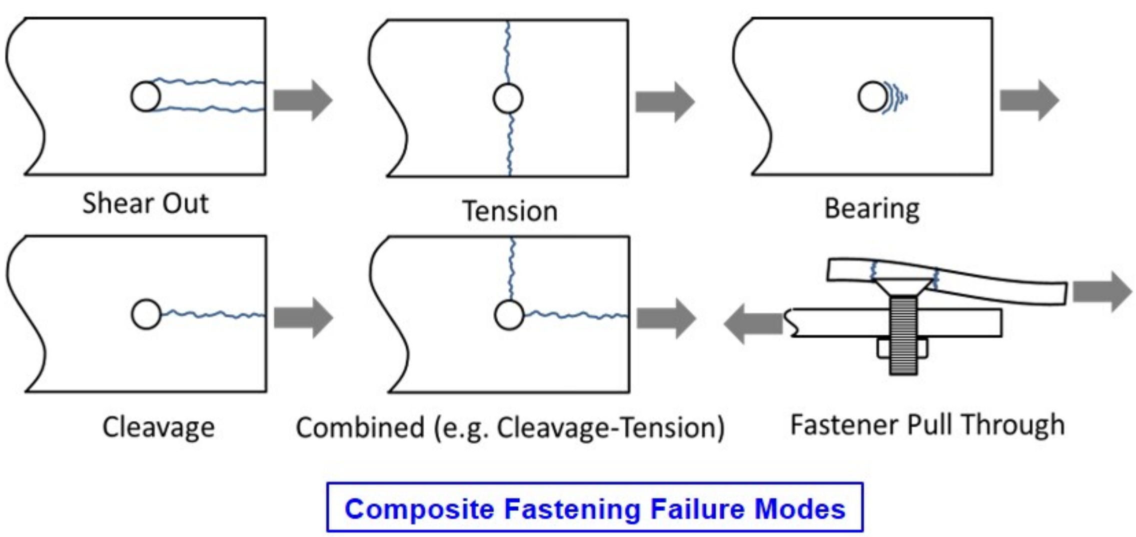


Figure 92. Mechanical fastening in composites

Adhesion is used to mechanically join two similar or dissimilar materials together in a fashion that allows them transmit stress from one part to another without undue stress concentration.

Advantages:

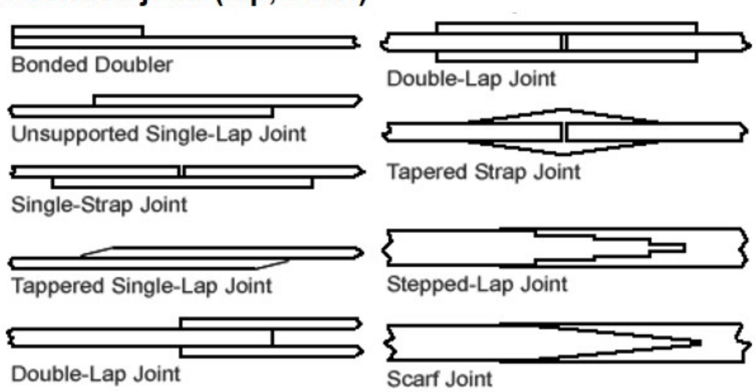
- Provides uniform stress distribution vs. a riveted connection
- Adhesive chemical composition can be tailored to provide a rigid or flexible joint connection (sound and vibration mitigation)
- Bond produces a sealing interface; is more corrosion resistant than rivets
- Correct adhesive selection can accommodate a mismatch in the CTE between the different adherends

Disadvantages:

- Heat, moisture, and micro-organisms can degrade the bond strength
- A bonded joint is hard to disassemble
- The bonded joint is not as strong in peel stresses
- Adhesive bonding require more assembly (interface cleanliness) care

Figure 93. Adhesive bonding in composites

- Joint loading condition (tension, compression, peel, shear)
- Joint geometry (e.g. lap, scarf, etc.)
- Adhesive load response desired (e.g. rigid, flexible).
- Service temperature range, exposure to solvents, fluids, vibration
- Adhesive wetting ability and surface preparation –
    - Metals are degreased, etched, anodized and primed
    - Composite surfaces require roughening and degreasing; chemical methods (e.g. silane) used to increase surface energy & improving wetting
- Type of bonded joint (lap, scarf)



Bonded Composite Structural Joints

Figure 94. Bonding joint considerations

Bonding adhesives come in both thermoset and thermoplastic (hot melt) types, just like matrix materials. Thermosets are more common.

Adhesives Forms and Processing

- **Pastes – can be loaded with fillers to tailor the mechanical properties**
- **Liquids – 2 part adhesives**
- **Adhesive tapes and films – provide a uniform, controlled bondline. Adhesive tape can be supported or unsupported (i.e have a thin cloth layer to help provide strength and control thickness)**
- **May require elevated temperatures or pressure to cure.**

Figure 95. Bonding—Adhesive and surface prep

Adhesive selection considerations: adherent materials, any CTE mismatch, joint geometry (bondline thickness), joint stresses and loading conditions, vibration spectrum, working temperature range, and environmental exposure (corrosion, moisture, oils)

- **Epoxies – most common two component (resin/curative) adhesive, can be room temp or elevated temp cure, provide good strength, have lower peel strength, are not as flexible (are brittle)**
- **Bismaleimide (BMI) – used for high temperature applications, bonding metals or composites. BMI may shrink during cure.**
- **Cyanate Ester – Excellent mechanical strength and toughness, low cure shrinkage, low moisture absorption.**
- **Hybrids – usually a modified resin system. The additives are used to improve the adhesives toughness, mechanical strength, or increase its working temperature**

Figure 96. Bonding—Types of adhesives

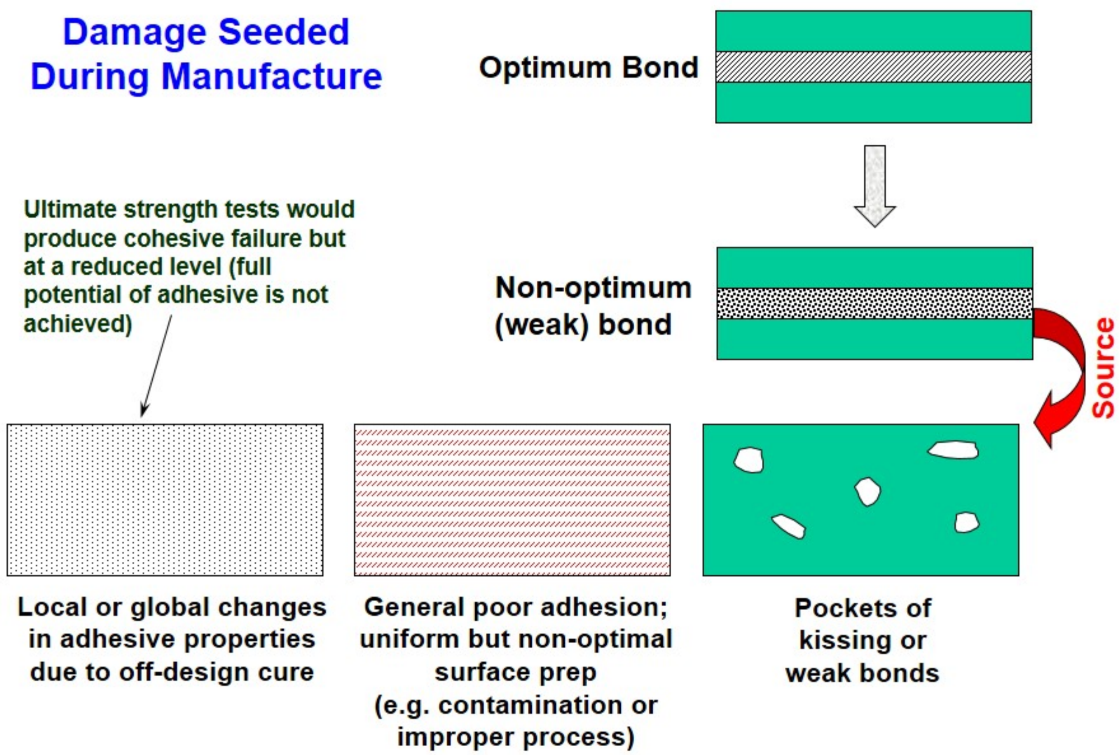


Figure 97. Mechanisms of weak bonds

- **Adhesive Failure – bonded joint fails at interface between the adhesive and the adherent. Failure due to incorrect adhesive selection for application, poor or improper surface preparation**
- **Cohesive Failure – failure of joint developing in the adhesive layer. Any failure of the adhesive below its mechanical properties may be due to an improper curing process, degradation due to moisture, contaminants, or elevated service temperatures**
- **Substrate Failure – failure of the joint in the laminate. May be due to improper stiffness matching between laminate and adhesive.**

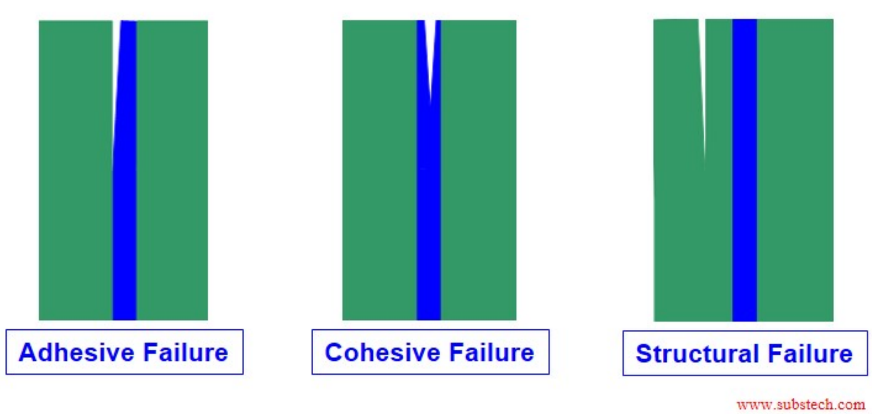
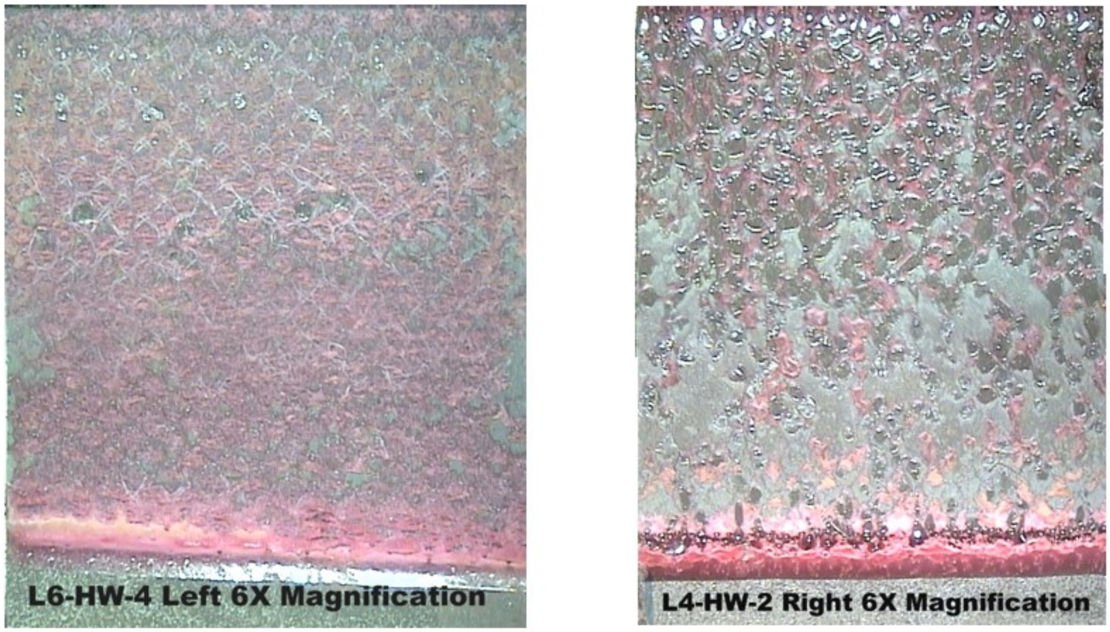


Figure 98. Bonding—bonded joint failure modes



Cohesive Fracture of Adhesive Film (Option 6 silane treatment) | **Adhesive Failure at Interface (Option 4 no chemical treatment)**

Figure 99. Adhesive vs. cohesive failure

- **Can range from filling superficial surface damage, injecting resin for internal delams to cutting out portions of laminate for replacement.**
- **General repair procedure:**
  - **Determine the extent of the damage**
  - **Remove the damaged portion**
  - **Prep area for repair (cleaning, potentially drying part, etc.)**
  - **Apply repair plies designed for the particular location**
  - **Inspect the repair**
- **General manufacturing techniques apply**
  - **Vacuum bagging to consolidate and eliminate excess resin**
  - **Ovens, heat blankets or heatlamps to increase temperature**
- **Usually bonded repairs are used but bolted repairs are possible as well.**

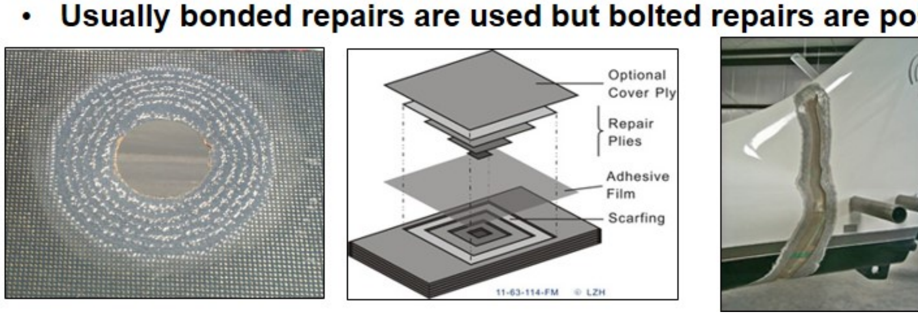


Figure 100. Repair of composites

Figure 101. Repair of composites— scarfing process

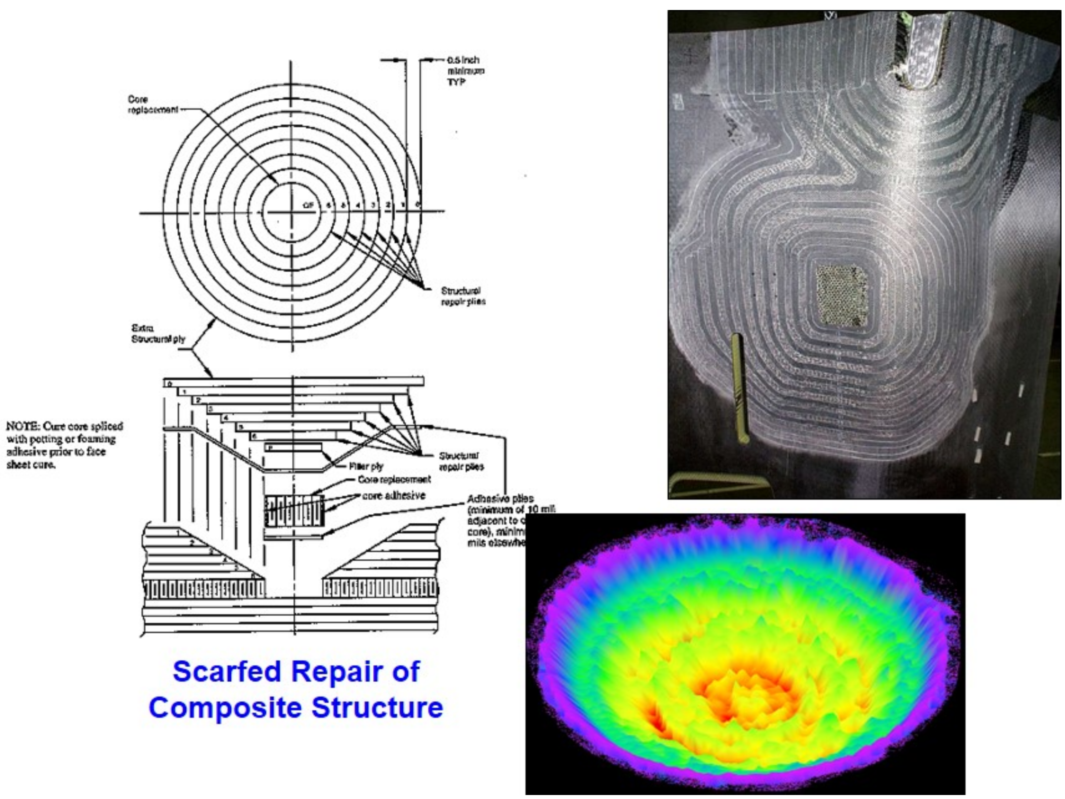


Figure 102. Sample repair of composite laminates

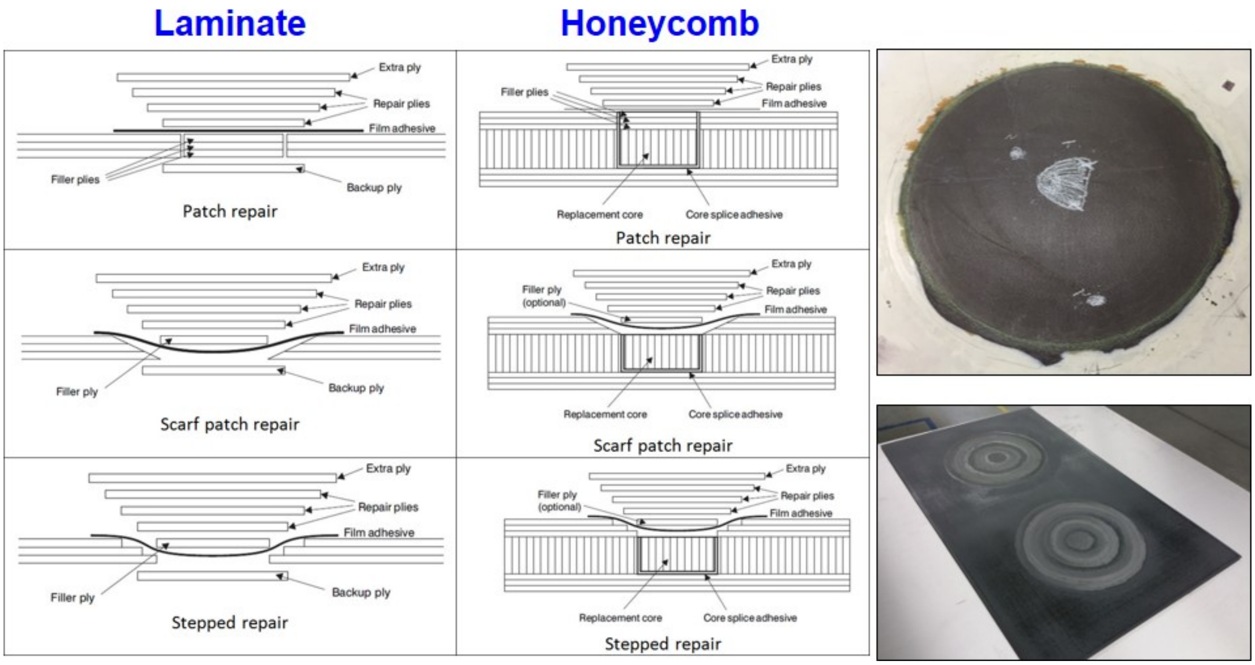
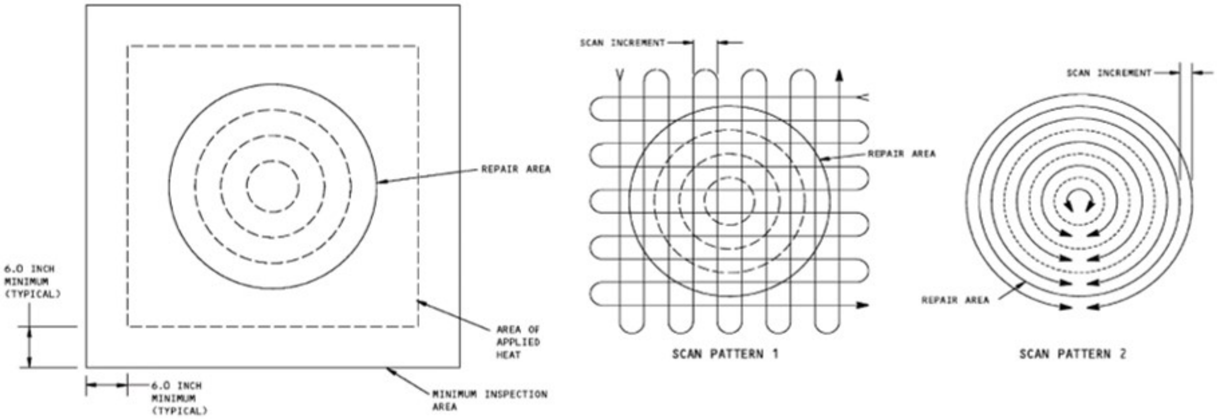


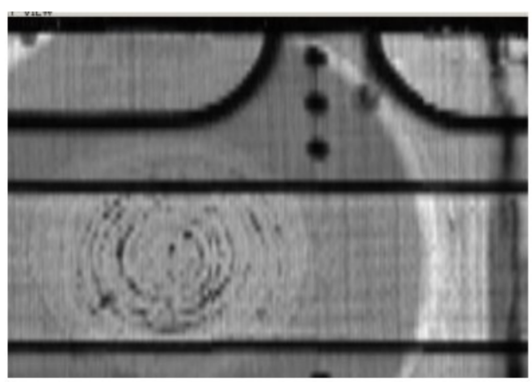
Figure 103. Types of repairs for composite honeycomb and laminates

Typical Inspection Areas- Contact UT A-Scans
Boeing 757 General Part 1

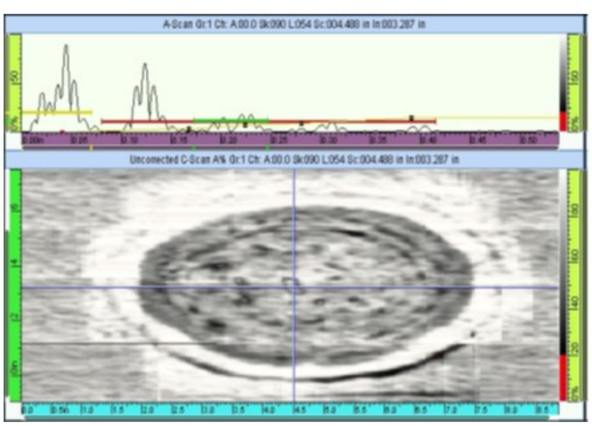


INSPECTION PROCEDURE	CALIBRATION DEFECT DIAMETERS [1]	
	METAL AND NON-METAL LAMINATES (DELAMINATION)	METAL AND NON-METAL HONEYCOMB SANDWICH (SKIN-TO-CORE DISBOND)
THROUGH-TRANSMISSION ULTRASONIC	0.50 INCH (12.7 mm)	0.50 INCH (12.7 mm)
LOW FREQUENCY BONDTESTER (NO COUPLANT)	1.00 INCH (25.4 mm)	1.00 INCH (25.4 mm)
HIGH FREQUENCY BONDTESTER (COUPLANT)	0.50 INCH (12.7 mm)	NOT RECOMMENDED
PULSE-ECHO ULTRASONIC	0.50 INCH (12.7 mm)	NOT RECOMMENDED
TAP TEST	NOT RECOMMENDED	1.00 INCH (25.4 mm) [2]

PE-UT C-scan of Scarfed Repairs



NOTE: C-Scan can be difficult to interpret. Resin rich can look like a weak bond. Plus, probe contact can give false flaws due to wobble



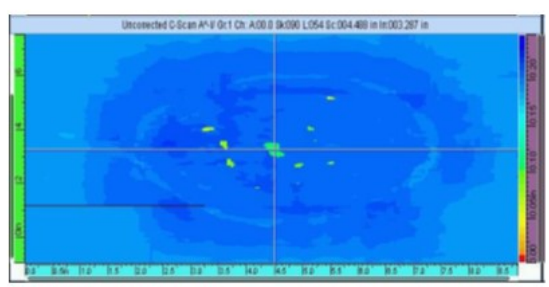


Figure 104. Repair of composites—Post repair inspection

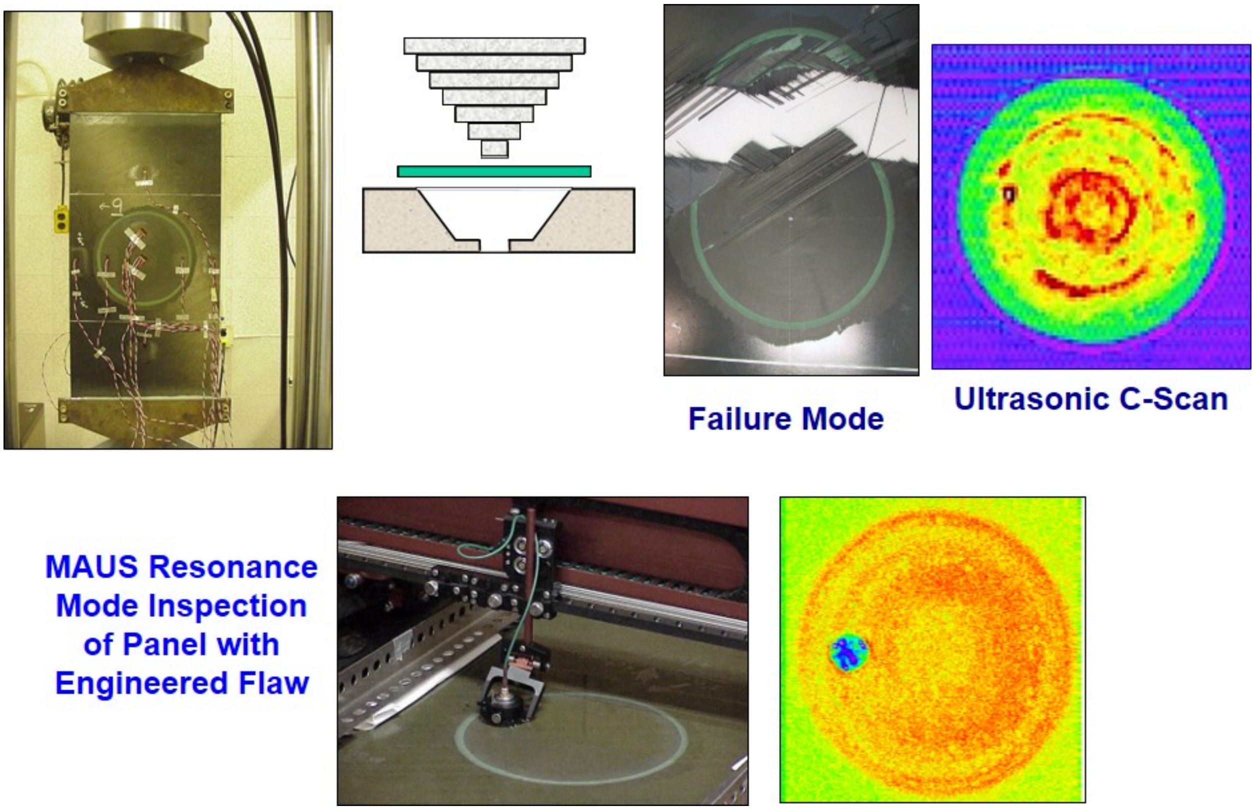


Figure 105. Composite repair—Compare mechanical and NDI performance

Figure 106. Composite NDI and laminate repairs—MIA inspection with Bondcheck V

- **Composites have many advantages for aircraft structures including their high strength/ weight ratio, resistance to fatigue damage, ability to tailor strength to needed directions, and extreme damage tolerance**
- **The aircraft industry continues to increase its use of composite materials, most noteworthy in the arena of principle structural elements.**
- **Composites exhibit more complex behavior than metals. These complexities show up in every aspect of composites: design, manufacturing, inspection and repair.**
- **This has placed greater emphasis on the application of accurate nondestructive inspection (NDI) methods for composite structures.**
- **Damage tolerance and durability is good but parts will sustain damage.**
- **Maintenance and training issues are being addressed at airlines and MROs to accommodate transition.**
- **Conventional and advanced NDI techniques are available to detect damage in composite structures and help engineers make proper decisions regarding the structural integrity and continued use of these structures**

Figure 107. Conclusions—Use of composite structures on aircraft

5. MODULE 3: COMPOSITE NDI – THEORY AND PRACTICE

Module 3 in the Composite Inspector Training Course lecture series addresses composite NDI theory and practice. Figures 108–192 encompass all of the lecture materials for "Module 3: Composite NDI Theory and Practice." This is the largest module in the Composite Inspector Training Course and is split into six sections. The sections are:

- Visual Inspection of Composites
- Basic Ultrasonic Inspection Theory
- Composite A-Scan Inspection for Damage
- Mapping Damage
- Composite C-Scan Inspection for Damage
- Solid Laminate Inspection Methods and Sample Results

The module begins with a section on visual inspection of composites. The materials cover a brief overview of types of damage composite aircraft structures are subject to and how to identify indications of damage. This section covers how to identify and define an abrasion, gouge, nick, scratch/score, and fraying. Additionally, indications of impact damage, lightning strike, stress damage, cracking, and overheating are discussed.

Module 3 continues to cover basic ultrasonic inspection theory beginning with Figure 117. The material in this section should be familiar to most inspectors, but is covered in the course for completeness and as a refresher for the students. The content begins with modes of ultrasonic vibration and basic wave theory. It continues on to cover reflection of sound waves, ultrasonic penetration, sound beam characteristics, and UT deployment.

The next section covers composite A-scan inspection for damage starting with figure 135. With the correct hardware and equipment setup, ultrasonic NDI can be used to detect delamination, fiber breakage, adhesive disbonds, matrix cracking, voids, and porosity. To properly characterize this damage, this section describes what equipment is necessary for conducting an ultrasonic A-scan on a typical composite aircraft component. It includes details regarding transducer selection, coupling methods, delay lines, and reference standards. It also covers ultrasonic equipment settings such as time-corrected gain (TCG) and full wave versus radio frequency (RF) A-scan display.

Once damage has been visually detected and the proper ultrasonic hardware and equipment setup has been achieved, the damage is inspected and mapped onto the surface of the aircraft. This process is covered in the next section, Mapping Damage, and begins with figure 145. The content provides various schematics of representative aircraft structures and describes hand-scanning best practices to accurately map composite damage. Structural configurations covered include areas of uniform thickness, co-cured stiffeners, tapered regions, and bonded stiffeners. A-scan signals from damaged and undamaged parts are also provided.

The next section covers ultrasonic C-scan inspection of composites and begins with figure 158. Although there are entire classes dedicated to phased array inspection, this section of the composite inspector training class focuses specifically on inspection of solid laminate composites using multiple element arrays and single-element scanning systems. It begins with a basic introduction to phased array UT theory and hardware setup, including transducer parameters, element patterns, aperture, and focus depth. It goes on to discuss wedges, scanners, wheel encoders, and gate settings. Next, amplitude, time of flight, and sectional B-scans are presented and compared.

Module 3 then describes various composite laminate inspection methods applied to solid laminates and provides example inspection results starting with figure 177. This section is intended to expose inspectors to new inspection methods. Many of the examples use ultrasonics.

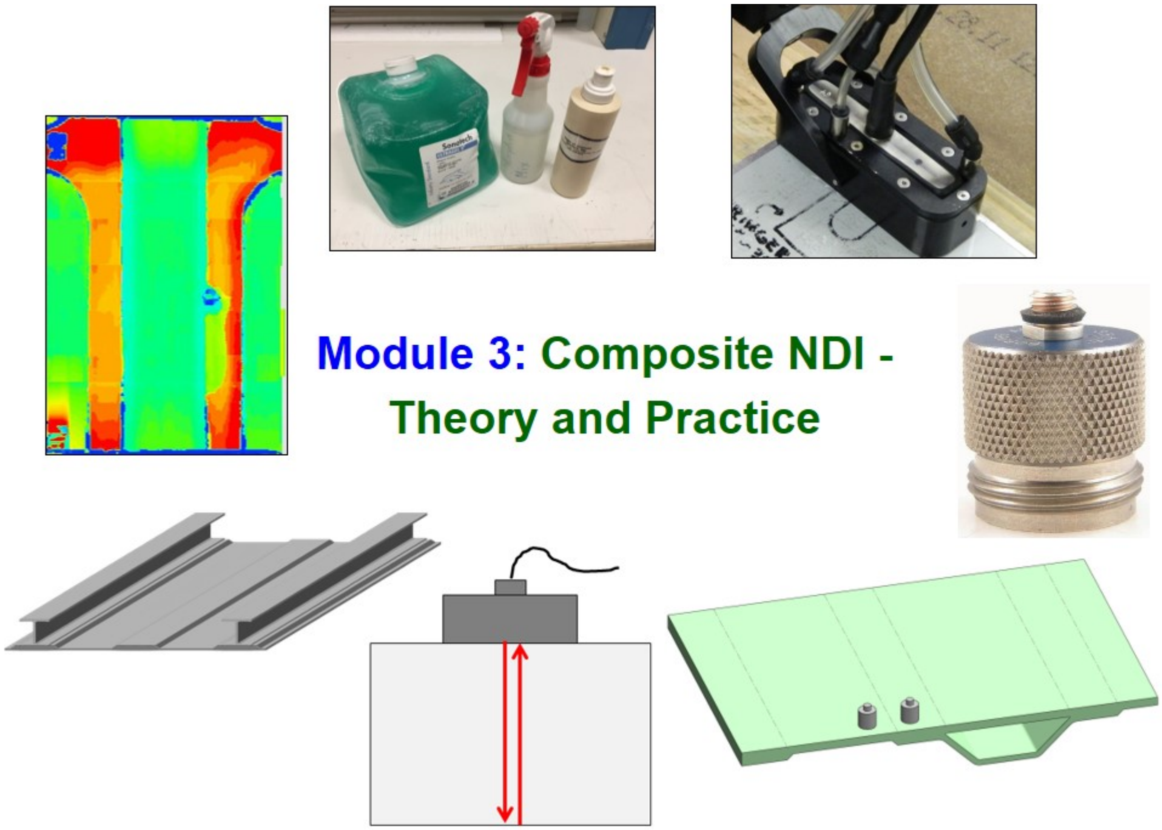


Figure 108. Composite laminate NDI training class

- Visual inspection of composites
- Basic ultrasonic inspection theory
- Ultrasonic deployment and options
- Ultrasonic equipment set up
- Mapping damage
- Ultrasonic signals from normal and damaged structure
- Phased array inspection
  - C-scan generation
- Solid laminate inspection methods and sample results

External References

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NAVAIR 01-1A-16
TM 1-1500-335-23
TECHNICAL MANUAL
NONDESTRUCTIVE INSPECTION
METHODS, BASIC THEORY

BOEING
787
NONDESTRUCTIVE TEST MANUAL
PART 4 - ULTRASONIC

A318/A319/A320/A321
NONDESTRUCTIVE TESTING MANUAL
CHAPTER 51 - STANDARDS PRACTICES AND STRUCTURES

Figure 109. Module 3 overview

Module 3a: Composite NDI - Theory and Practice

Visual Inspection of Composites

Figure 110. Introduction to visual inspection of composites

- Visual inspection is a quick and powerful tool for detecting damage in composite structures
- Impacts often leave surface visible indications of subsurface damage
- A slight wave, ripple , or dent on the surface may indicate a subsurface disbond or delamination

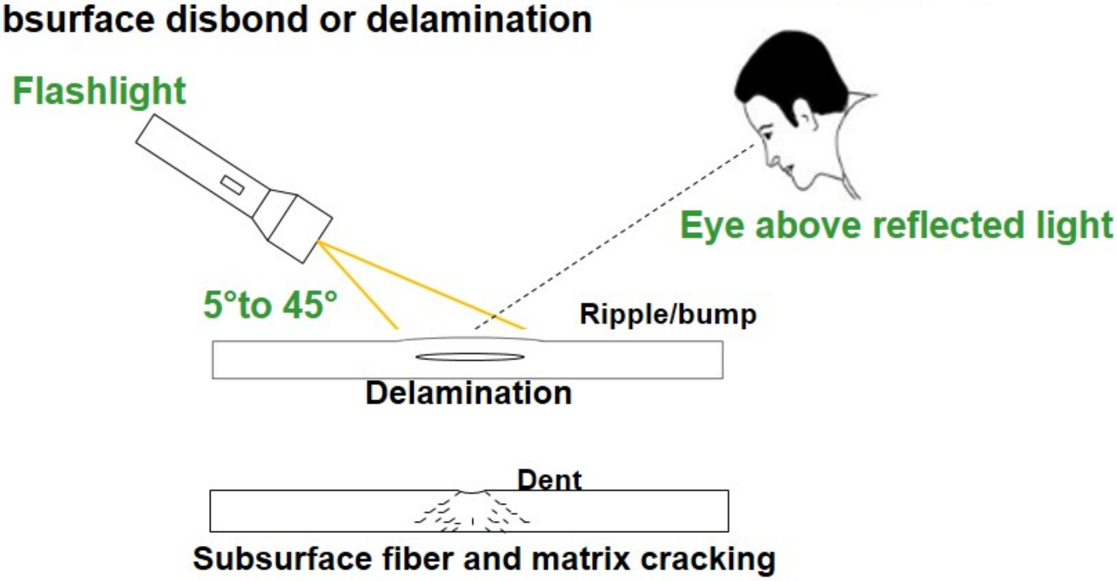


Figure 111. Visual indications of damage

While conducting a visual inspection, lighting, viewing angle and distance contribute to detection.

Same carbon laminate panel with impact damage viewed in two different lighting scenarios.

Figure 112. Example of visual indication of damage

- **Make sure that the inspection area is thoroughly clean, remove any loose or flaking paint**
- **Use adequate lighting and inspect the suspect area with a magnifying glass if necessary**

Abrasion – A damage area of any size which causes a change in cross-sectional area because of scuffing, rubbing, scraping, or other surface erosion. The geometry is usually rough and irregular.
Gouge – A damage area of any size which results in a cross-sectional area change. Gouges are usually caused by contact with a relatively sharp object which produces continuous, sharp or smooth channel-like groves in the material.
Nick – A local removal of material due to a knock/impact at the edge of a member or skin.
Scratch/Score – A line of damage caused by a sharp object at any depth and length in the material which causes a cross-sectional area change.
Fraying – Broken or loose fibers on edges.

Figure 113. Visual indications of damage—Terms

Tools that may aid in visual inspection include: magnifying glass, extending mirrors, a flashlight, and a borescope.

Visual indications of impact damage include:
1) Cracked or chipped paint
2) Dents on the surface of the structure
3) Cracked or fractured plies, missing edge plies

Visual indications of lightning strike include:
1) Blistered, scorched, or chipped paint
2) Frayed fibers
3) Some missing plies
4) Delamination
5) Complete removal of the plies
6) Signs of stress around the fasteners


Dent and surface cracking

Indications of stress damage include:
1) Fastener hole damage such as:
a) Chipped, loose, or lifted paint
b) Fasteners that have been pulled through the skin.

Indications of possible crack damage includes:
1) Linear cracks in the paint
2) Structural movement or separation

Indications of burning or overheating include:
1) Blistered and/or discolored paint


Lightning Damage


Foreign Object Damage

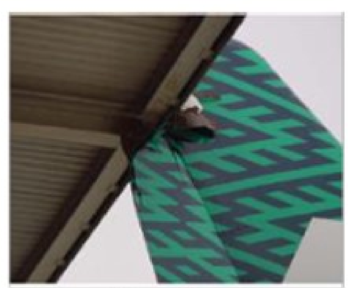

Ground Handling Damage

Figure 114. Visual indications of different types of composite damage

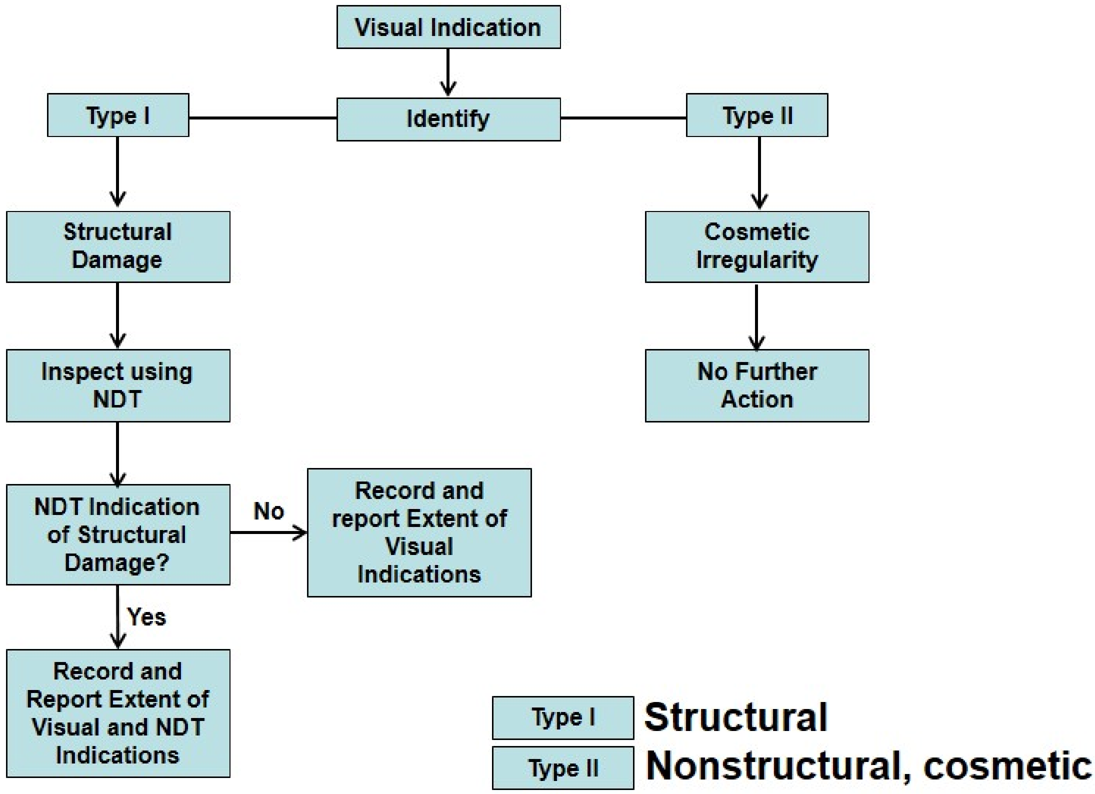


Figure 115. Simplified inspection flowchart

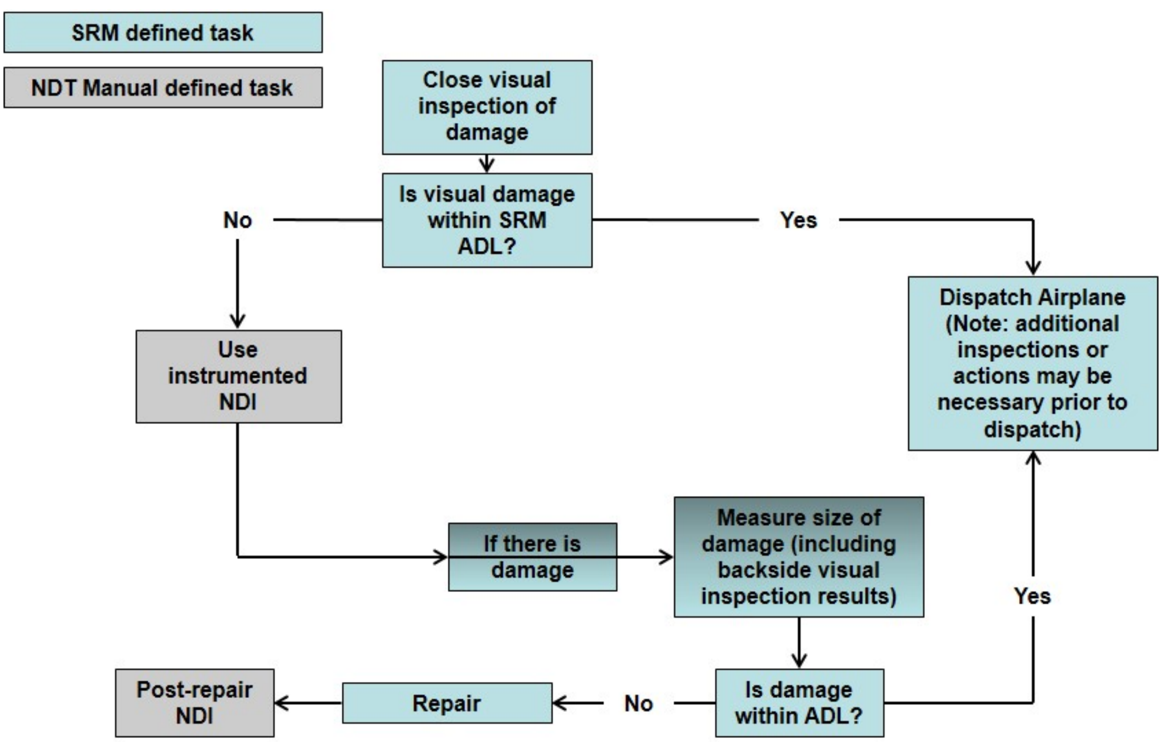


Figure 116. Inspection and repair flowchart

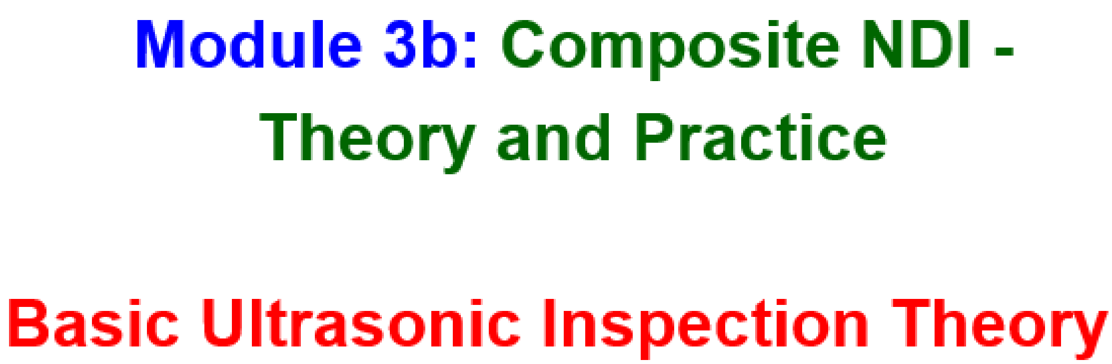
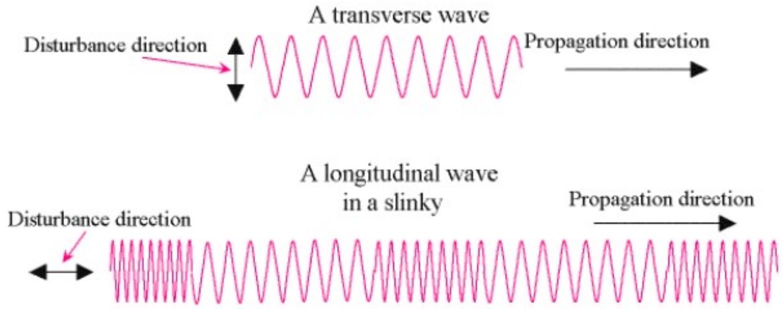


Figure 117. Basic introduction to composite ultrasonic inspection theory

Sound propagates through a part via particle vibrations. As a stress is applied, particles move away from their resting position. This occurs continually until the source of vibration is removed.



- **Longitudinal wave – Straight beam; particle movement is parallel to the direction of wave travel**
- **Transverse – Angle beam or shear wave; particle movement is perpendicular to the direction of wave travel**
  - **travels at about ½ the speed of longitudinal waves**

Longitudinal Wave - also known as compression waves or L-waves. The wave motion is in the same direction (parallel) as sound wave propagation. This type of wave can be generated in solids, liquids, and gases.
Shear Wave - also known as transverse wave, S-Wave, or angle beam. Shear waves travel perpendicular to the direction of propagation. The velocity of a shear wave is about ½ that of a longitudinal wave.
Surface Wave - Surface Waves travel across the surface with a penetration ~1 wavelength deep, with detectability of flaws lying ½ the wavelength to the surface. As such, surface waves are most desirable for detecting defects close to the surface of the part. The energy of this wave mode decays rapidly below the surface of the part.

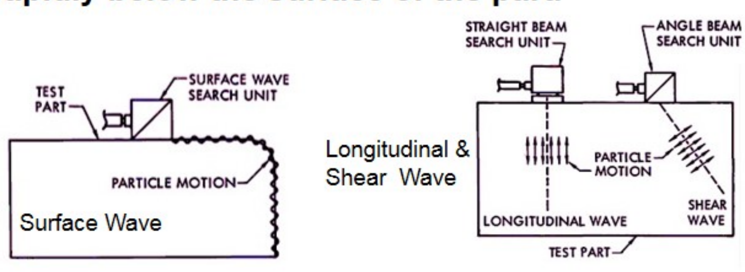
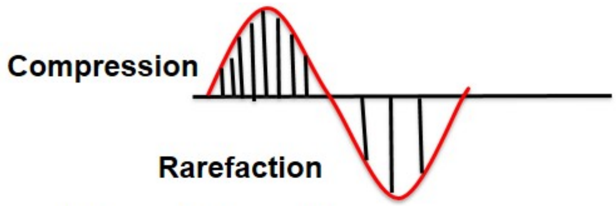


Figure 118. UT theory—Modes of vibration



- **Cycle** - One complete particle motion
- **Wavelength** - The distance a wave travels while going through one cycle. Wavelength consists of two sides, compression side and a rarefaction side
- **Frequency** - The number of complete cycles of vibration in one second (or other specified time period)
- **Velocity** - the distance a wave travels in a given material per unit of time (second)

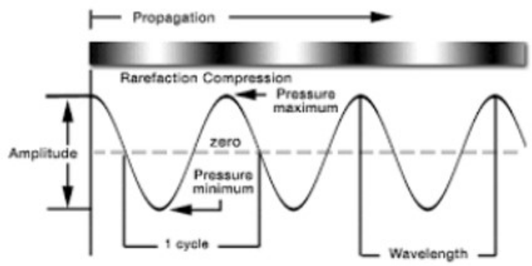


Figure 119. UT theory—Wave theory definitions

Frequency Ranges:

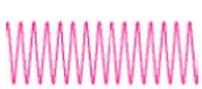
- **Subsonic: <20Hz**
- **Sonic: 20 to 20,000 Hz**
- **Ultrasonic: >20,000 Hz** → **Ultrasonic Testing**

200,000 Hz to 25,000,000 Hz

0.2 MHz to 25 MHz

Low Frequency has long wavelength

- **Thick solid laminates, large fiber diameter materials, larger defects**

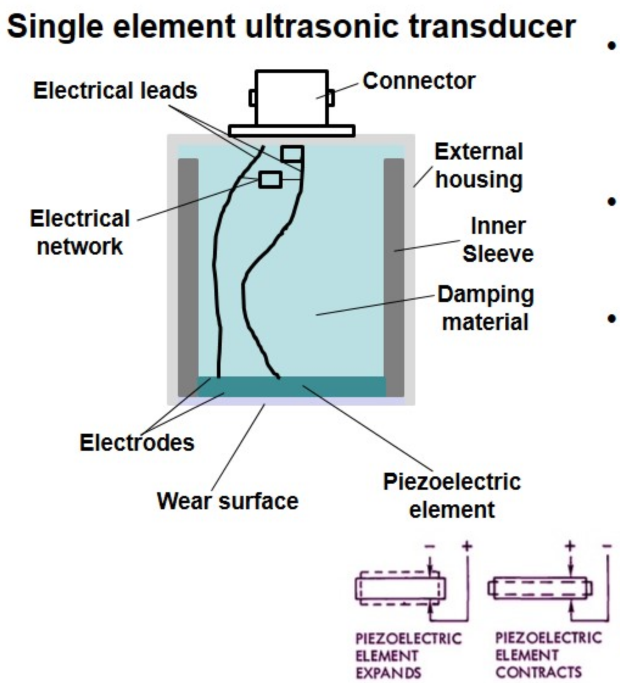


High Frequency has short wavelength

- **Thinner laminates, small fiber diameter materials, small defects**

Question: Low frequency has [greater] depth of penetration than high frequency ultrasonics?

Figure 120. UT theory—Frequency



- Ultrasonic vibrations are generated by applying electrical energy to a piezoelectric element within a transducer
- The system converts electrical energy to mechanical energy and then back to electrical energy
- Frequency is determined by the transducer element. The thicker the element, the lower the frequency

Element expands and contracts when a voltage is applied

Figure 121. UT theory—Wave generation

The Wave Equation

$$Wavelength = \frac{Velocity}{Frequency}$$

Rule of Thumb – Minimum detectable flaw size = ½ the wavelength

Example: Determine minimum detectable flaw size.

- 3.5 MHz longitudinal wave transducer
- Carbon laminate inspection: Velocity = 0.114 in/μs

$$W = \frac{V}{F}$$

$$Wavelength = \frac{.114x10^6\ (\frac{in}{sec})}{3.5x10^6\ (Hz)} = .033 \text{ in} \qquad Flaw_{min} = \frac{.033\ in}{2} = .016 \text{ in}$$

Example: Determine probe frequency.

- Intended flaw size: .05 in
- Fiberglass inspection: Velocity = 0.108 in/μs

$$F = \frac{V}{W}$$

$$Wavelength = .035\ x2 = .07 \text{ in} \qquad Frequency = \frac{.108x10^6\ (\frac{in}{sec})}{.07(in)} = 1.54 \text{ MHz}$$

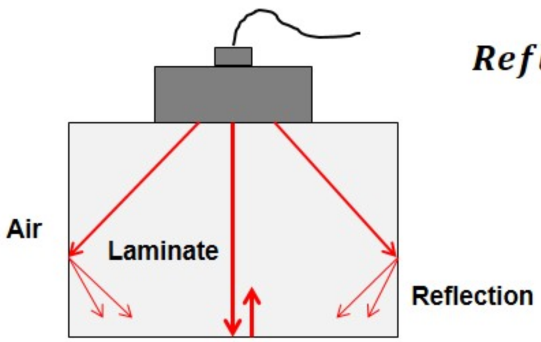
Figure 122. UT theory—Wave theory example

Sound reflects off of interfaces of dissimilar materials

- **The amount of sound (acoustic energy) transmitted depends on the <u>angle of incidence</u>, and the <u>acoustic impedance mismatch</u> between the two materials**

- **<u>Acoustic impedance</u> - determined by the density and the sound velocity of the material. It is the resistance of the material to transmit the propagation of sound waves**

  Acoustic Impedance (Z) = Density x Velocity

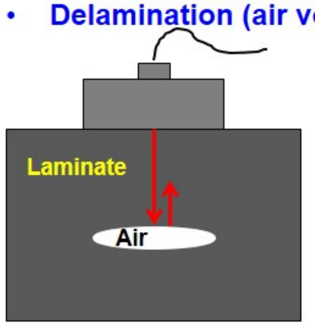


$$\boldsymbol{Reflected\ Energy\ \% = \frac{(Z_2 - Z_1)^2}{(Z_2 + Z_1)^2}}$$

Figure 123. UT theory—Reflection of sound waves

Example: Determine the reflected energy.

- **Longitudinal wave transducer**
- **Carbon laminate inspection: D = .057 lb/in^3, V = 0.114 in/μs**
- **Delamination (air void): D = 4.43x10^{-5} lb/in^3, V = 0.014 in/μs**



Z_1 = 4.65

Z_2 = .00041

$$Reflected\ Energy\ \% = \frac{(Z_2 - Z_1)^2}{(Z_2 + Z_1)^2}$$

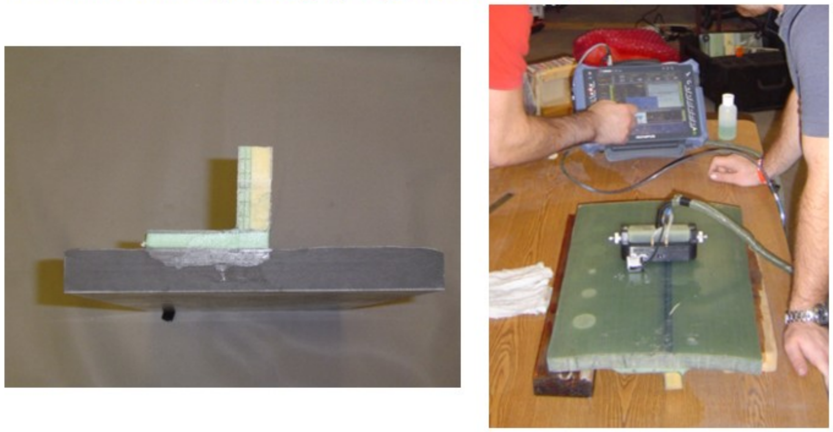
$$RE\ \% = \frac{(.00041 - 4.65)^2}{(.00041 + 4.65)^2} = .999$$

Multiplying the reflection coefficient by 100 yields the amount of energy reflected as a percentage of the original energy.

Figure 124. UT theory—Reflection of sound waves example

Ultrasonic Penetration

- **The ability to overcome ultrasonic attenuation**
- **Penetration depends on the amount of sound produced and the frequency of transducer**
- **To increase penetration:**
  - **Increase transducer diameter**
  - **Decrease frequency (larger wavelength)**



Thick composite wind blade parts

Figure 125. UT theory—Ultrasonic penetration

Snell's Law

- **Describes the relationship between the angle of incidence and refraction**
- **Various angles can be generated within the part to meet inspection needs by changing the angle of wedge**

$Sin\ \theta 1$ = Angle of incident wedge cut

$Sin\ \theta 2$ = Sound Beam Angle Produced In Part

$V1$ = Velocity of the Wedge

$V2$ = Velocity of the Part

Snell's Law

$$\frac{Sin\ \theta 1}{Sin\ \theta 2} = \frac{V1}{V2}$$

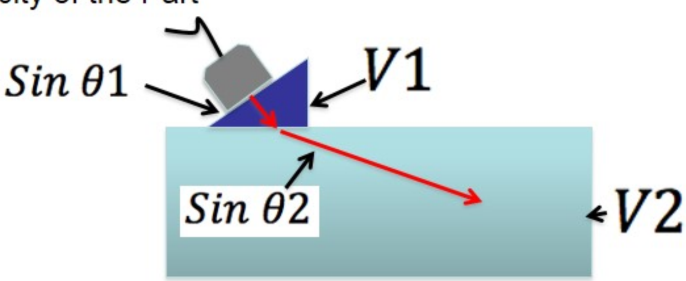
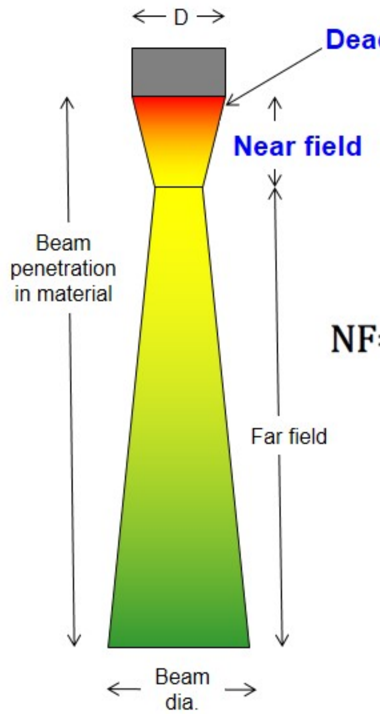


Figure 126. UT theory—Refraction and mode conversion



$$NF=\frac{D^2}{4\lambda}$$

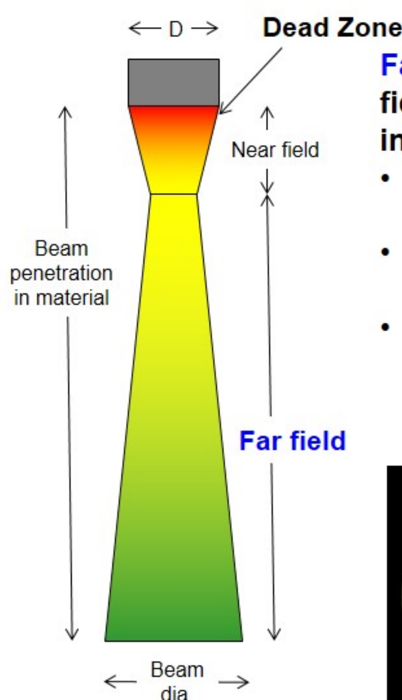
Dead Zone - The area directly beneath the transducer.

- Considered a non-inspection area
- Limits ability to detect near surface flaws
- Can be mitigated with a delay line

Near Field (Fresnel Zone) - The zone within the sound path that has wide variations of intensities, not recommended as an inspection zone.

- Caused by interfering sound waves
- Can change the length of the near field by changing the size of the transducer diameter or transducer frequency
- If you decrease the transducer diameter or decrease the frequency, you will decrease the length of the near field

Figure 127. UT theory—Sound beam characteristics



Far Field (Fraunhofer Zone) - The zone beyond the near field absent of interference effects. Many ultrasonic inspections are designed to occur in the far field

- As the sound extends past the near field, the intensity of the beam falls off
- The beam divergence angle can be calculated using the below formula
- This field is also adjusted by changing the diameter or frequency of the transducer

$$\sin\theta = 1.2\frac{V}{DF}$$

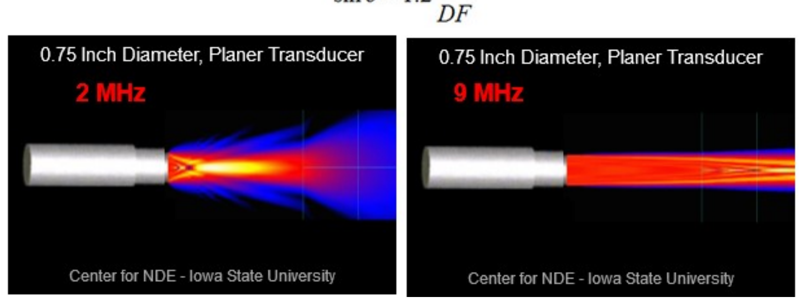


Figure 128. UT theory—Sound beam characteristics

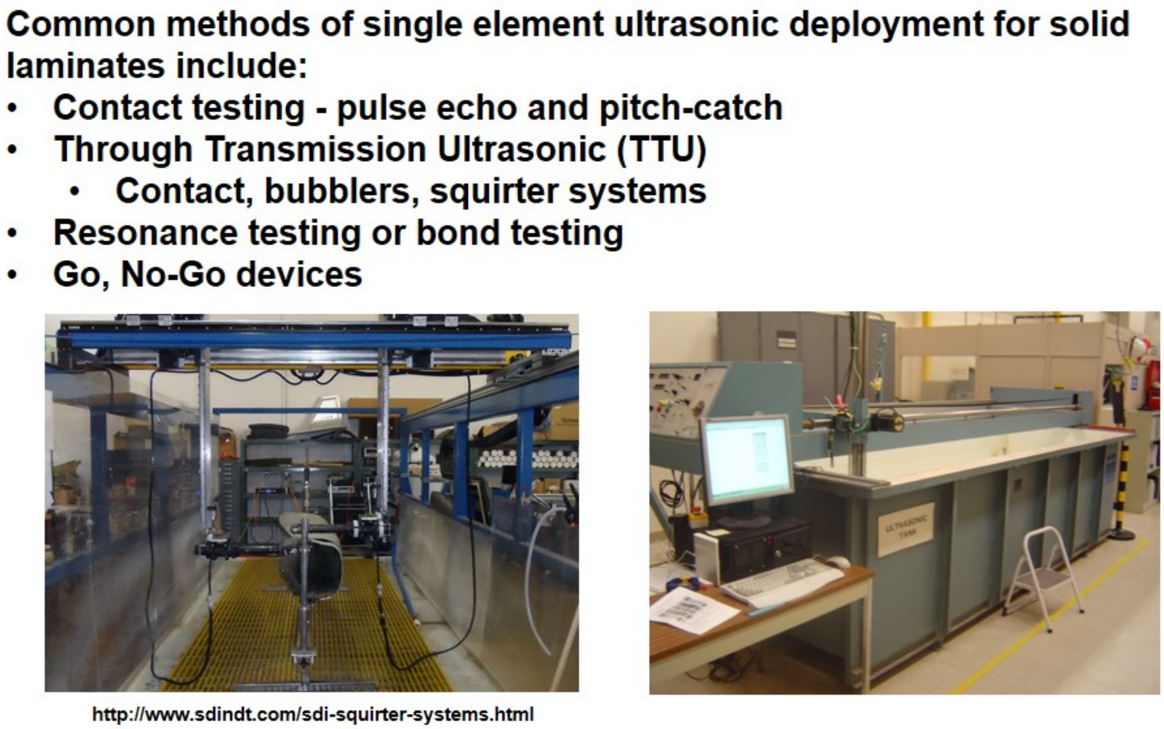


Figure 129. UT deployment

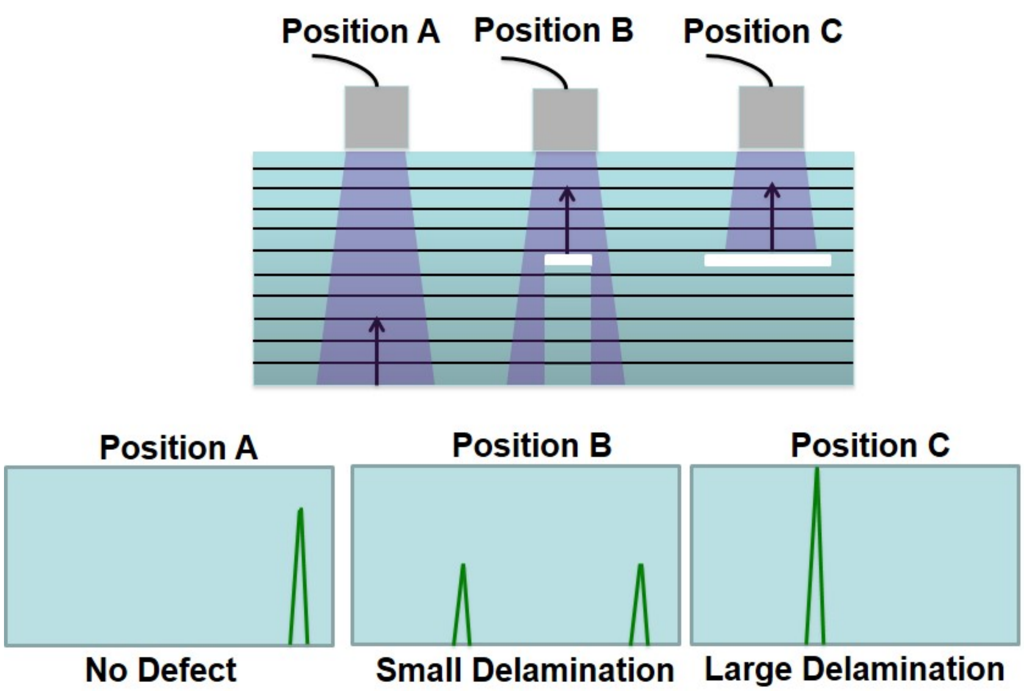


Figure 130. Ultrasonic deployment—Pulse-echo contact testing

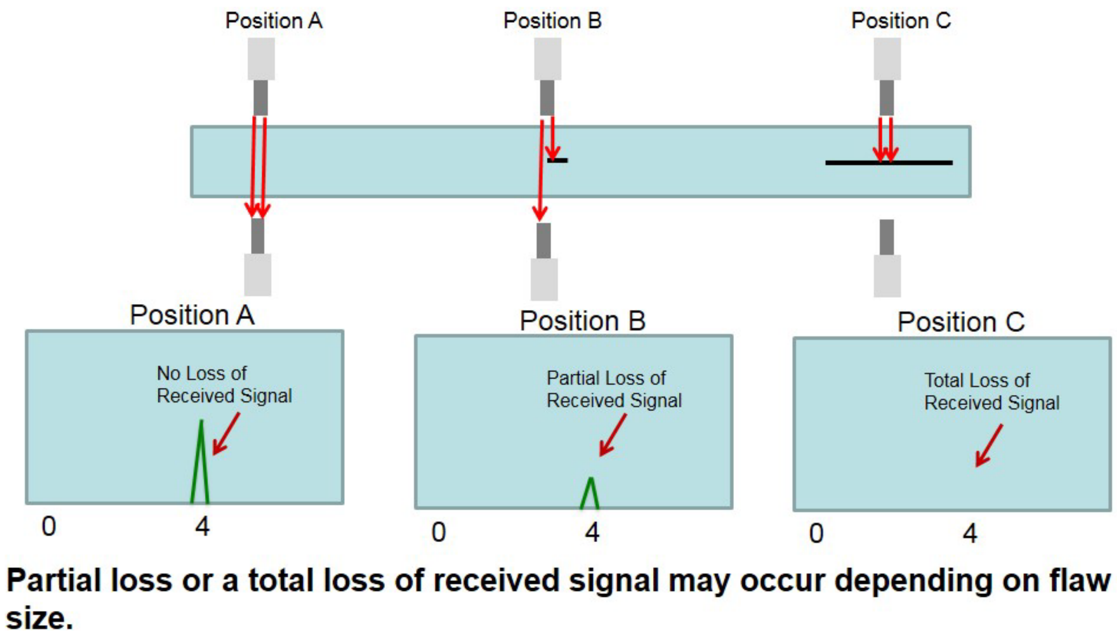


Figure 131. Ultrasonic deployment—through-transmission ultrasonic (TTU) immersion testing

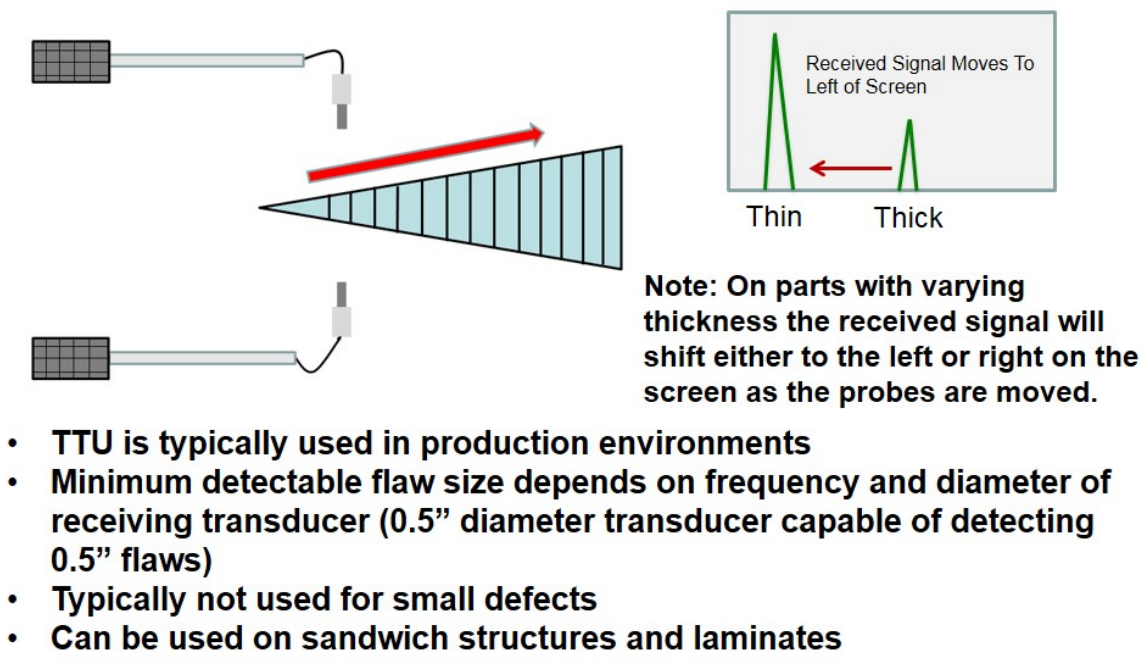


Figure 132. Ultrasonic deployment—TTU immersion testing angled parts

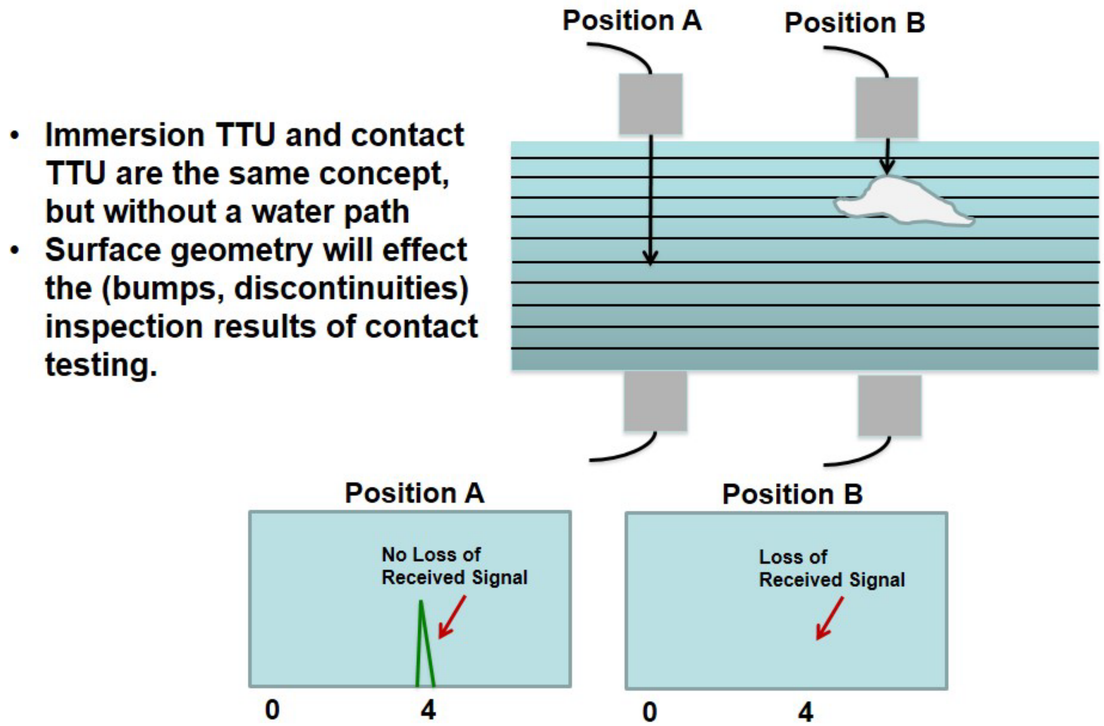


Figure 133. Ultrasonic deployment—TTU contact testing

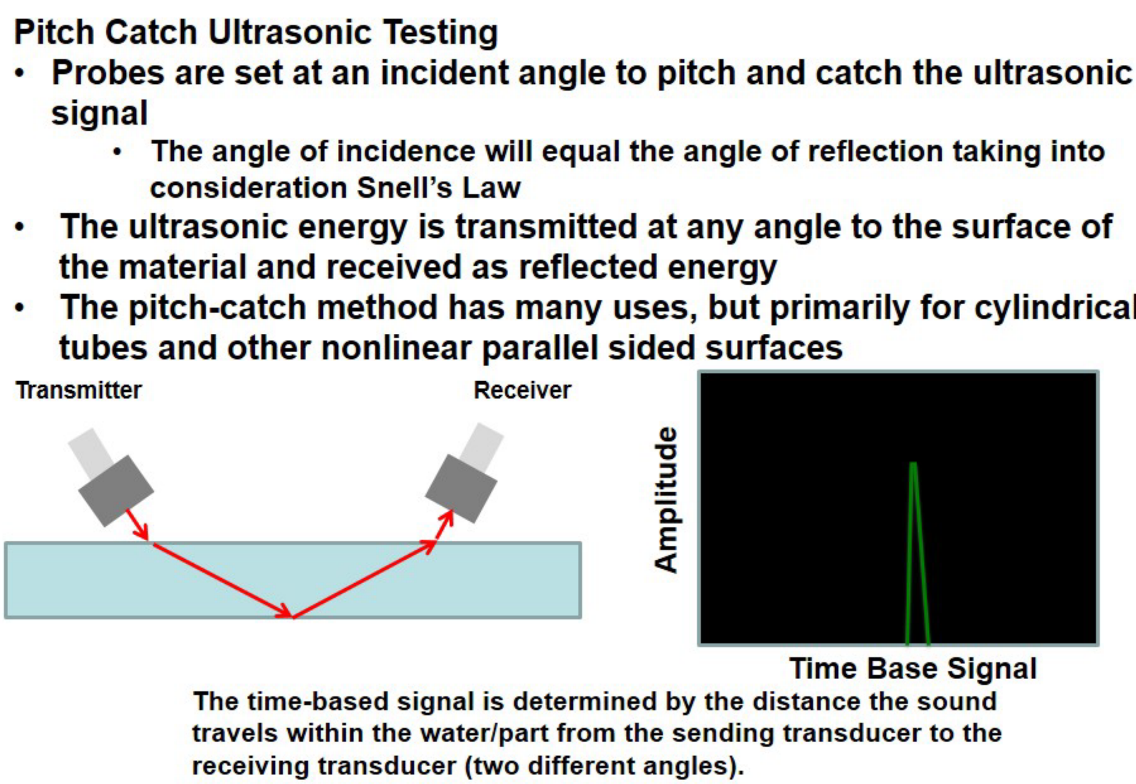


Figure 134. Ultrasonic deployment—Pitch catch UT

Module 3c: Composite NDI - Theory and Practice

Composite A-Scan Inspection for Damage

Figure 135. Composite A-scan inspection for damage

Ultrasonic NDI is commonly used to detect:
- Delamination
- Fiber breakage
- Adhesive disbonds
- Matrix cracking
- Voids
- Porosity

When a defect has been detected, it can then be further evaluated :
- Sizing
- Location
- Type
- Severity

Figure 136. Ultrasonic NDI of composites

UT Instrument

- **Operates at frequency range between 2.5 to 5 MHz**
- **Has Time Corrected Gain (TCG)**
- **Ability to adjust overall gain with active TCG**

UT Probe

- **Recommend frequency between 2.5 and 5 MHz**
- **Delay line thick enough so that the material back wall surface signal occurs before the delay line multiple**
  - **Smaller transducers attenuate more and have difficulty penetrating thicker laminates**

Example transducers for A-Scan damage detection

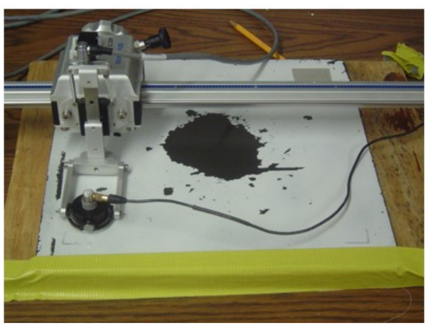
- **GE Inspection Technologies: ALPHA, high energy 3.5 MHz, 0.5" diameter transducer with delay line**
- **Olympus C2002: high energy 3.5 MHz, 0.5" diameter transducer with delay line**

GE Transducer

Olympus Transducer

Couplant

- **Water or water soluble gel is recommended**
- **Grease or oil-based couplants are not recommended**
- **Slow water feed devices are recommended**



Straightedge system or X-Y scanning frame

- **A straight edge is recommended for inspection – not "free-handing"**
- **Index distance no larger than 0.2"**

Figure 137. Ultrasonic setup—A-scan equipment recommendations

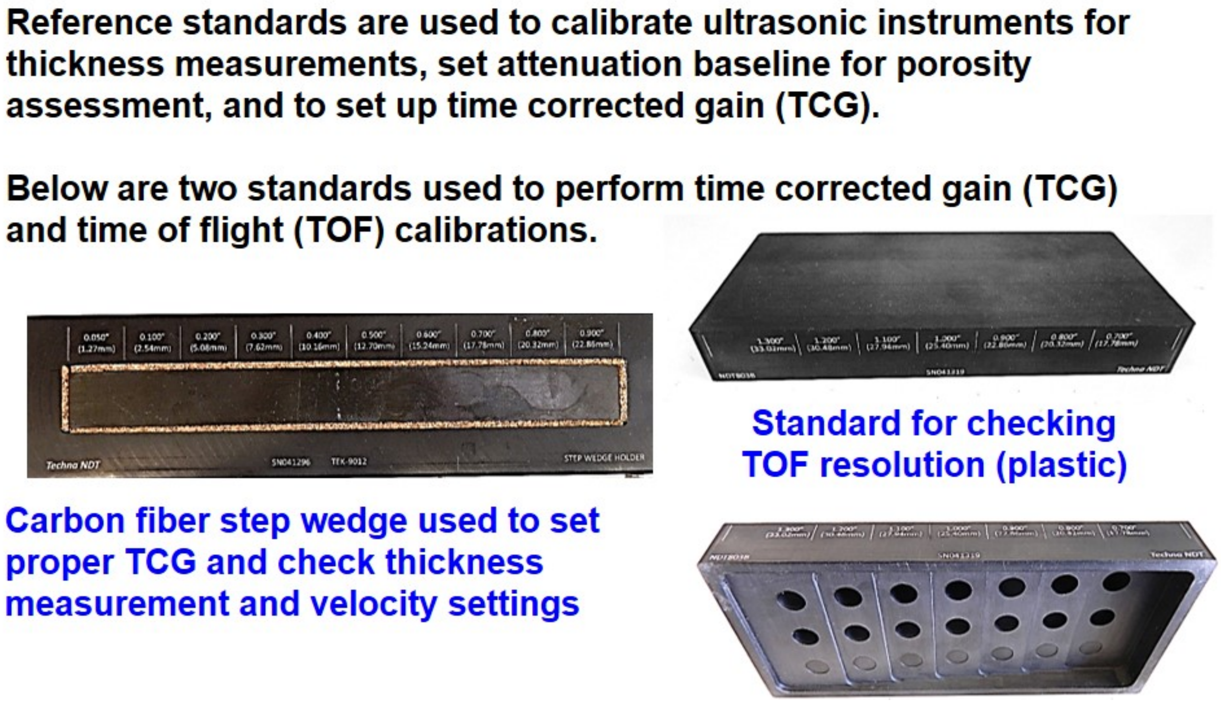
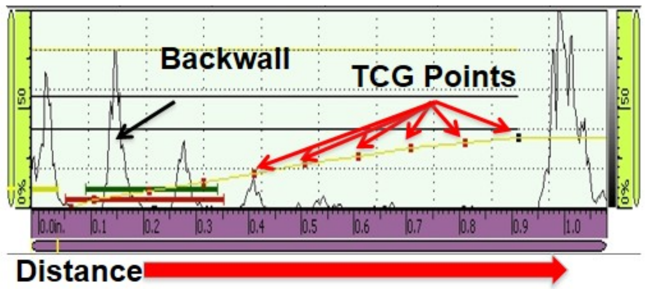
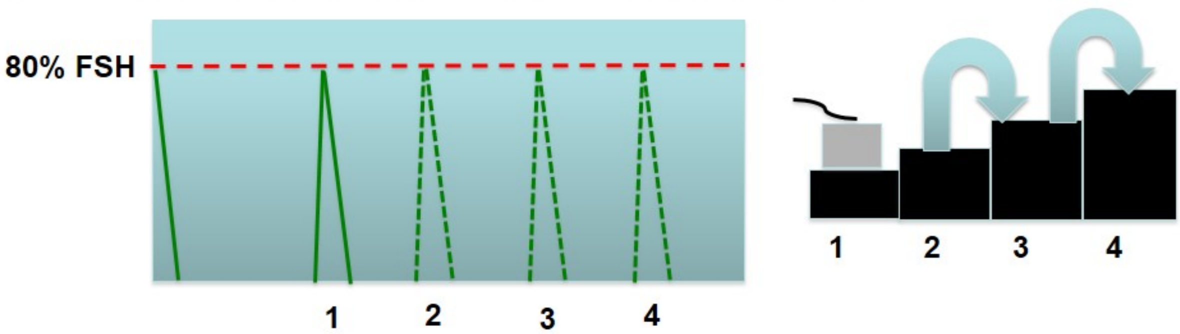


Figure 138. Ultrasonic setup—Composite reference standards



10 TCG points set using the step wedge at 10 different thicknesses ranging from 0.05-0.9"

Time Corrected Gain (TCG) is a method of compensating for a reduction in signal amplitude with increasing thickness. This is achieved by increasing the system gain with time so that the signals appear of equal amplitude.



- **When using TCG, similar reflectors at different depths result in the same % amplitude on the flaw detector screen**
- **A benefit of TCG is that the evaluation of indications can be easily supported by the use of a monitor gate on the flaw detector and the amplitude of the indication above the threshold displayed**

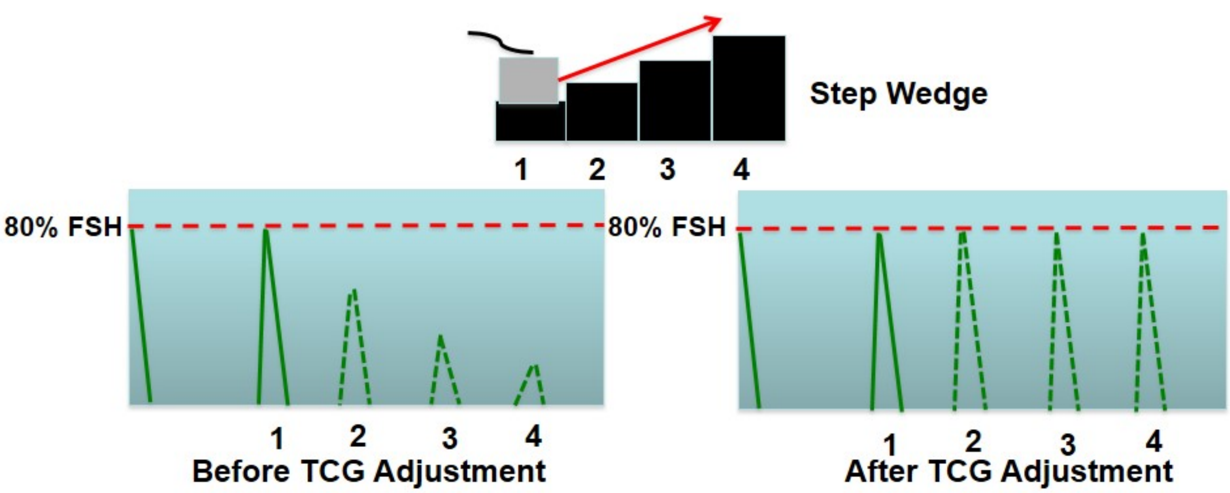


Figure 139. Ultrasonic setup—Utilization of TCG

A-Scan:

- Displays the amount of received ultrasonic energy as a function of time.
- The amount of received energy is plotted along the vertical axis and the elapsed time (related to the sound velocity within the material) is displayed along the horizontal axis

Most instruments with an A-scan display allow the signal to be displayed in:

- Natural radio frequency form (RF)
- Positive or negative half of the RF signal

Defect size can be estimated by comparing the signal amplitude obtained from an unknown reflector to that from a known reflector. Reflector depth can be determined by the position of the signal on the horizontal sweep.

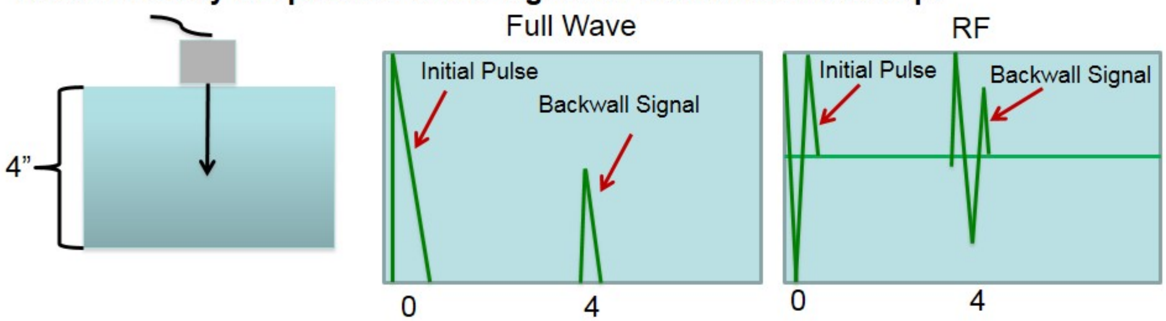
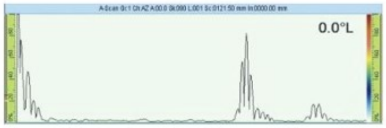


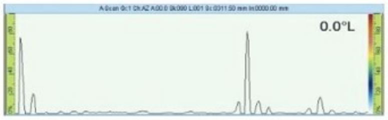
Figure 140. Ultrasonic setup (A-scans) full wave vs. RF

- **Radio Frequency (RF)** - This waveform has 50% of full screen height, and shows the full waveform with both the positive and negative peaks. It contains all signal information.

- **Full Wave (FW)** - Shows the positive and negative peaks, but the negative peaks are reversed and made positive



- **Positive Half Waves (HWP)** - shows only positive peaks



- **Negative Half Waves (HWN)** - shows only negative peaks

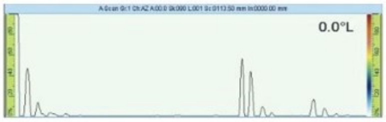


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Figure 141. Ultrasonic setup—Types of ultrasonic waveforms

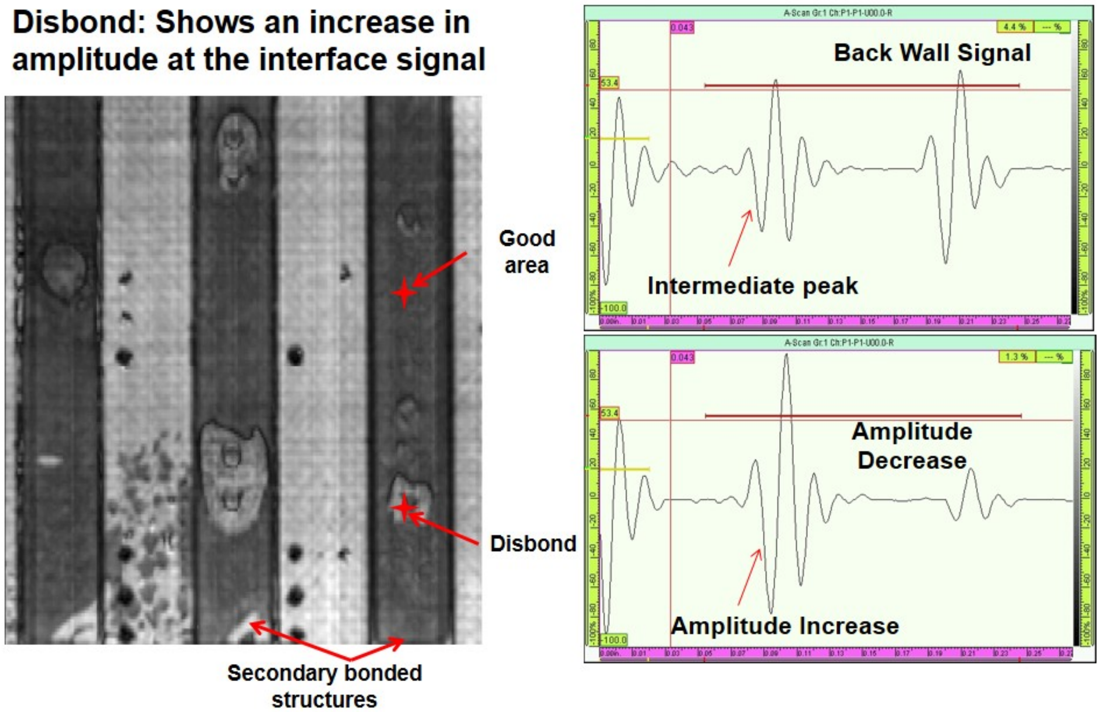


Figure 142. Ultrasonic setup—Example RF display of good and bad bond line

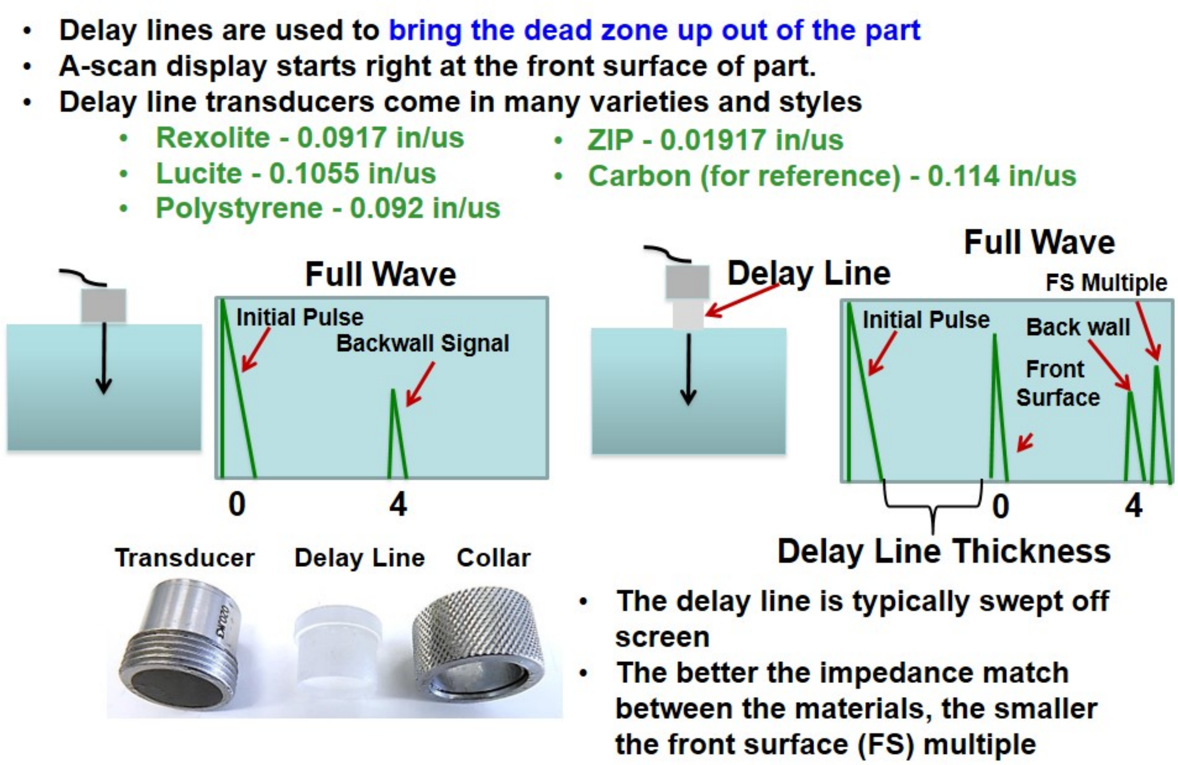


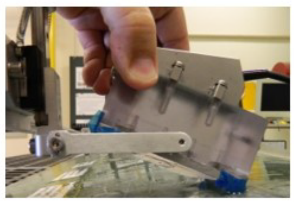
Figure 143. Ultrasonic setup—Use of delay lines

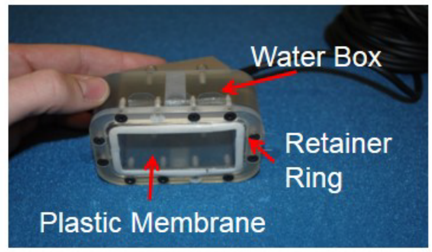
Ultrasonic Delay Lines

- **Used on thin materials where the Dead Zone could interfere with near surface defects**
- **Delay line thickness depends on thickness of part**
  - **where the multiple pops up**
- **ZIP Probes are acoustically matched plastic to the carbon fiber part**
  - **Improved near surface resolution**
- **A water box can be used as a delay line with flexible surface adaptable to contours**

Zip Tip Transducer

Plastic Delay Lines





Water Box With Plastic Membrane

Ensure delay line is properly coupled to transducer

Figure 144. Ultrasonic setup—Ultrasonic delay lines

Module 3d: Composite NDI - Theory and Practice

Mapping Damage

Figure 145. Mapping damage

Composite Laminate Inspection

- Nondestructive inspection is typically only required following a visible indication of damage on the surface of the part
- Determine inspection area and clean the surface
- Visibly inspect the surface surrounding the damage
- Determine the thickness range and material used
  - Refer to the structural repair manual (SRM)
  - Refer to engineering drawings
- Check UT calibration
- Use the relevant calibration reference standard to check the thickness
- Mark substructure on the surface of aircraft
- Inspect areas of uniform thickness
  - Then inspect over substructure elements

Figure 146. Mapping damage—Preparing for inspection

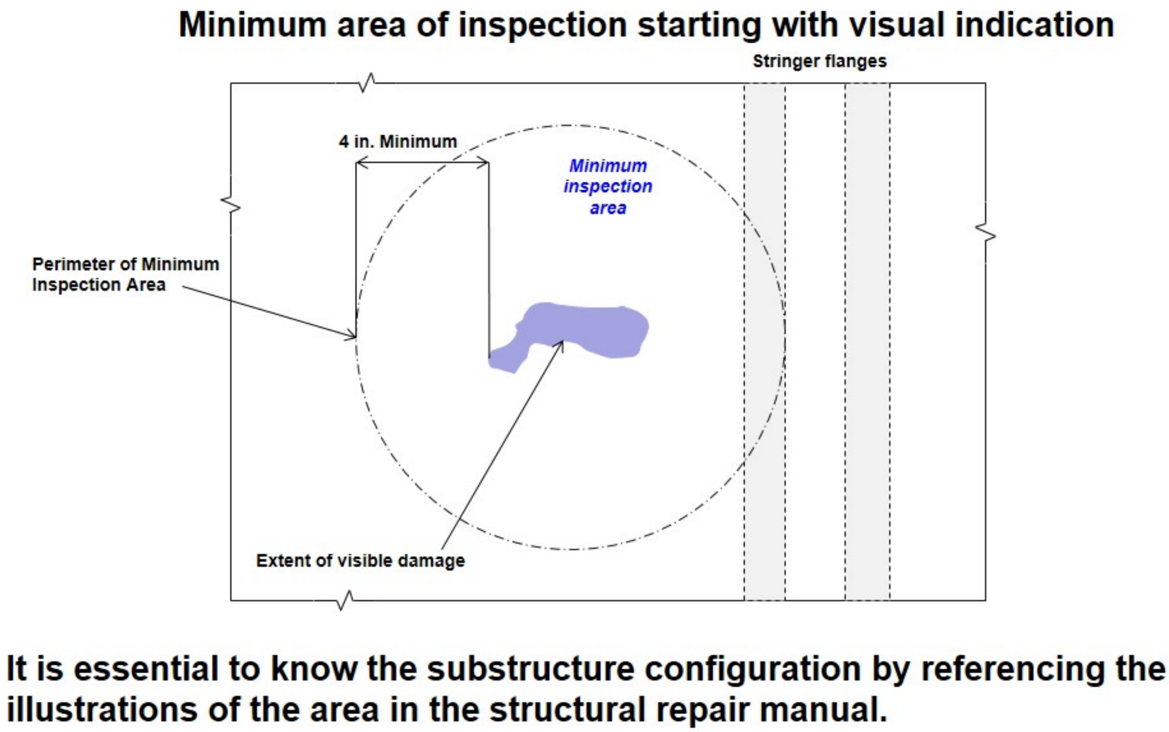


Figure 147. Mapping damage—Minimum area of inspection

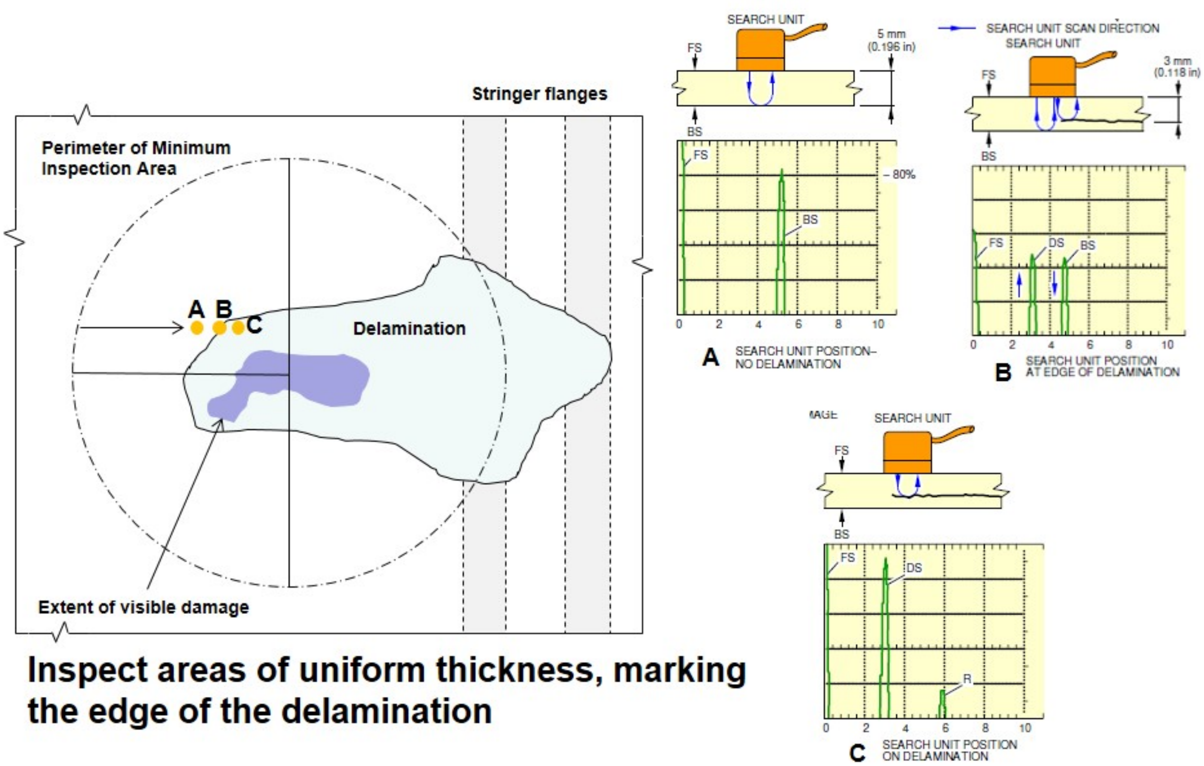


Figure 148. Schematic of mapping damage

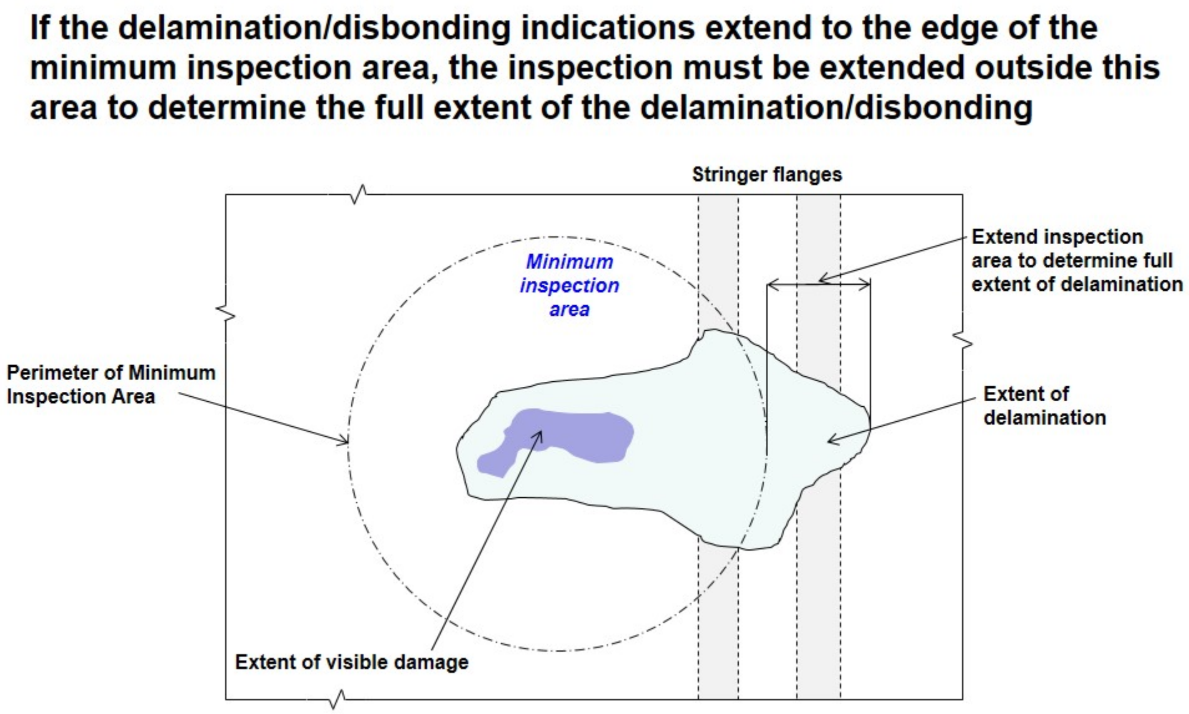


Figure 149. Mapping damage—Extending the minimum area of inspection

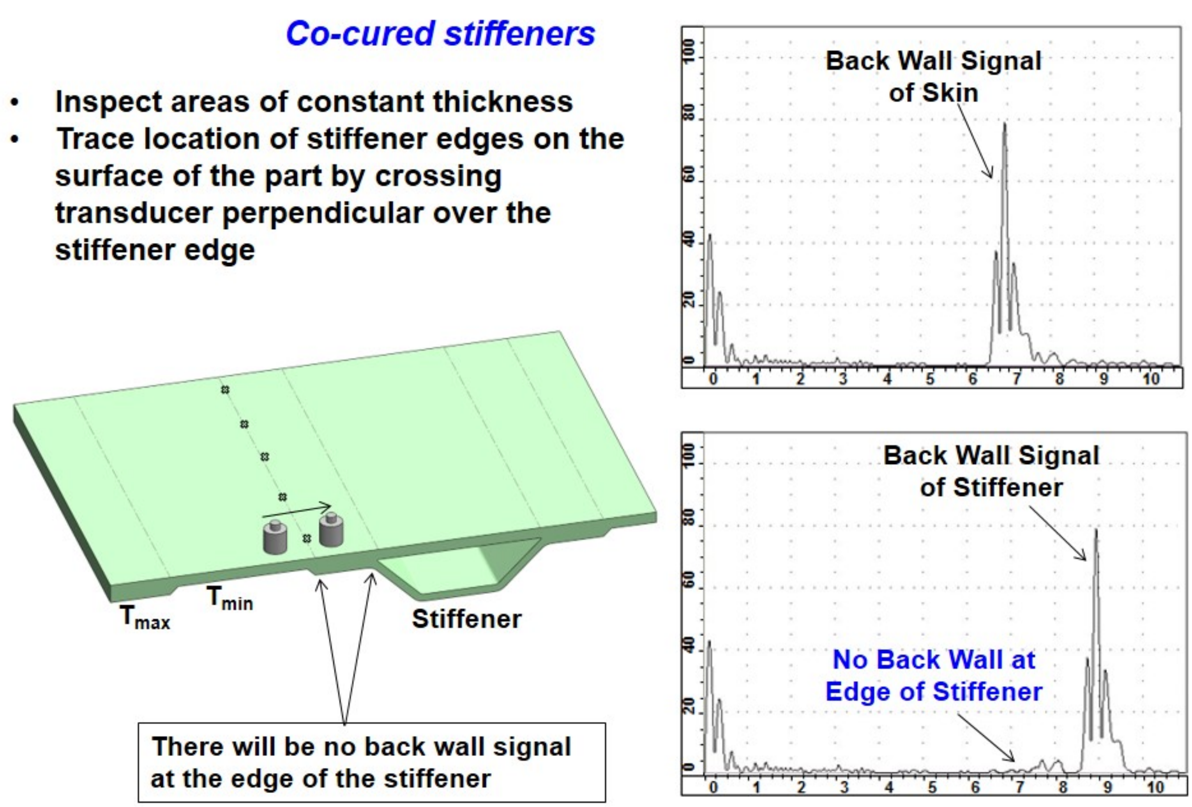


Figure 150. Scanning procedure—Co-cured laminate

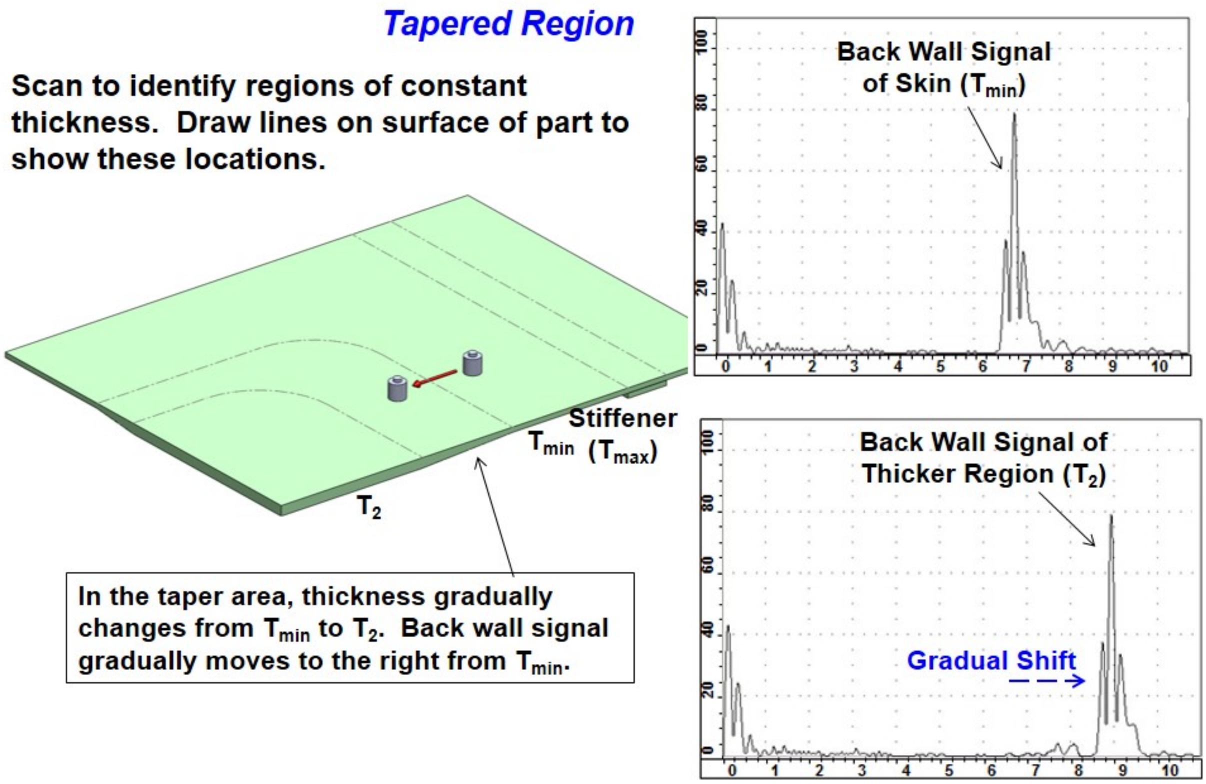


Figure 151. Scanning procedure—Tapered laminate

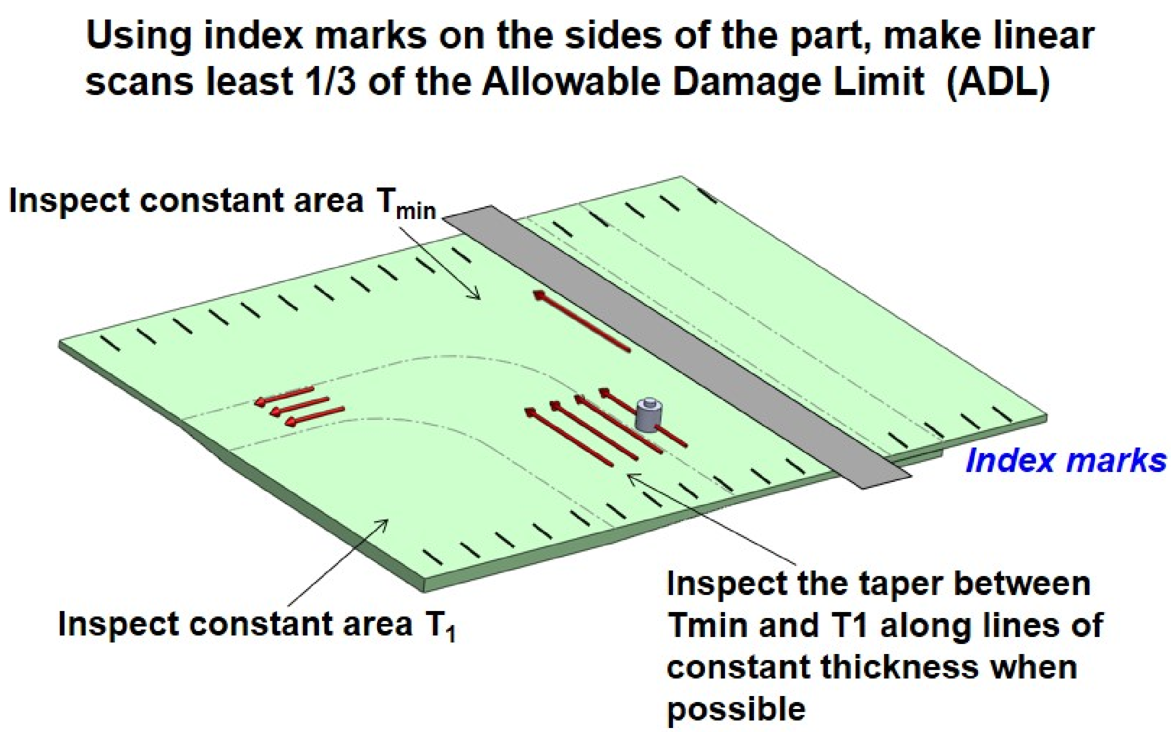


Figure 152. Scanning procedure—Indexing

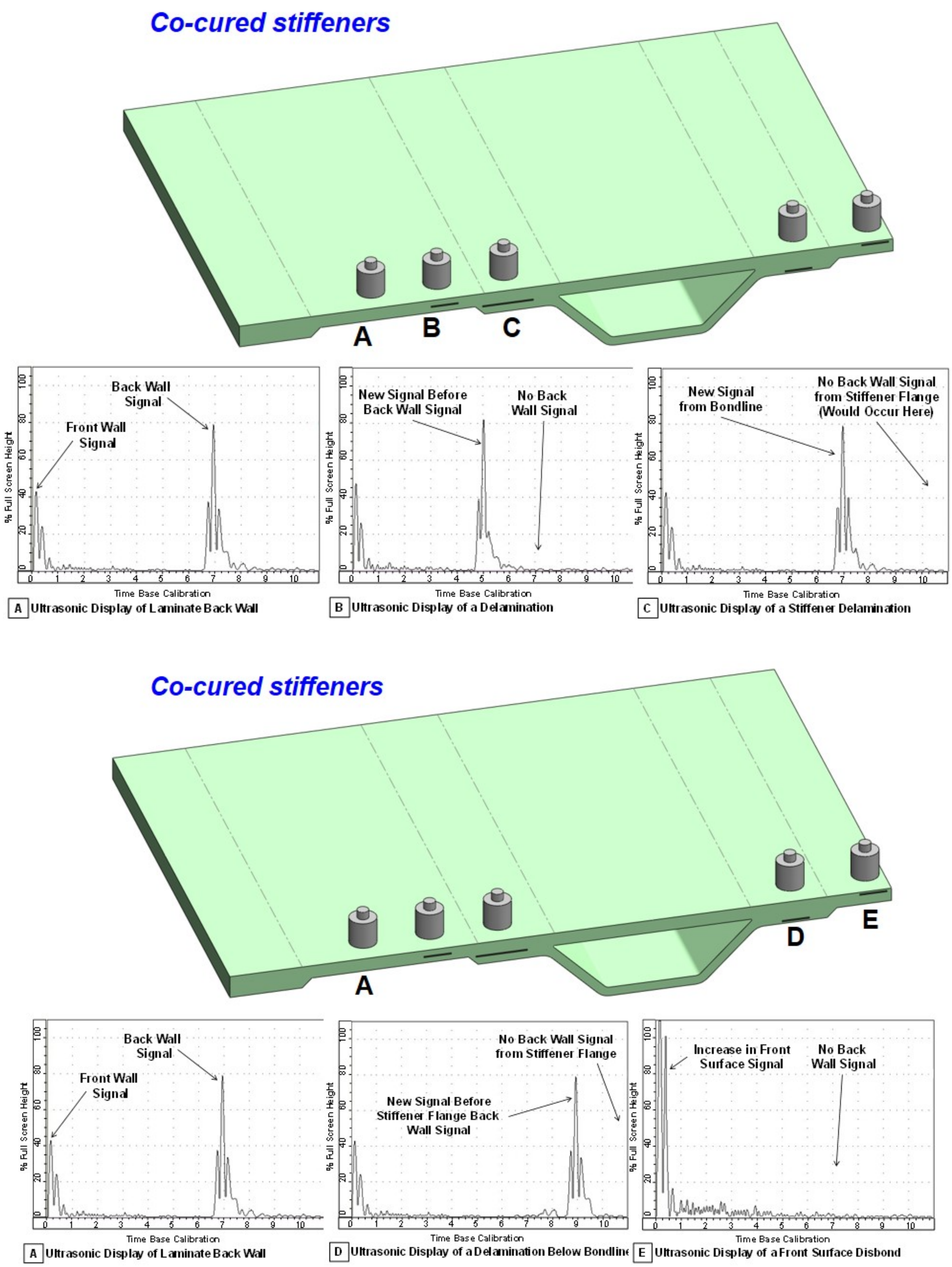


Figure 153. Ultrasonic signals from damaged structure—Co-cured stiffeners

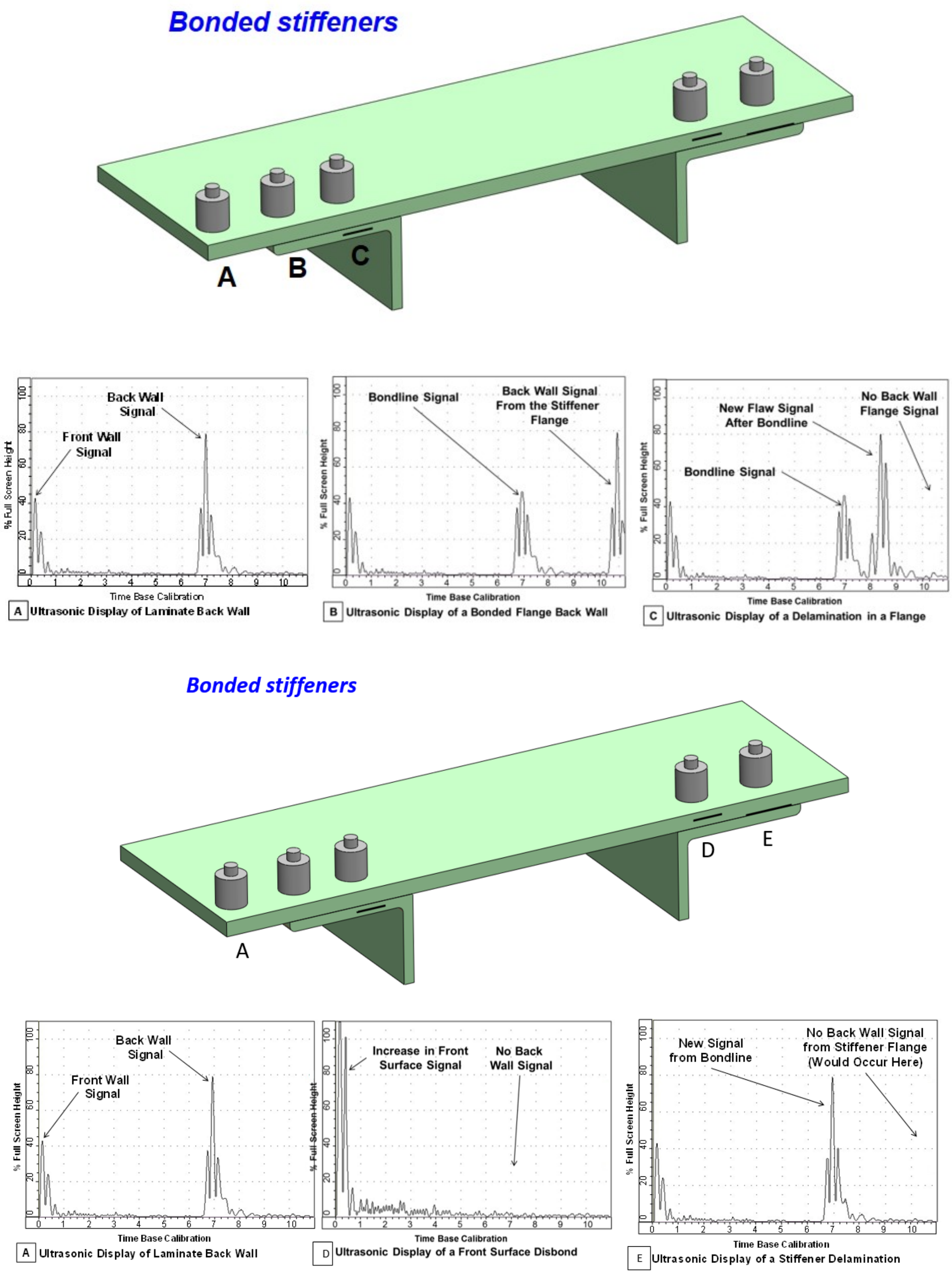


Figure 154. Ultrasonic signals from damaged structure—Bonded stiffeners

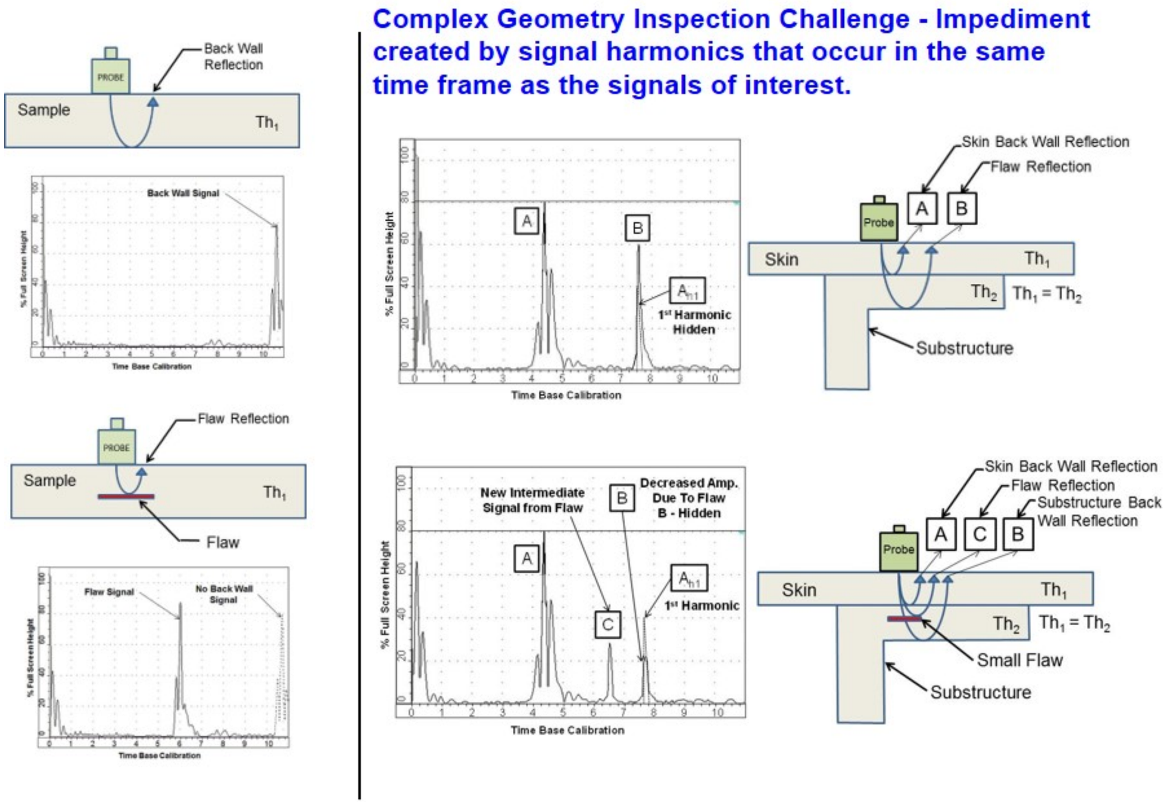


Figure 155. Transitioning inspectors from “average” to “good” to “outstanding” guidance on specific composite challenges

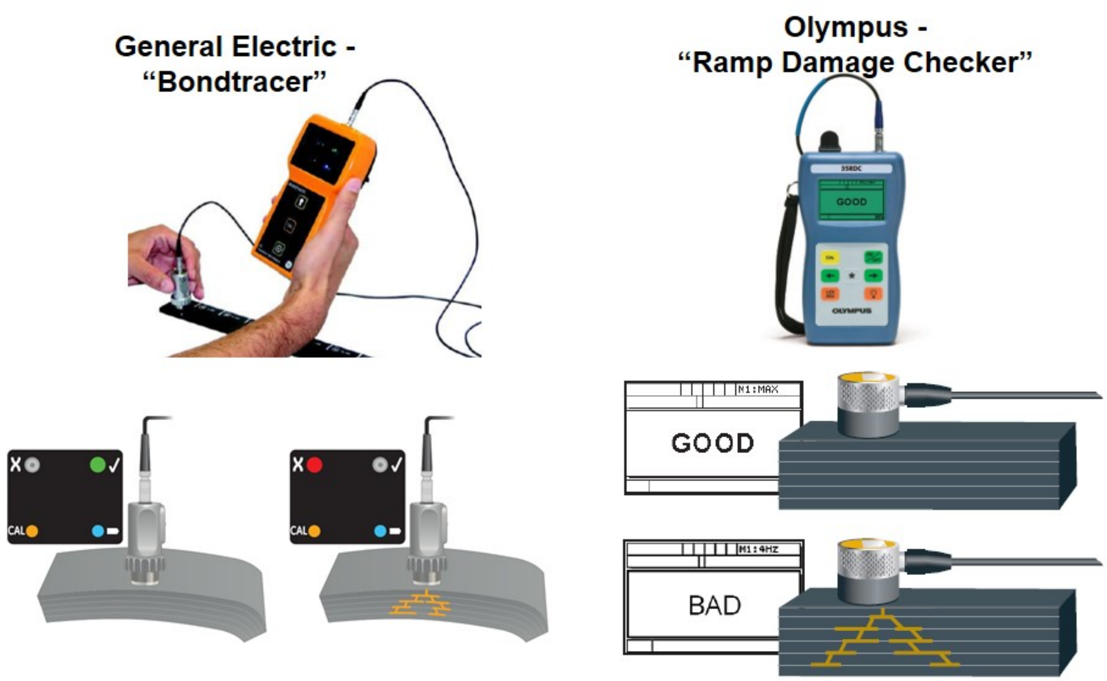


Figure 156. Ultrasonic deployment options—“go” & “no-go” devices

Inspection Method				
	Resonance	Pitch-Catch	MIA	Pulse Echo
Advantages	Applicable to structures with multiple layers with or without honeycomb. Detects unbonds between any layer or in honeycomb. Detects small defects (larger than the dieameter of the receiving transducer). Locates layer position of un-bonds. Applicable to laminate or honeycomb structures. Applicable to complex shapes.	Applicable to honeycomb structures with thick and thin skins. Detects small unbonds (search unit diameter or smaller). No couplant required. Fast scanning rates.	Applicable to complex shapes. Detects small near surface unbonds (larger than diameter of search unit). No couplant required. Can be used on irregular or curved shapes. Most effective on honeycomb structures with skin-to-core disbonds and core disbonds.	Works on structures with muliple layers or substructure. Able to be detect disbonds between the layers of laminate. Able to determine flaw size and location. Can determine resin rich or resin starved. Primarily used on metal or carbon laminate structures.
Limitations	Access to both sides is required. Can not determine layer position of unbond. Couplant required.	Inspection from both sides required. Does not detect far side disbonds. Applicable only to honeycomb sandwich structures with single layer skins. Depending on set up, couplant could be required. Reduced effectiveness for unbonds greater than 0.80" below inspecton surface. Probe is directional with respect to locating boundaries of unbonds.	Applicable to only near surface unbonds. Works best on unbonds between top sheet and adhesive layer. Works best on metals, but can be used on composites.	Requires Couplant. A-scan screen intrepretations can be difficult. Must have a 3:1 noise to signal ratio.

Source: USAF 33B-1-1

Figure 157. Ultrasonic method comparison

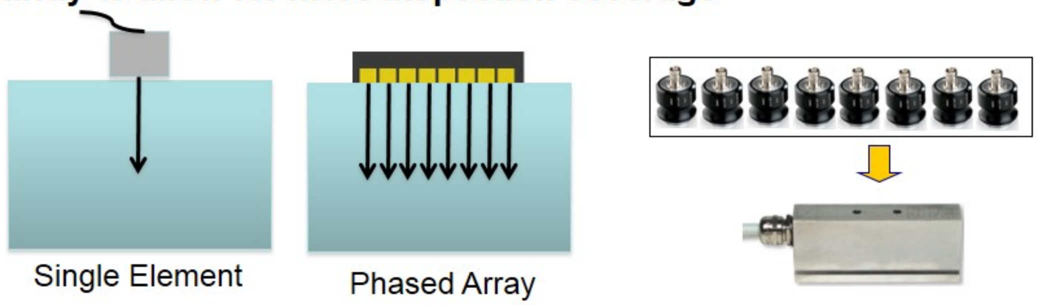
Module 3e: Composite NDI - Theory and Practice

Composite C-Scan Inspection for Damage

Figure 158. Composite C-scan inspection for damage

Phased Array

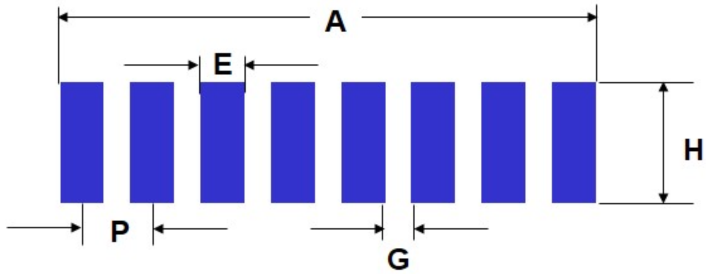
- Same basic physics as conventional ultrasonic inspection
- Uses multiple PZT elements phasing across a single probe
- The advantage of using an array vs. conventional single element transducer on composites is increased coverage
- Instead of indexing across the part, you can utilize the width of the array to allow for more inspection coverage



- Conventional UT - single transducer typically sends and receives ultrasonic waves
- Phased Array UT - transducer assembly with 16 to 256 small, individual elements that can each be pulsed separately

Figure 159. UT theory—Phased array ultrasonic inspections

- Frequency: 0.5 MHz - 10 MHz
- Total number of elements in array: 16 - 256



Probe Parameters

- A - Active aperture in steering or active direction
- H - Height or Elevation, aperture in mechanical or passive direction
- E - Elevation, width of an individual element
- P - Pitch, center-to-center distance between two successive elements
- G - Spacing between elements

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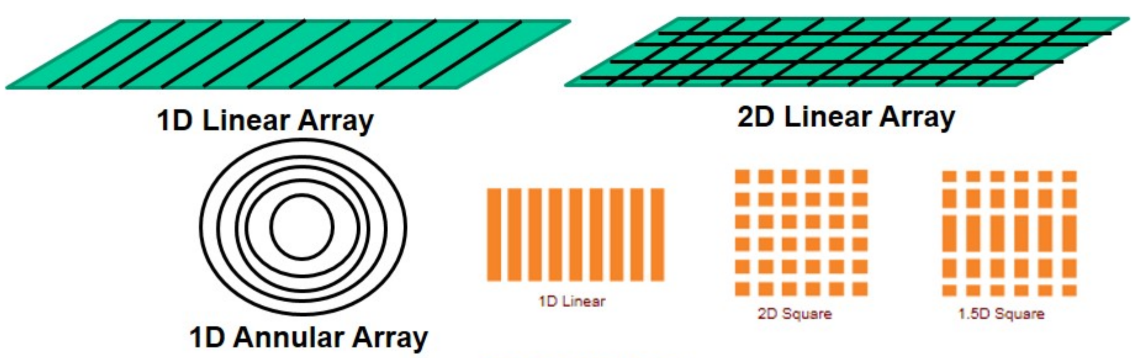
Figure 160. UT theory—Transducer parameters

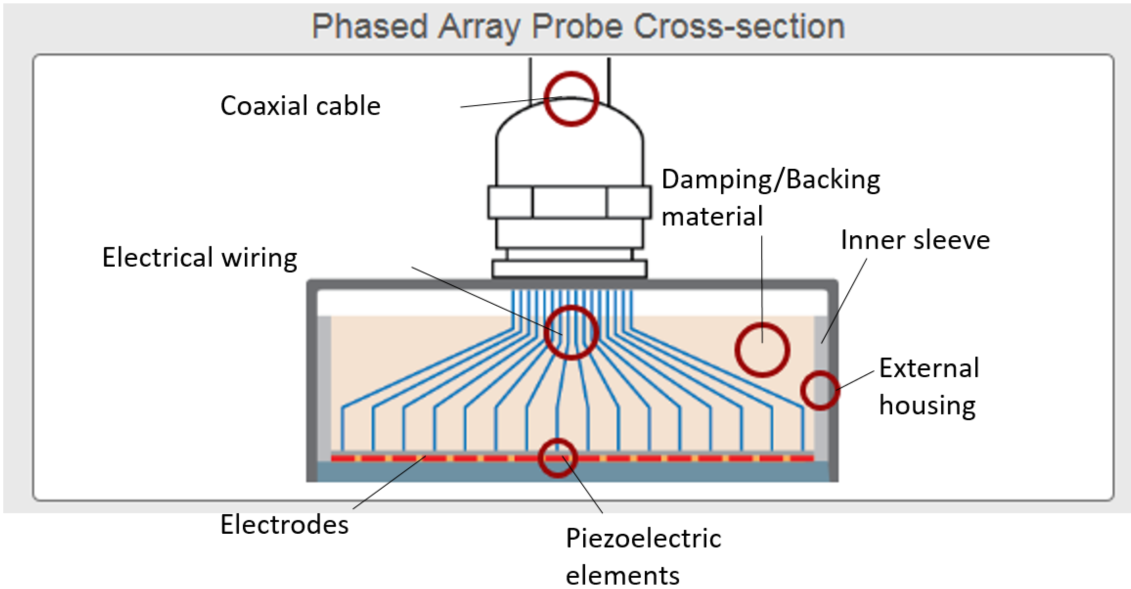
Array Probe Element Patterns

- 1D linear array is the most widely used array in composite inspection.
- Linear array vs. phased array probe? They are the same, software is different
- 2D vs. 1D linear array - allows for better steering and focus of the sound path
- Annular array - typically used in industrial manufacturing for fillets, forgings, and large material inspections (small defects)



OLYMPUS

Figure 161. UT theory—Element patterns



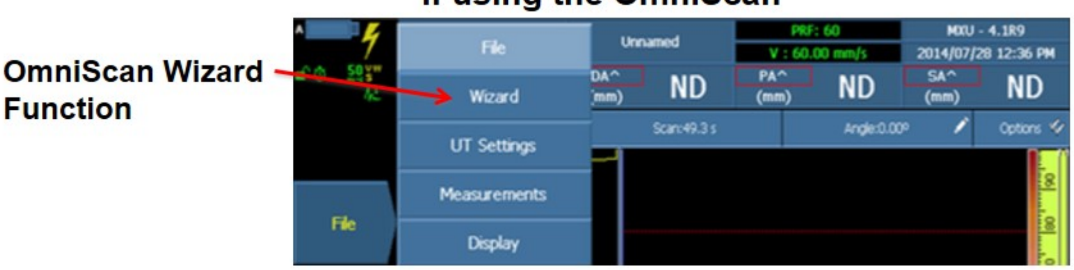
OLYMPUS

Figure 162. Composite phased array equipment set up—Typical schematic of phased array probe

General Equipment Settings

- **Set them material – composite**
- **Probe (auto recognize) and wedge information**
- **Specify the element quantity in an aperture - between 5 and 8, five is recommended for composite laminates**
- **Specify the first and last element of the array. First element = 1, and last element = # of elements in the transducer**

If using the OmniScan



OmniScan Screen Capture Showing Wizard

Figure 163. UT theory—Phased array settings

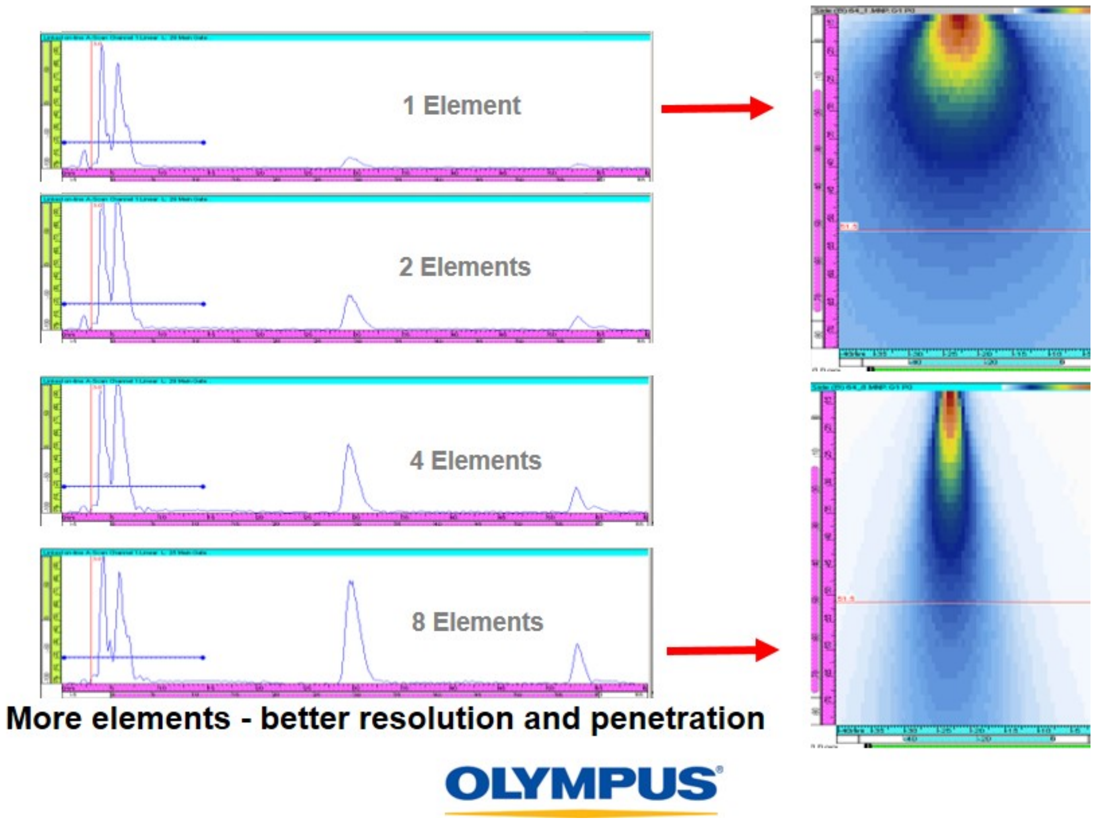


Figure 164. UT theory—Effects of changing aperture

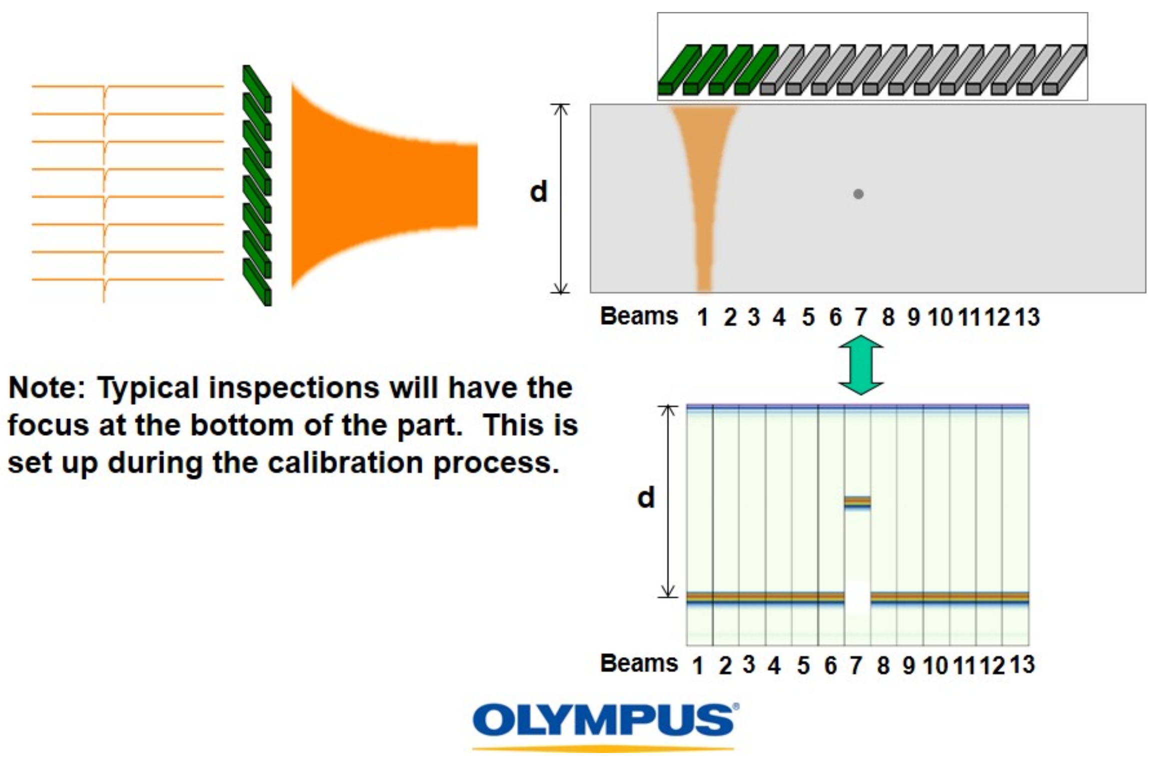
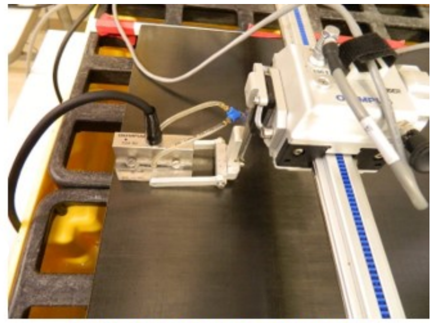


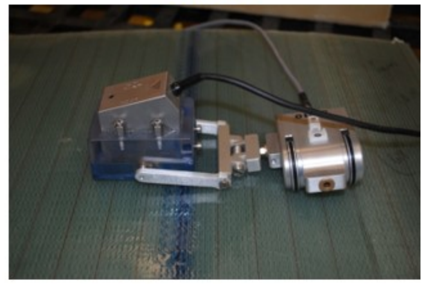
Figure 165. Phased array equipment—Setup focus depth determination

Wedges:

- **Couple sound energy from the transducer to the test piece**
- **Used in shear wave, longitudinal and straight beam and linear scans.**
- **Protect transducer from excessive wear**
- **Select:**
  - **Material**
  - **Thickness**
  - **Angle**
  - **Contour**
  - **Scanner attachments**



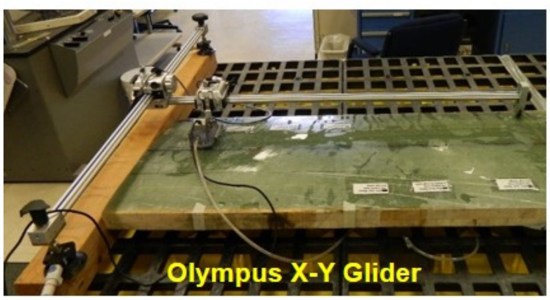
3.5L64 PA-UT with X-Y Glider

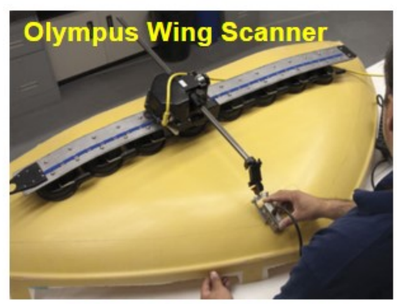


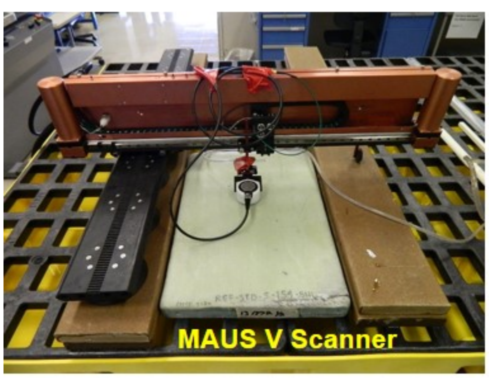
1.5L16 PA-UT Sealed Water Box with Mouse

Figure 166. Ultrasonic deployment options—Wedges

X-Y Scanners: Used to encode scans in both the X and Y direction







Used with both phased array or single element transducers

Figure 167. Ultrasonic deployment options—Scanners

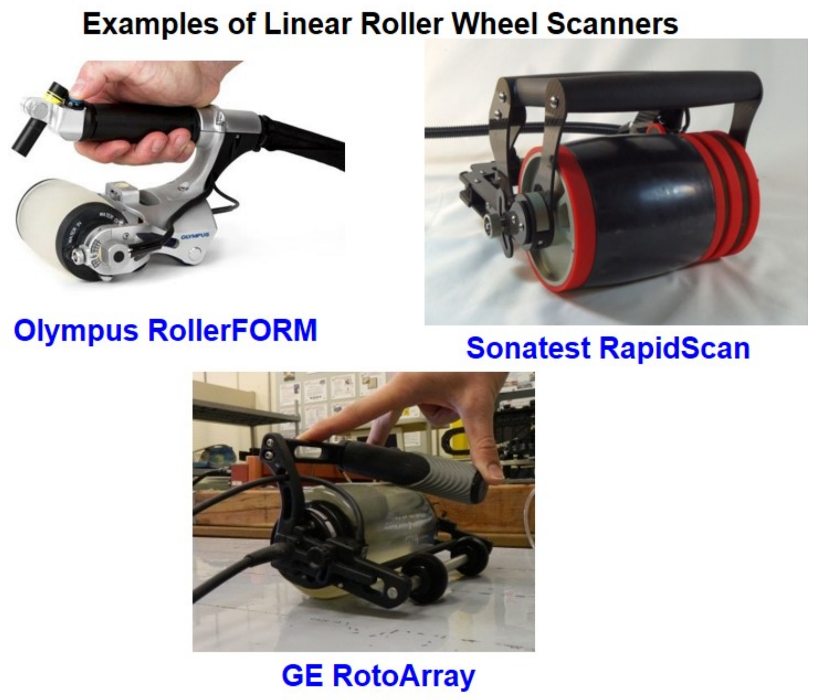


Figure 168. Ultrasonic deployment options—Rolling wheels

Gate	Function	Location	Synchro	Peak Selection	Peak Measure
Interface Gate (Yellow)	Track Changes in position of Interface Signal	Start: Immediately before entry signal. End: Immediately after entry signal	Pulse	Max Peak	Edge
TOF Gate (Red)	Monitor the Time-of-Flight	Start: Immediately after entry signal End: After T_{max} backwall signal	I Gate	Max Peak	Peak
Amp Gate (Green)	Monitor backwall signal amplitude	Start: Before T_{min} backwall signal. End: After T_{max} backwall signal.	I Gate	Max Peak	Peak

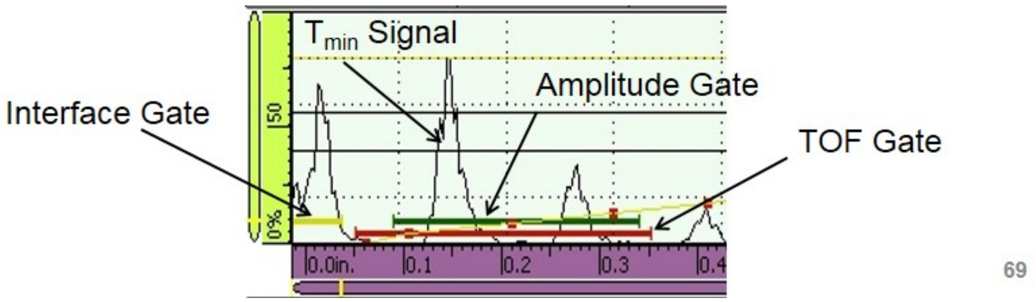


Figure 169. Ultrasonic setup—Gate settings

Initial set up files may be available from the aircraft OEM for both A-scan and C-scan array inspections:

Caution

Downloaded files will provide initial set up of the instrument with OEM recommended settings.

Final calibration using reference standards is required before use.

Figure 170. Ultrasonic setup files—Caution

B-Scan

- The B-scan presentations is a cross-sectional view of the test specimen.
- Time-of-flight is displayed along the vertical axis and the linear position of the transducer is displayed along the horizontal axis
- The B-scan is typically produced by establishing a trigger gate on the A-scan

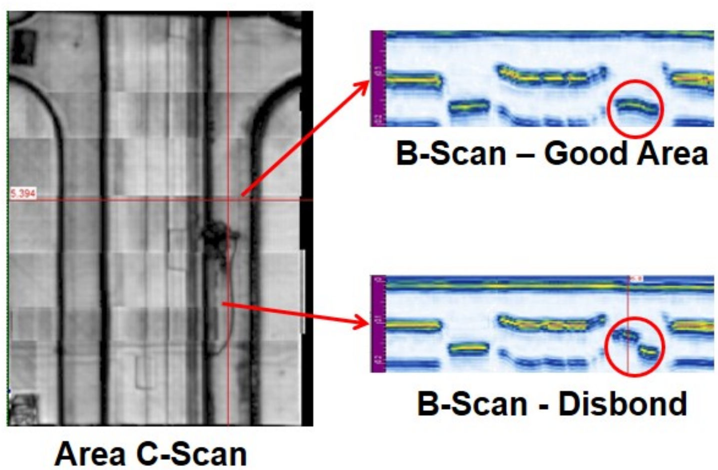


Figure 171. Ultrasonic setup—B-scan display of good and bad bond lines

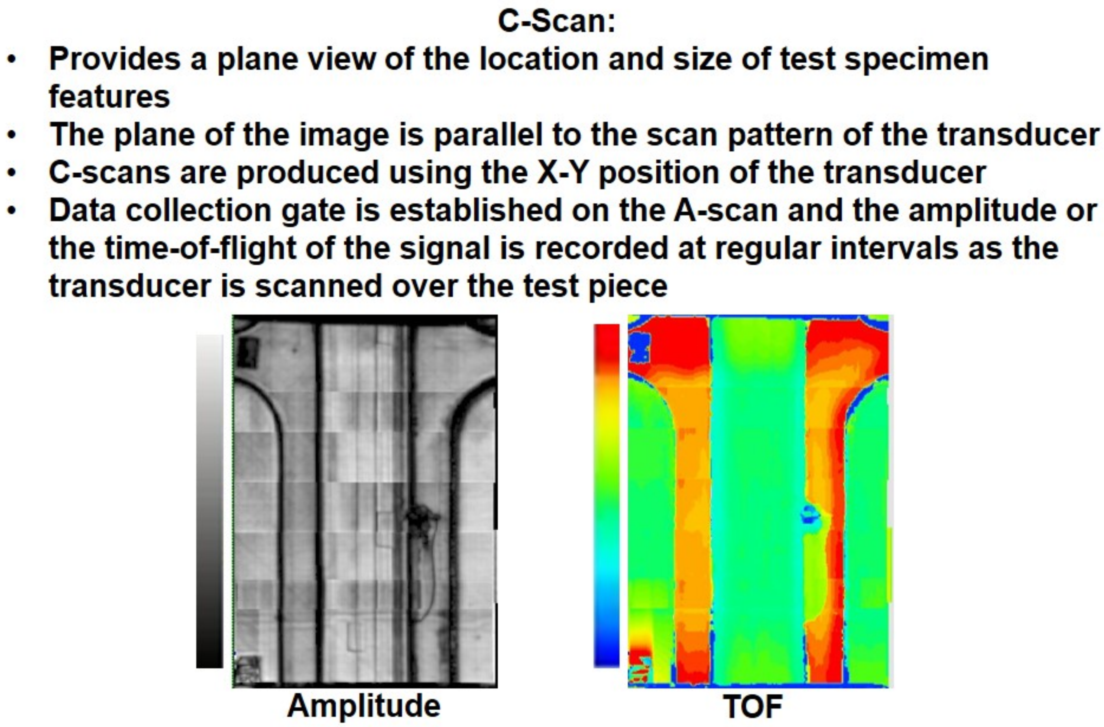


Figure 172. Ultrasonic setup—C-scan display of good and bad bond line

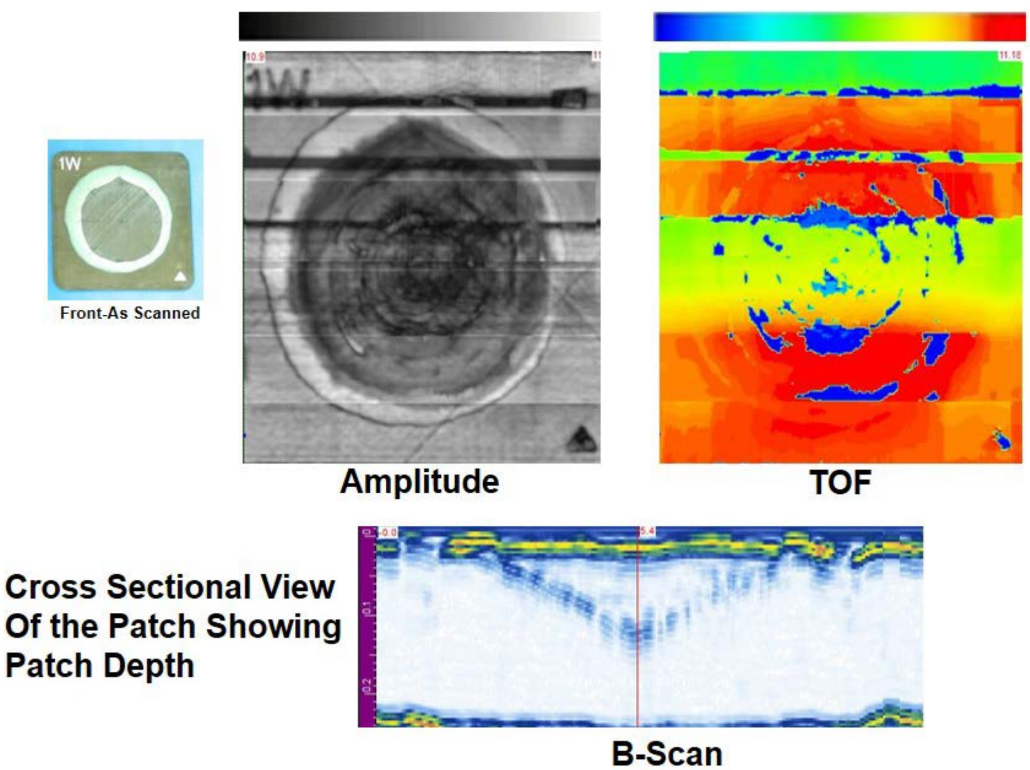


Figure 173. Ultrasonic setup—C-scans and B-scan

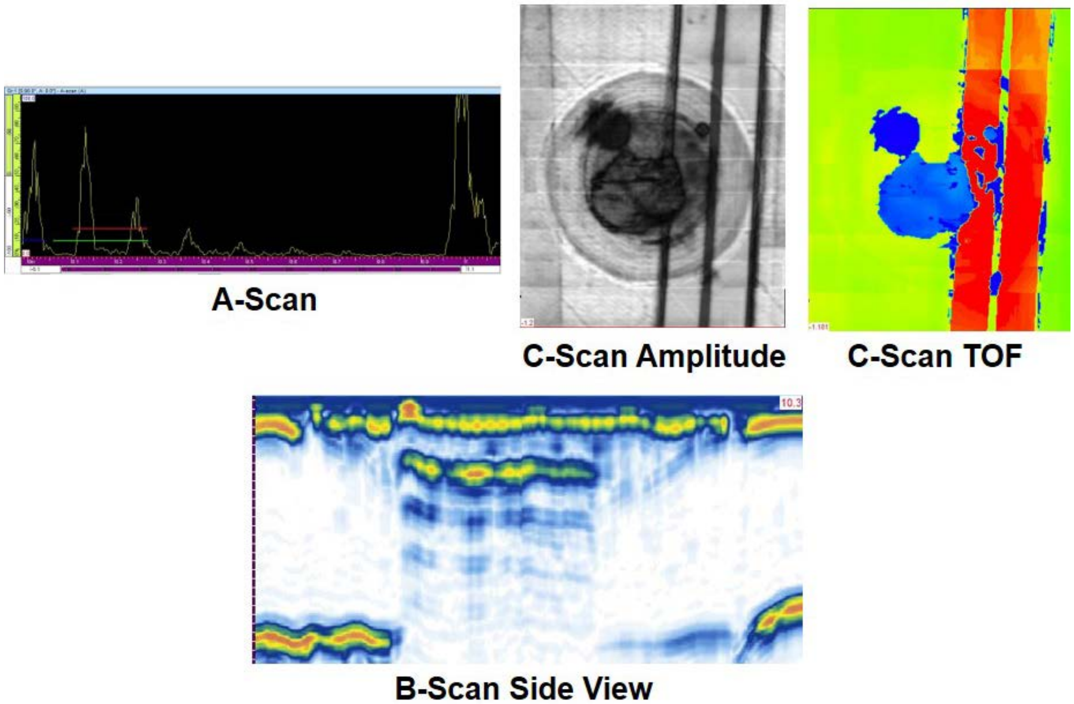


Figure 174. Example of A-scan, B-scan, and C-scan

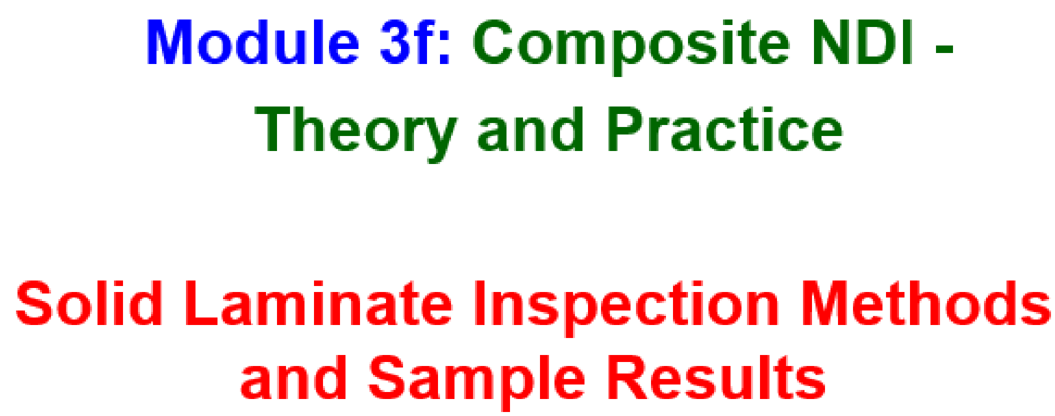


Figure 175. Solid laminate inspection methods and sample results

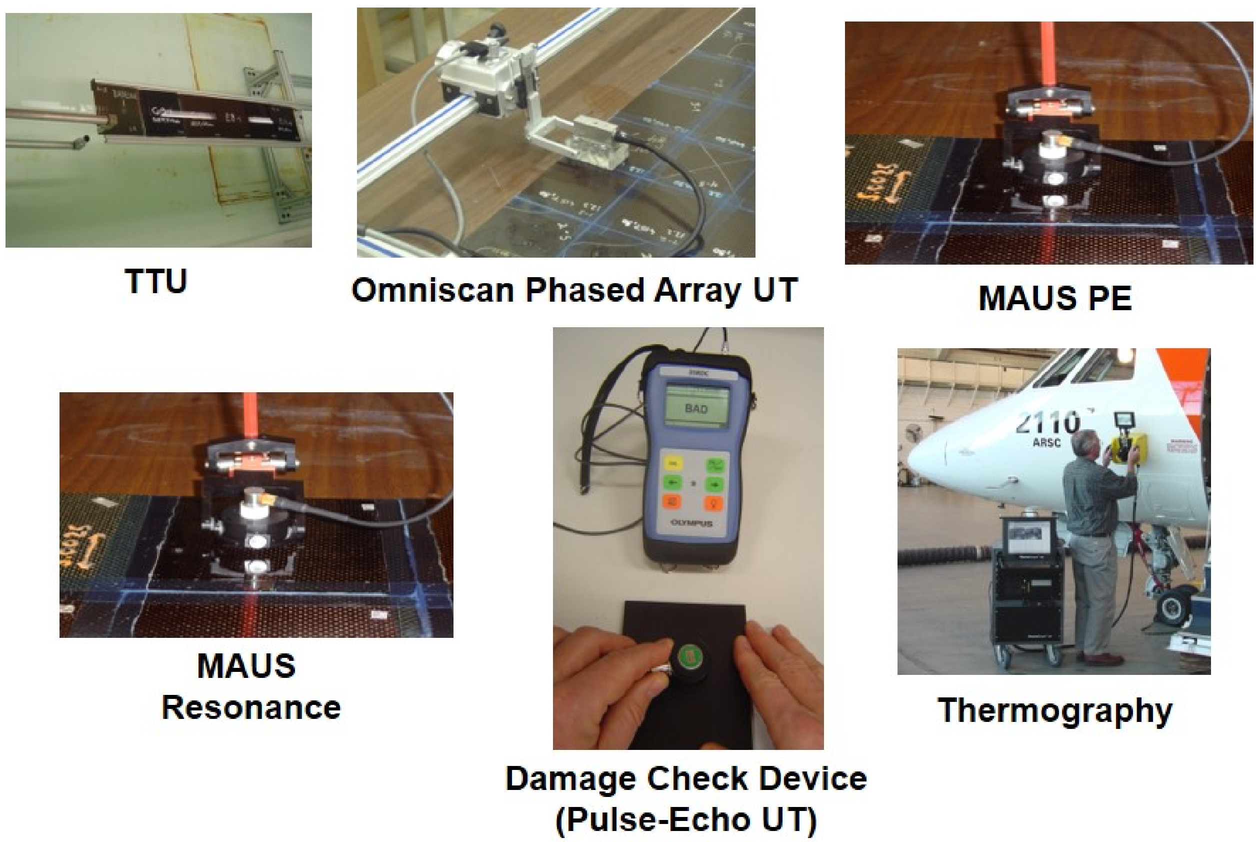


Figure 176. Composite inspection methods

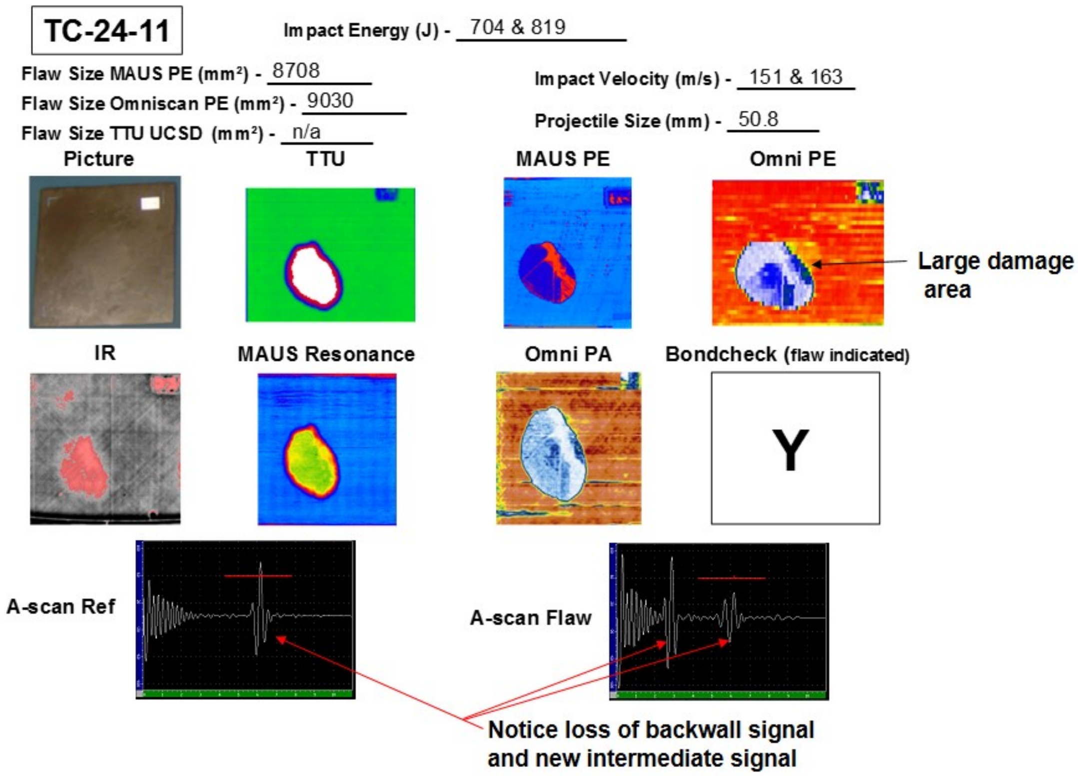


Figure 177. Sample results of solid laminate impact damage

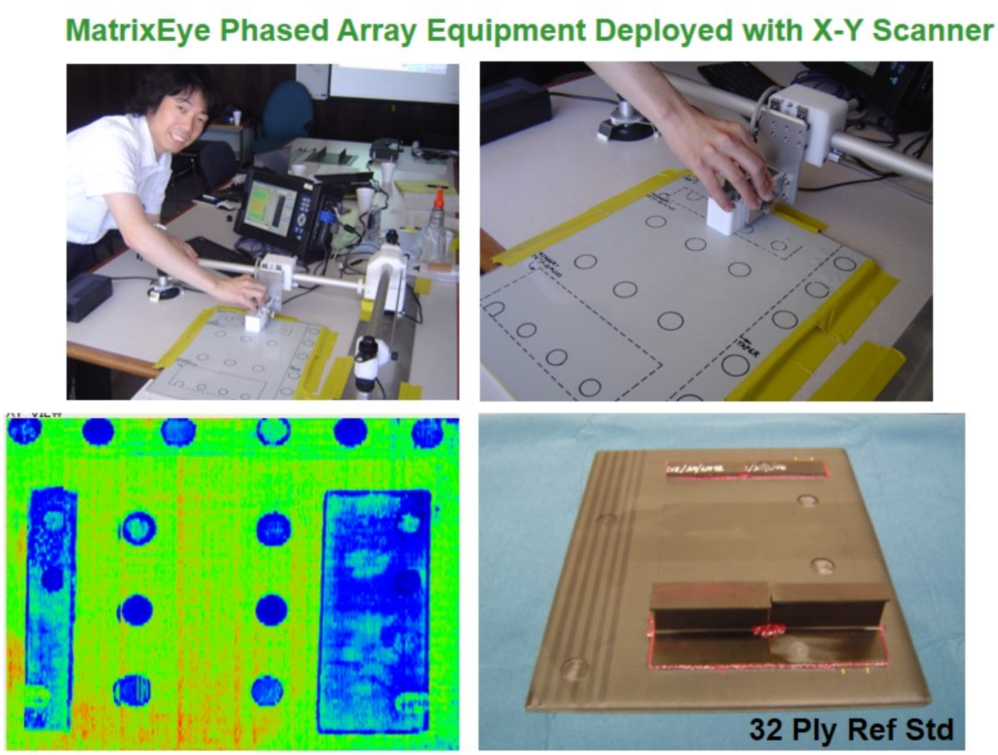


Figure 178. MatrixEye™ phased array

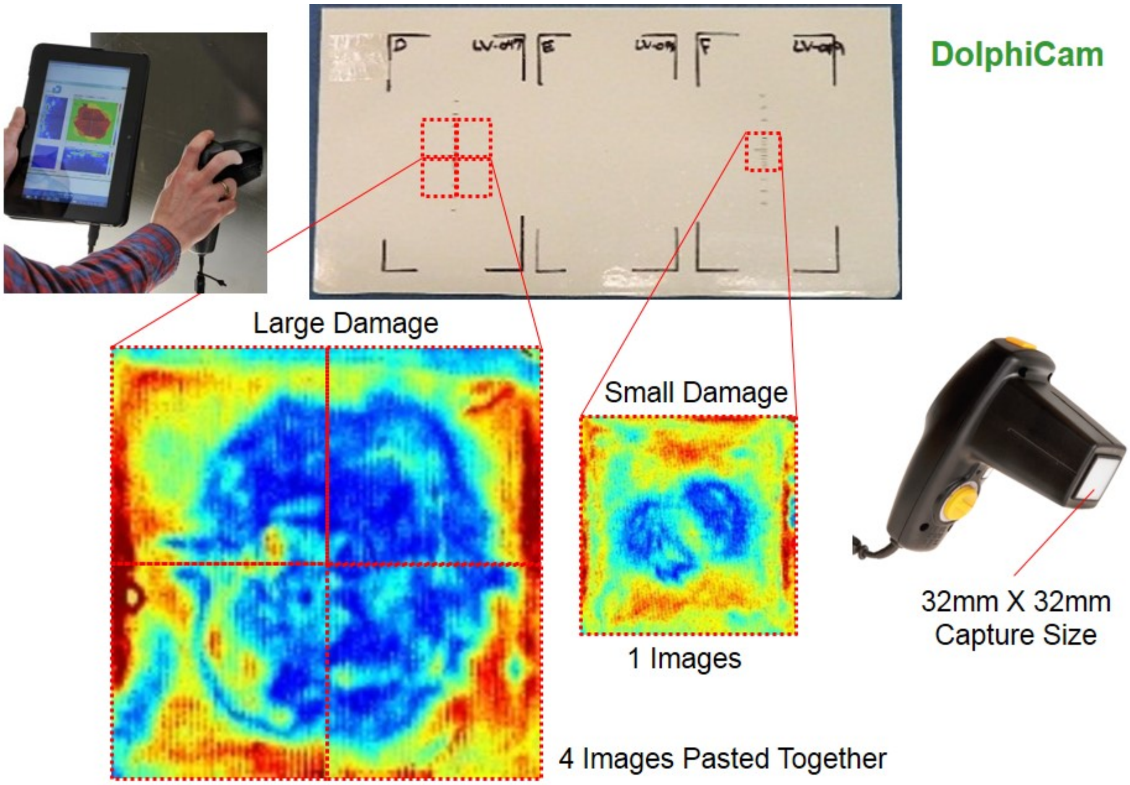


Figure 179. Example of DolphiCam inspection for impact damage

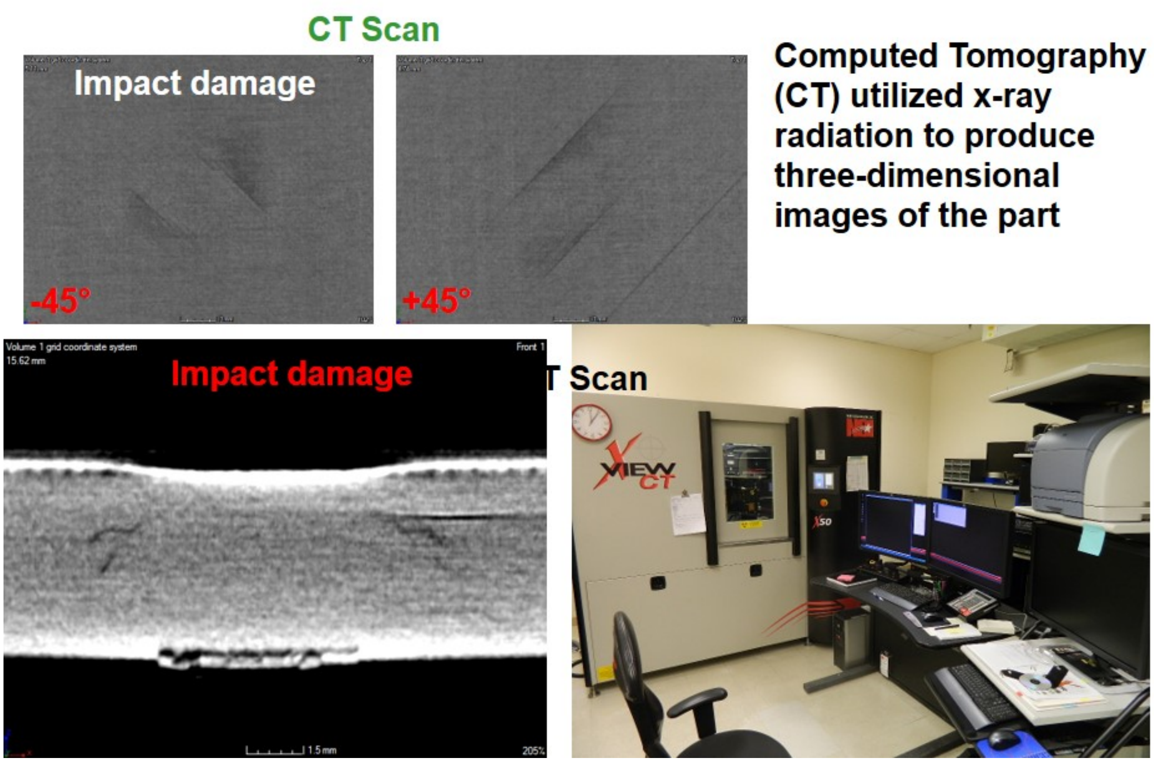


Figure 180. Computed tomography example of impact damage

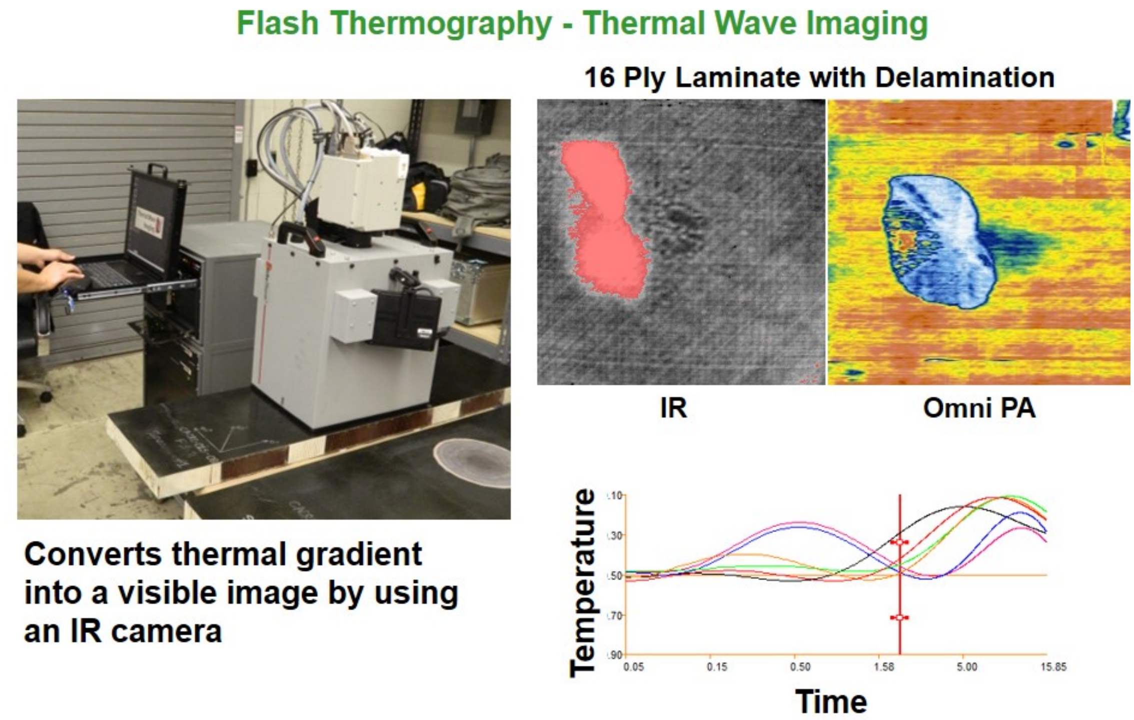


Figure 181. Thermal wave imaging inspection result of impact damage

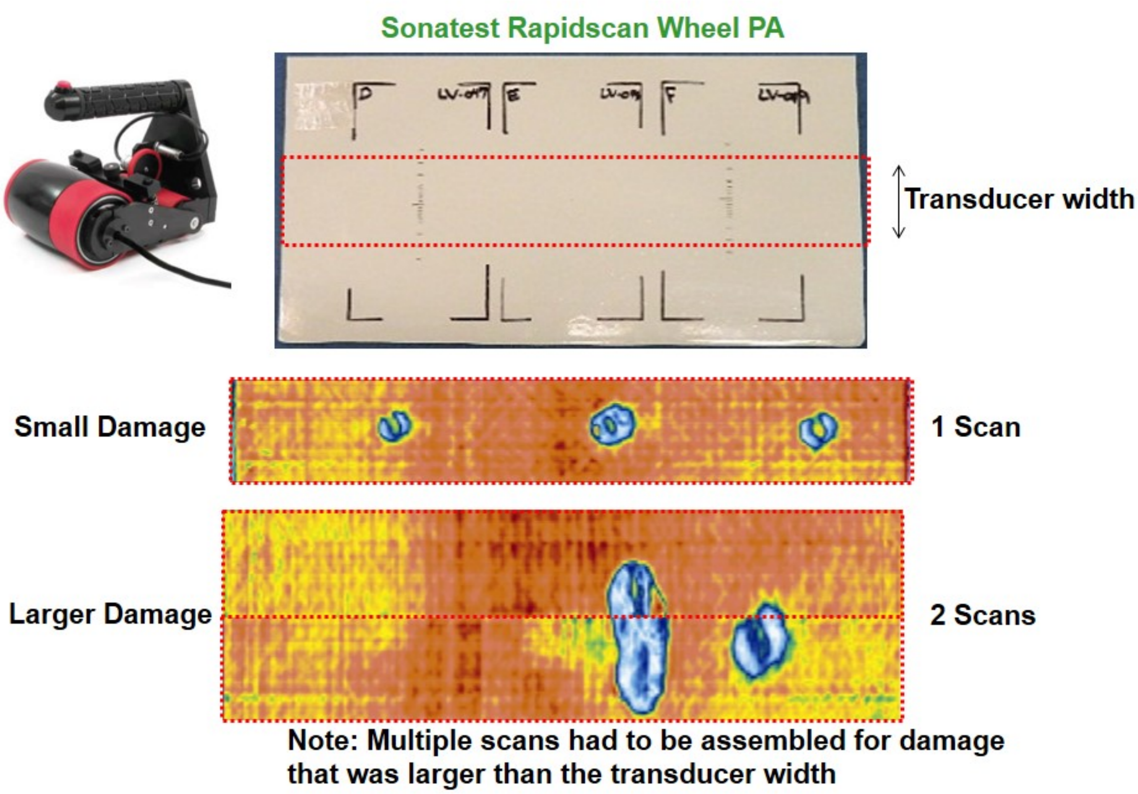


Figure 182. RapidScan phased array inspection result of impact damage

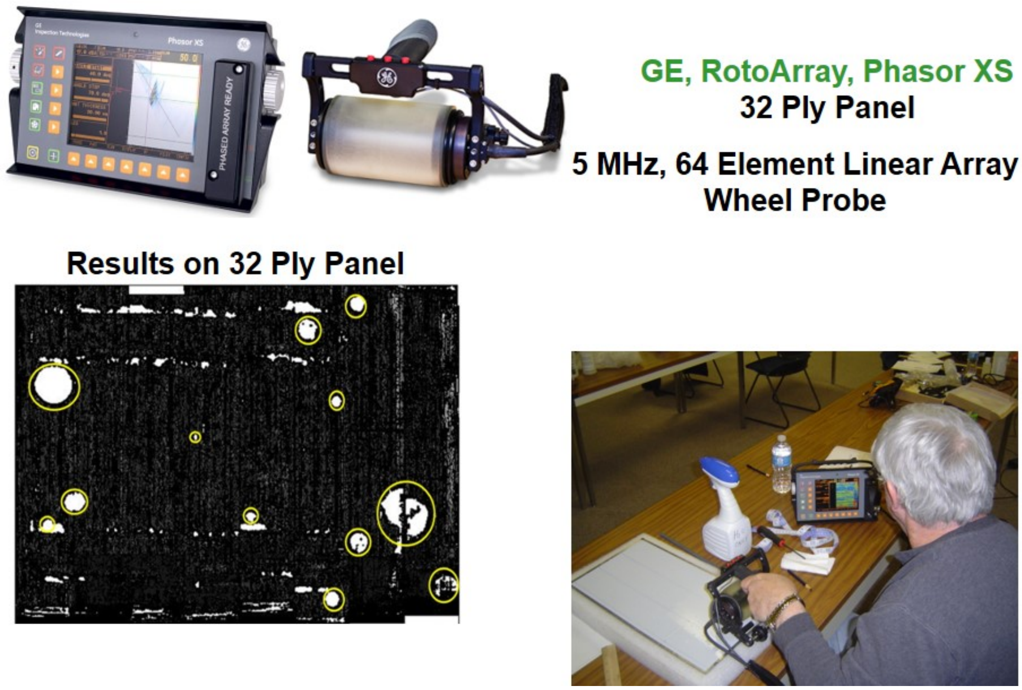


Figure 183. Example of General ElectricE RotoArray inspection result on solid laminate feedback specimen

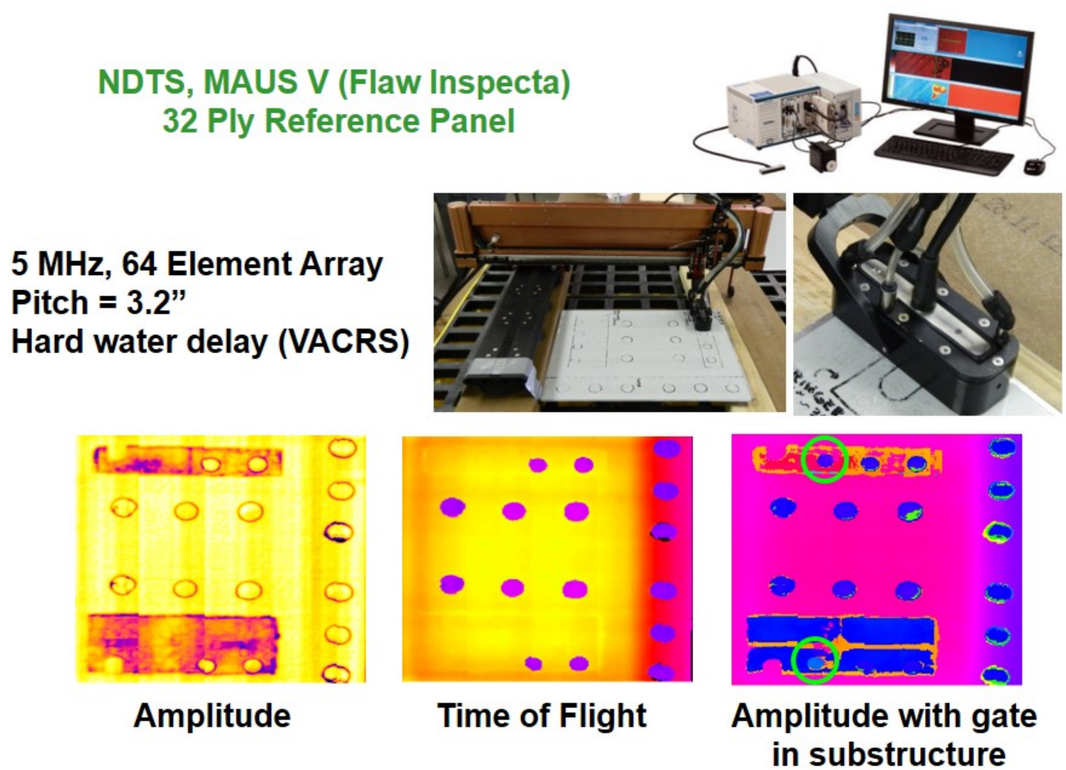


Figure 184. MAUS® V with FlawInspecta® phased array inspection result on solid laminate feedback specimen

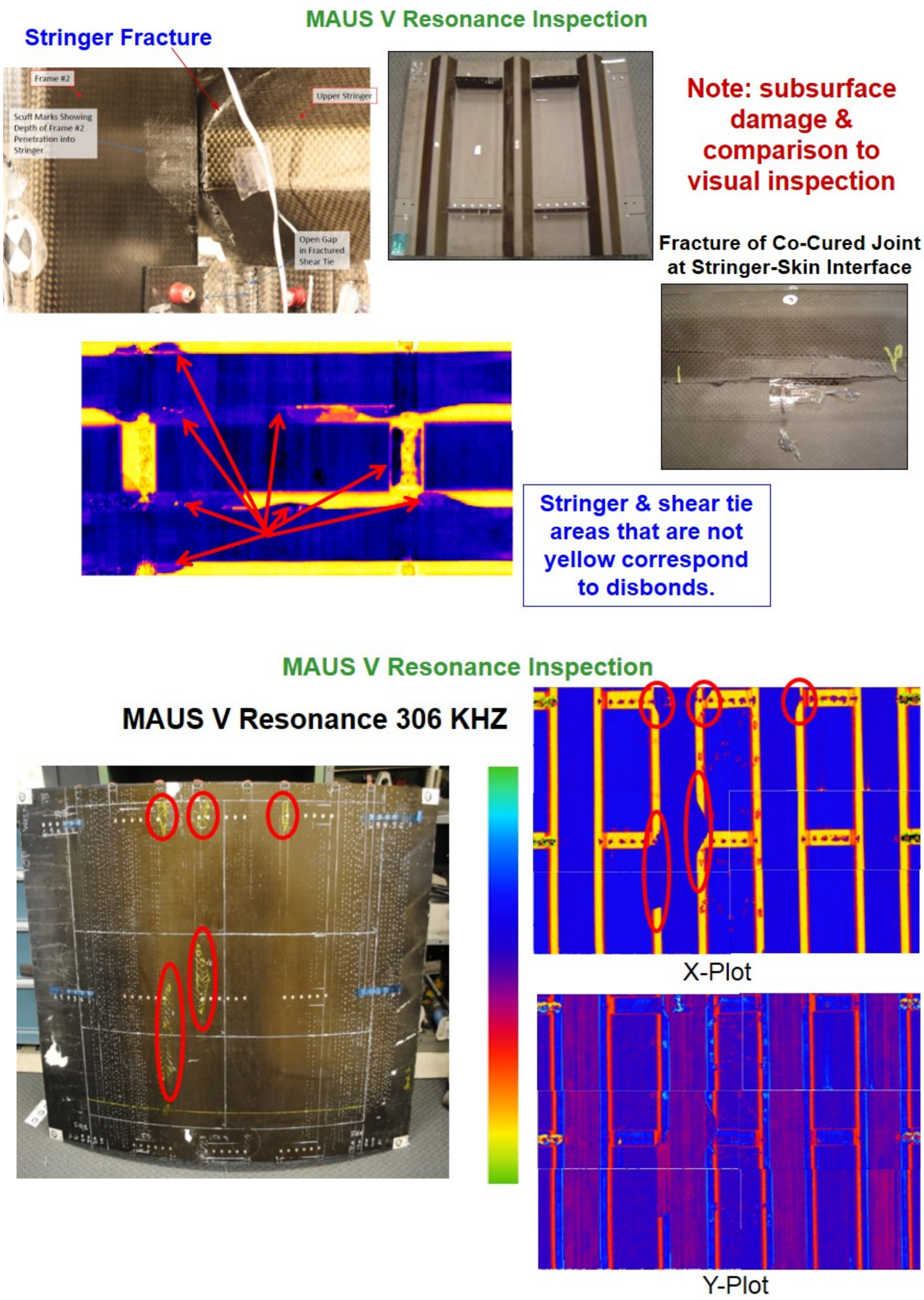


Figure 185. MAUS® V resonance inspection of full-scale impact panel with substructure

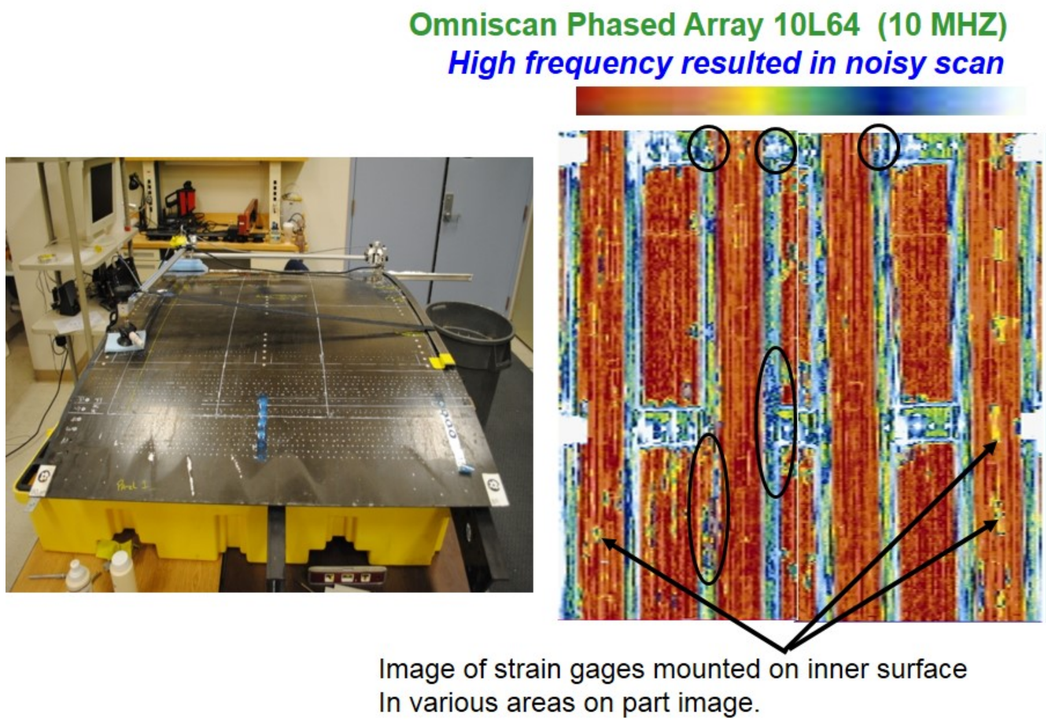


Figure 186. 10 MHz inspection of composite impact damage

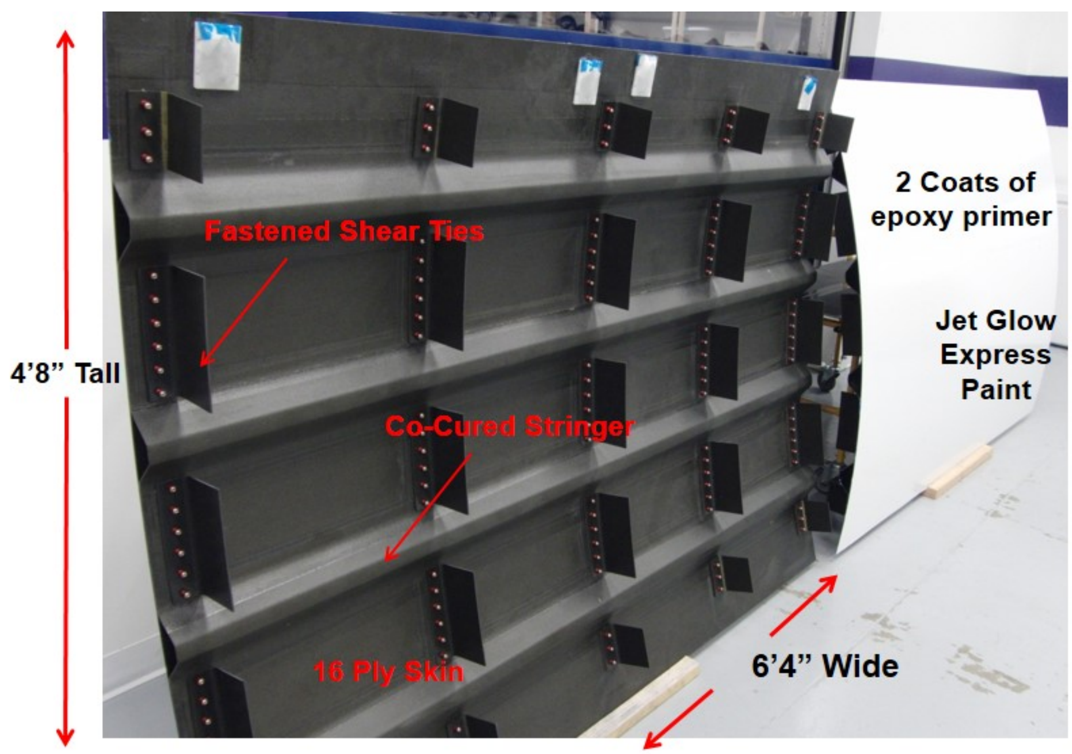


Figure 187. Example of full-scale test specimen impact tested with simulated hail

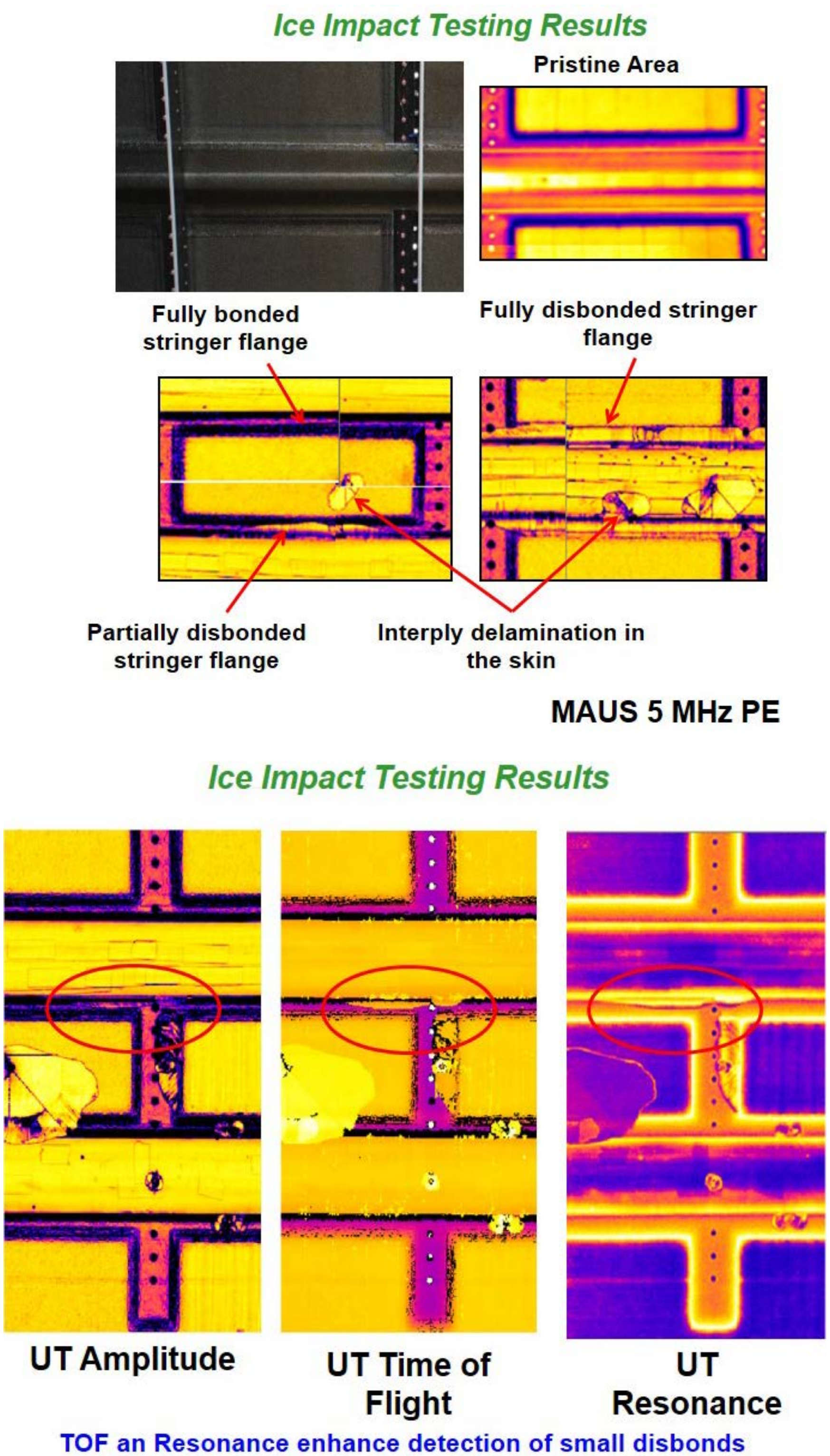


Figure 188. Example of inspection result of impact damage to structure with co-cured stringers

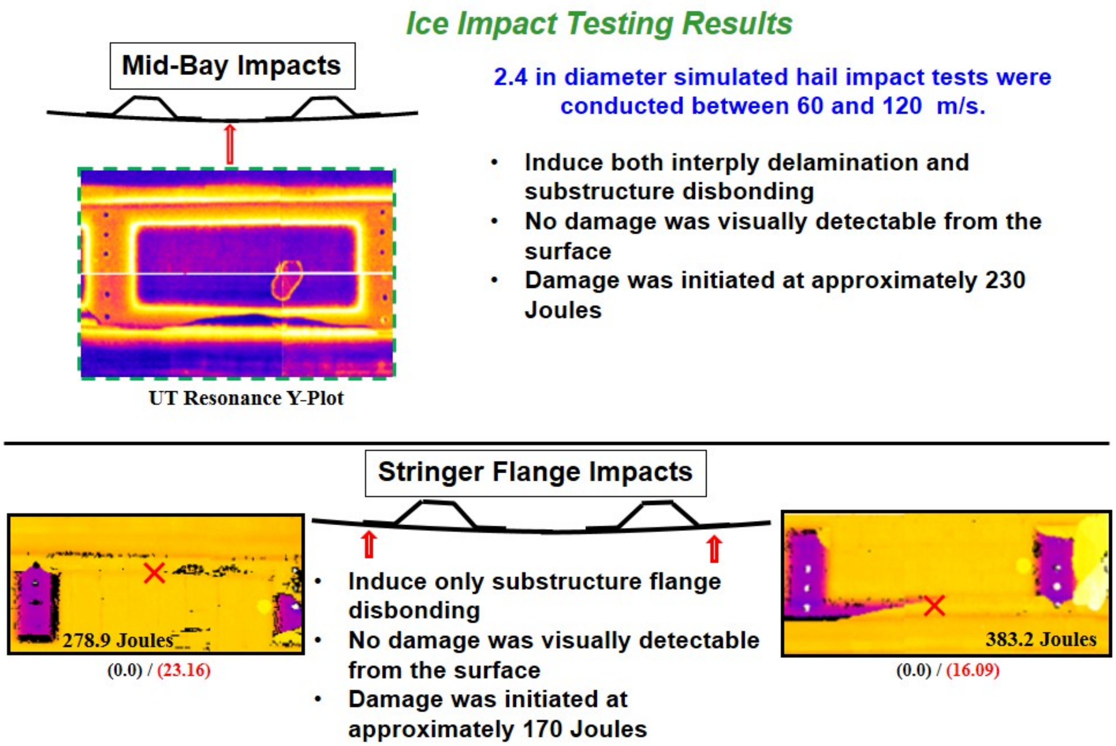


Figure 189. Results of impact damage induced at mid-bay and stringer flange

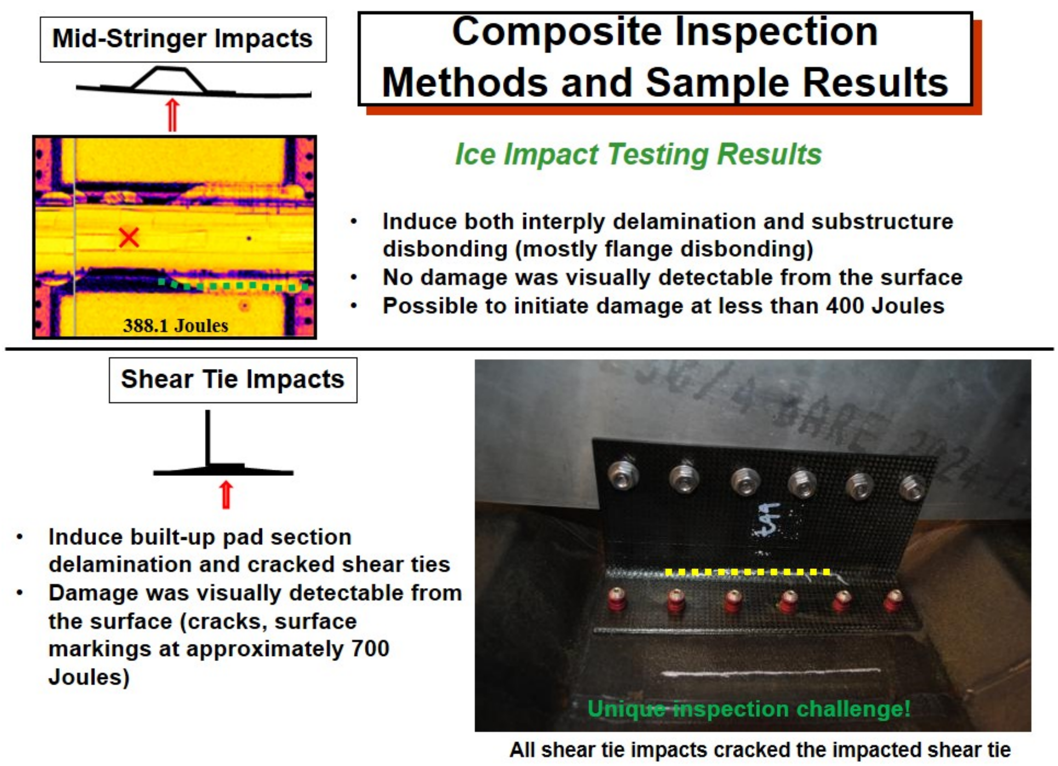


Figure 190. Results of impact damage induced at shear tie flange

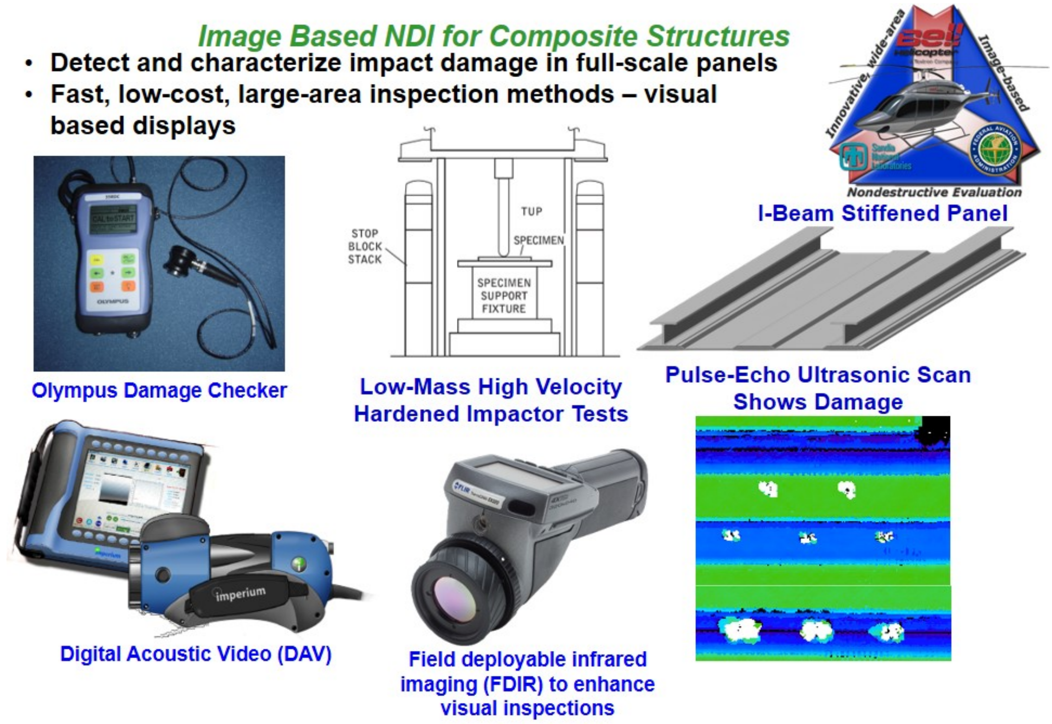


Figure 191. Image based NDI for composites

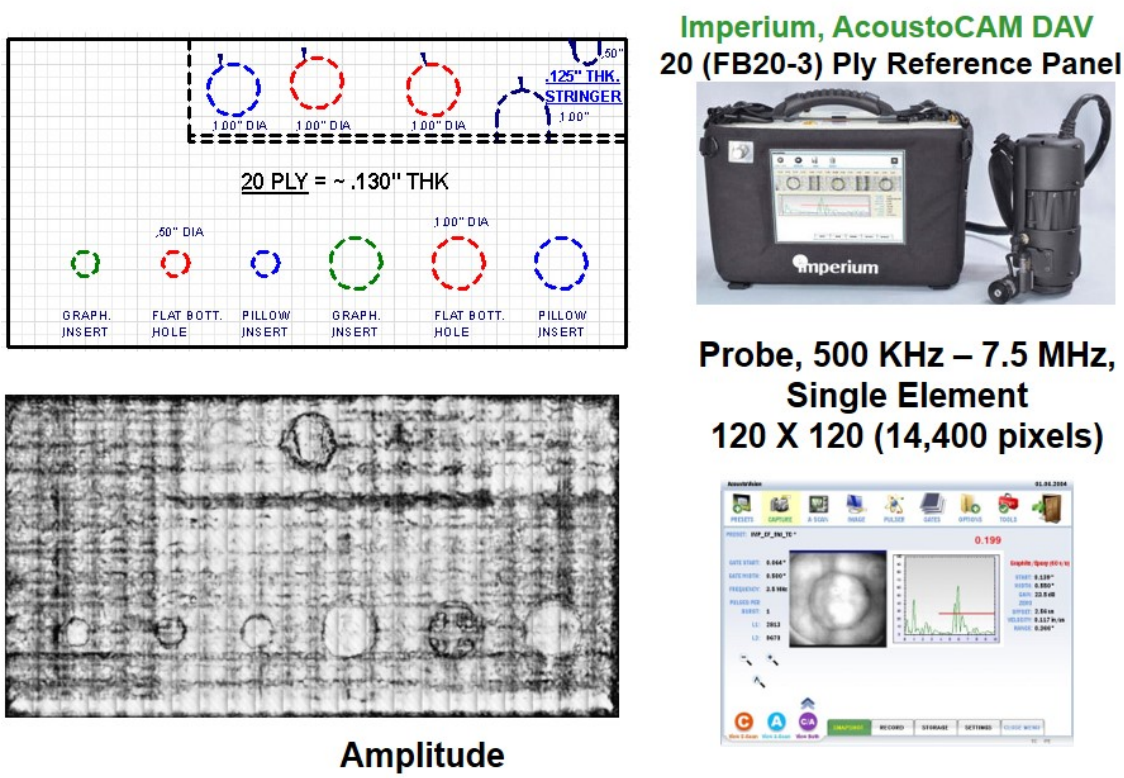


Figure 192. Sample of AcoustoCam™ inspection of solid laminate feedback specimen

6. MODULE 4: LESSONS LEARNED FROM OPERATION OF COMPOSITE AIRCRAFT

This chapter contains all lecture materials associated with Module 4: Special Cases—Challenges and Lessons Learned. Figures 193–226 encompass the lecture materials used to relay information gathered from operators of composite aircraft and the procedures they use to address maintenance issues or damage discovered during the operation of these aircraft. This experience from various airlines operating composite aircraft can be used to aid other airlines as they deal with similar or related composite maintenance issues. In general, a lesson learned is knowledge or understanding gained by experience. The experience may be positive, as in a successful composite inspection or repair process, or negative, as in a mishap or failure. A lesson is always significant in that it has a real impact on operations; valid in that is factually and technically correct; and applicable in that it identifies a specific design, process, or decision that reduces or eliminates the potential for failures and mishaps in the future, or reinforces a positive result.

The field examples provided by airlines operating composite aircraft include: high-energy wide-area blunt impact (HEWABI) damage, thermal damage from fire or engines, lightning strike, ground-handling impacts, inspection signal complexities, and challenges produced by complex structural configurations. The case studies include a description of the problem encountered, methods used to detect the damage and its severity/size, the repair process, and the outcome of the entire maintenance action including post-repair NDI, if necessary. The case history examples also highlight the type of damage that can arise from operation and repair procedures (e.g., porosity, disbonds). Also, it can be seen that pulse-echo ultrasonics, primarily deployed in C-scan and phased-array equipment, is the method of choice for conducting all composite inspections. This further validates the importance of the Composite Inspector Training Class. Other topics covered include:

- Problems created by the presence of fasteners to hold interfaces tight and mask the presence of delaminations
- Value of repetitive post-repair inspections
- Use of advanced NDI data analysis systems to quantify porosity levels
- Inspection challenges with hidden, subsurface impact damage
- Impact damage and possibilities for substructure (e.g., stringers, frames) disbanding from skins
- Proper setup and use of TCG lines
- Value of C-scans in located stiffeners and back-up structure
- Considerations for both internal and external inspections

Class Definition – Special Cases - Challenges & Lessons Learned

- Robustness of composites – detecting required damage levels
- HEWABI
- Thermal damage – fire
- Lightning strike
- Repair processing (drying, FOD control, transfer of repair data to NDI)
- Signal complexities
- Geometry and structural configuration effects
- Case studies - Examples with photos, detection, outcome

Figure 193. Special cases—Challenges and lessons learned

Accidental Damage

- Ground Handling Equipment hit the fuselage skin
- Temporary repair (speed tape application) used for return to home base
- Inspection to ensure no additional damage:
    1. External Detail Visual Inspection
    2. Internal Detail Visual Inspection
    3. PE-UT A Scan inspection for Intercostal/Stiffener Attach Flange
- Inspection to determine damage size: UT C-scan
- Permanent Repair (200° F Wet layup)
- Inspection to ensure all damage was removed before repair: PE-UT C-scan
- Inspection to confirm repair condition: PE-UT C-scan
- Repair porosity check is not required due to 3 ply depth determined by NDI

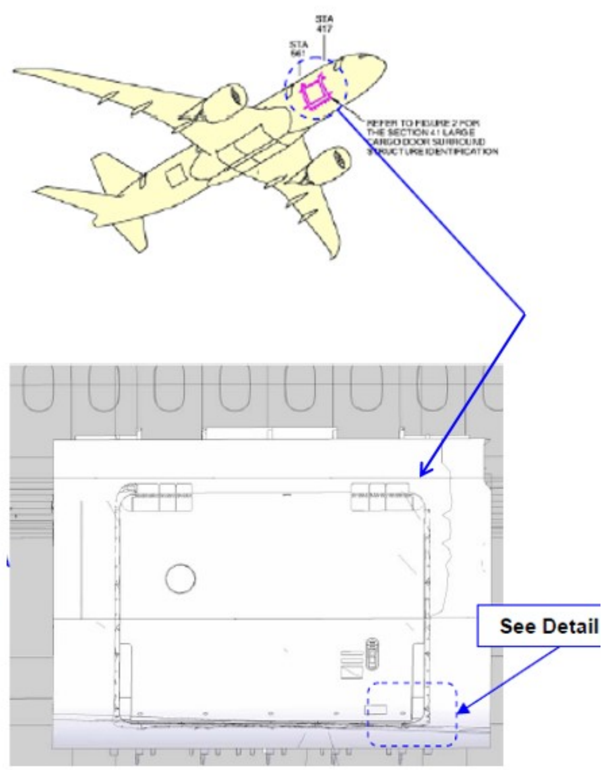


Figure 194. Cargo door surrounding structure damage

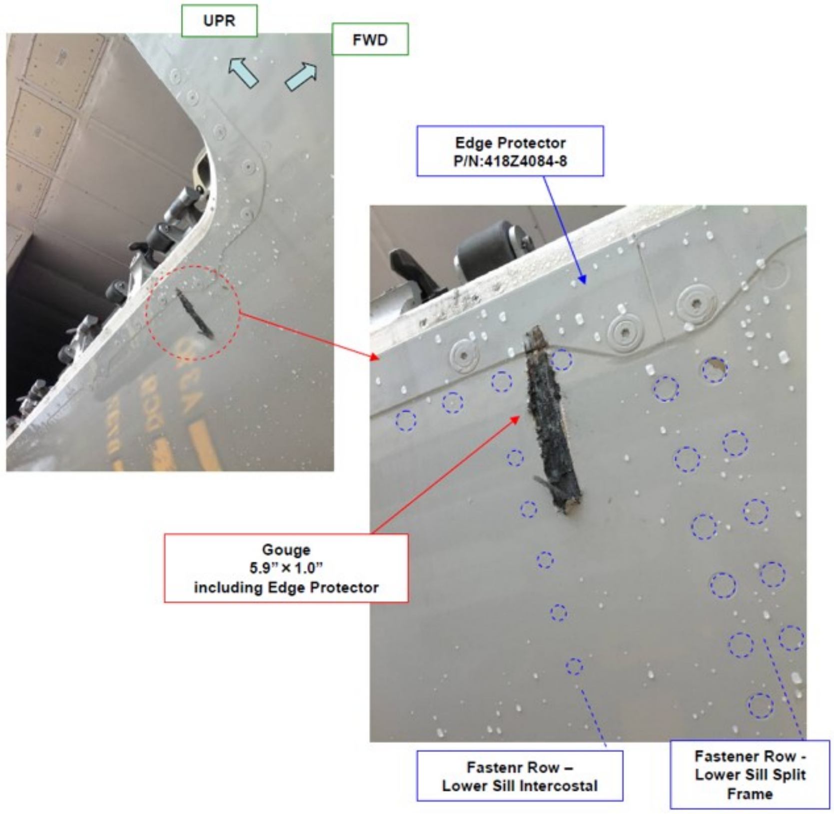


Figure 195. Accidental damage from ground handling equipment, cargo door surrounding structure damage

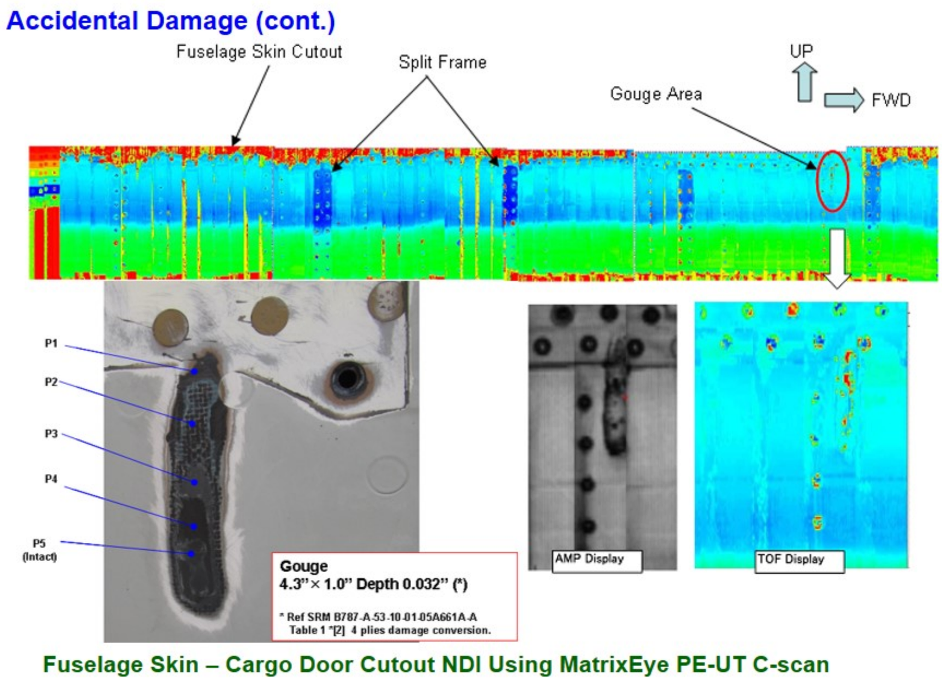


Figure 196. NDI of cargo door surrounding structure damage

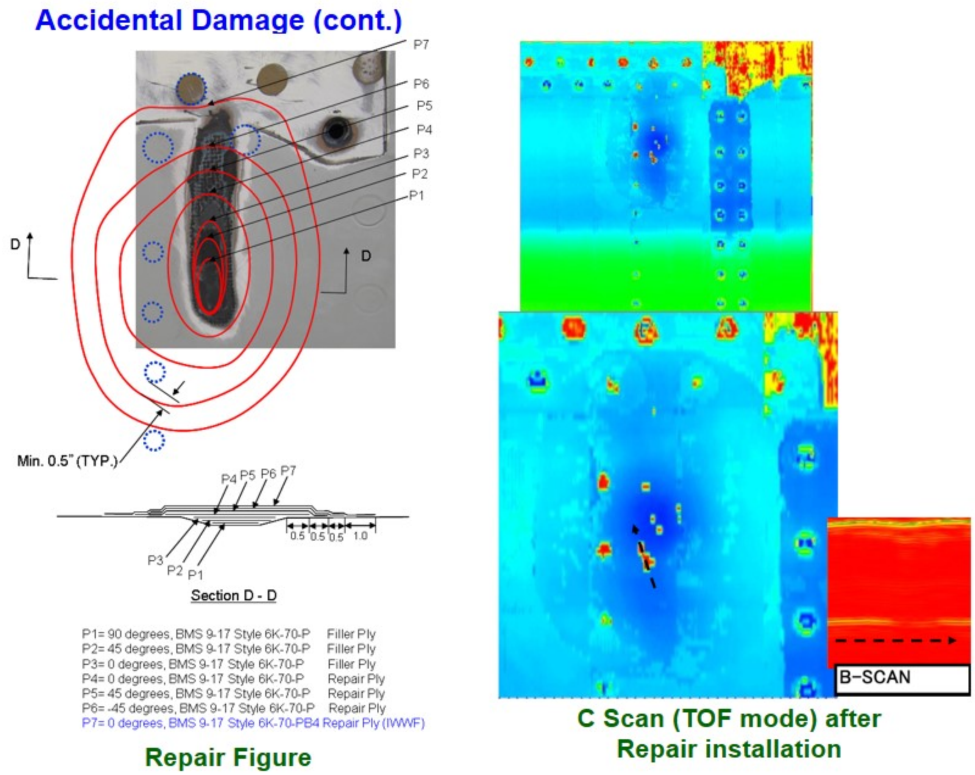


Figure 197. Repair of cargo door surrounding structure damage

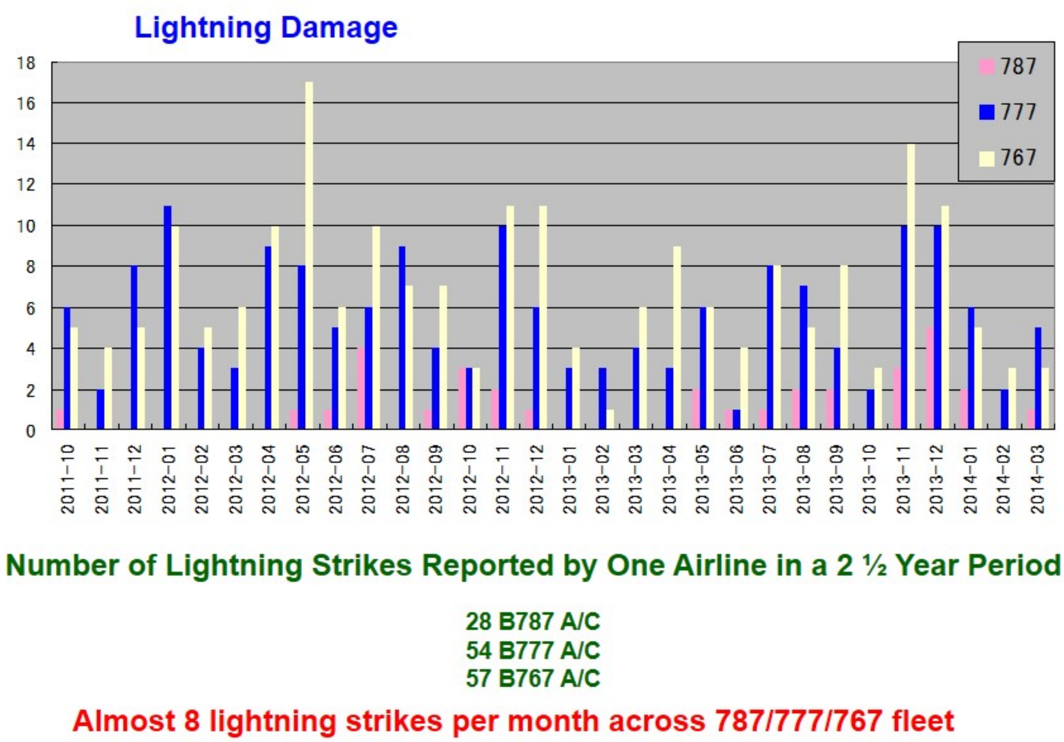


Figure 198. Aircraft lightning damage statistics

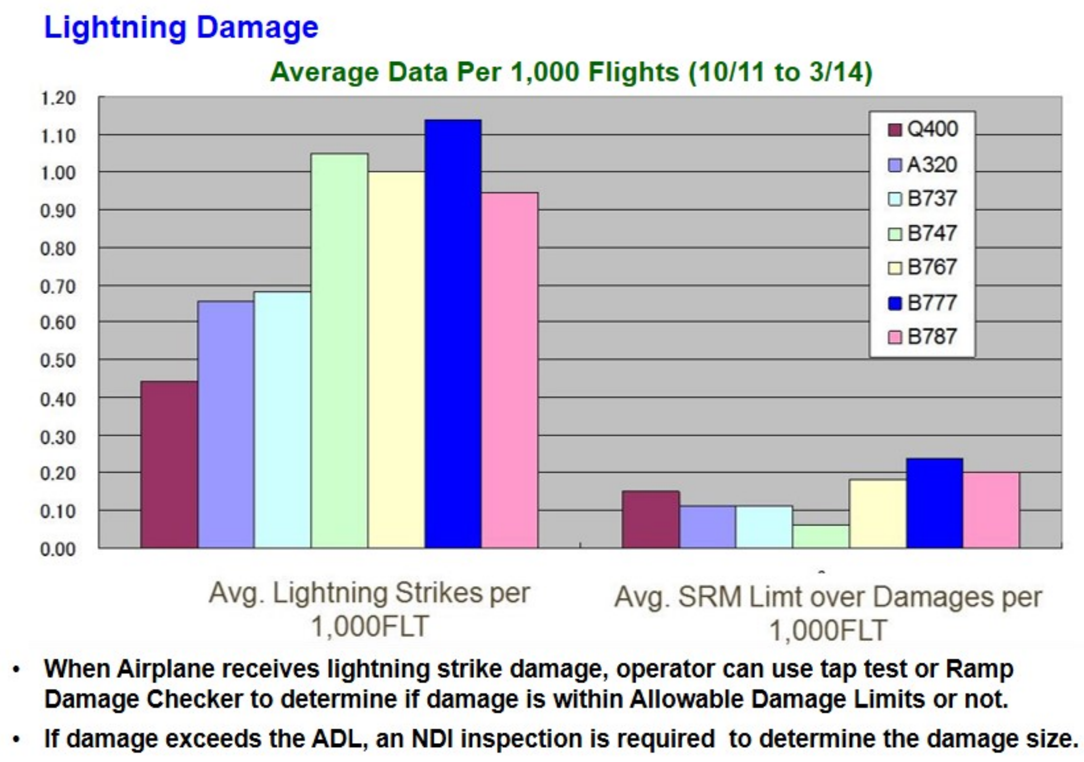


Figure 199. Aircraft lightning damage strikes per flight

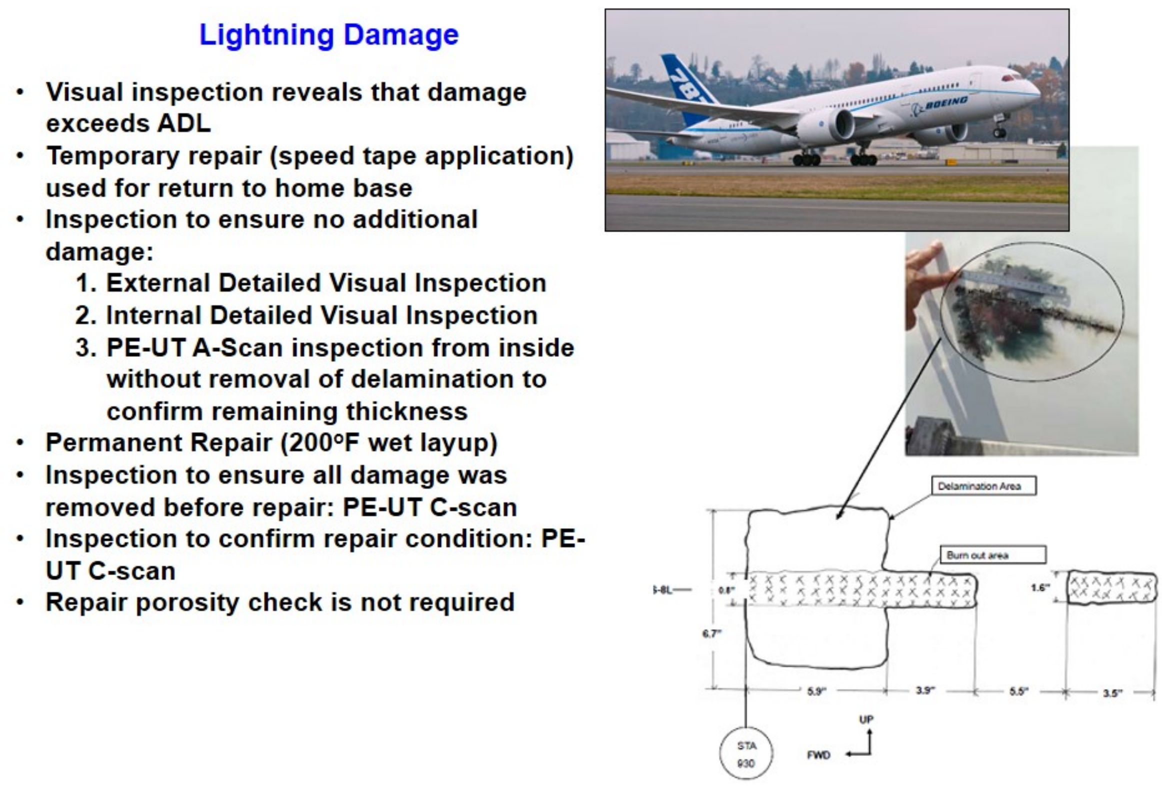
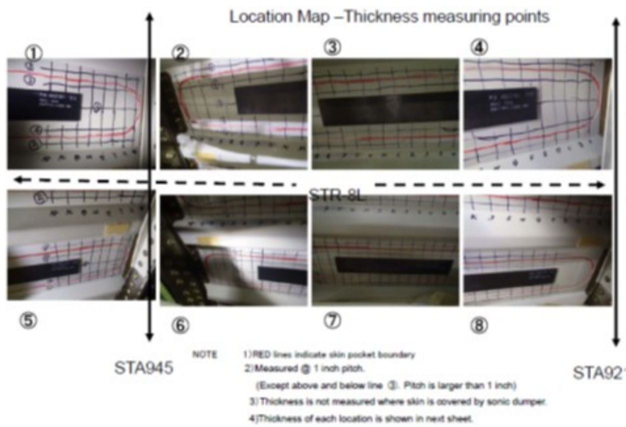


Figure 200. Lightning damage to fuselage

Lightning Damage (cont.)

Why Pulse Echo Ultrasonic A-scan was performed from inside the aircraft -

- **Airline was not able to transport UT C-scan system in a timely manner**
- **Airline was only allowed to remove the loose fiber at first**
- **In order to determine the temporary repair procedure and repair classification (major/minor repair), the airline had to measure the remaining thickness from inside.**



Unit: Inch XX: Not measured ANA-ANA-13-2

Location	AD	AC	AB	AA	Z	Y	X	W		V	U	T	S	R	Q
1	0.150	0.152	0.153	0.152	0.152	0.152	0.174	0.176	STA	0.175	0.156	0.149	0.148	0.148	0.149
2	0.083	0.079	0.083	0.096	0.096	0.097	0.116	XX	945	XX	0.095	0.093	0.093	0.094	0.093
3		XX	XX	0.085	0.084	0.084	0.094	0.115		0.117	0.092	0.085	0.085	0.085	XX
4	0.076	0.077	0.077	0.076	0.076	0.088	0.108	XX		XX	0.080	0.092	0.075	0.090	0.076
5	0.147	0.146	0.148	0.148	0.148	0.148	0.150	0.170		0.168	0.138	0.147	0.147	0.147	0.147
	AFT											STRINGER - 8L			
6	0.149	0.148	0.149	0.148	0.148	0.149	0.172	0.171		0.169	0.149	0.149	0.148	0.149	0.149
7	0.092	0.091	0.091	0.090	0.091	0.090	0.112	XX		XX	0.094	0.077	0.076	0.077	0.078
8	XX	XX	0.082	0.082	0.082	0.082	0.108	0.112		0.100	0.092	0.070	0.069	0.090	0.090
9	0.080	0.080	0.090	0.089	0.089	0.089	0.111	XX		XX	0.080	0.076	0.075	0.074	0.075
10	0.148	0.149	0.148	0.148	0.153	0.159	0.169	0.167		0.172	0.140	0.149	0.149	0.147	0.149

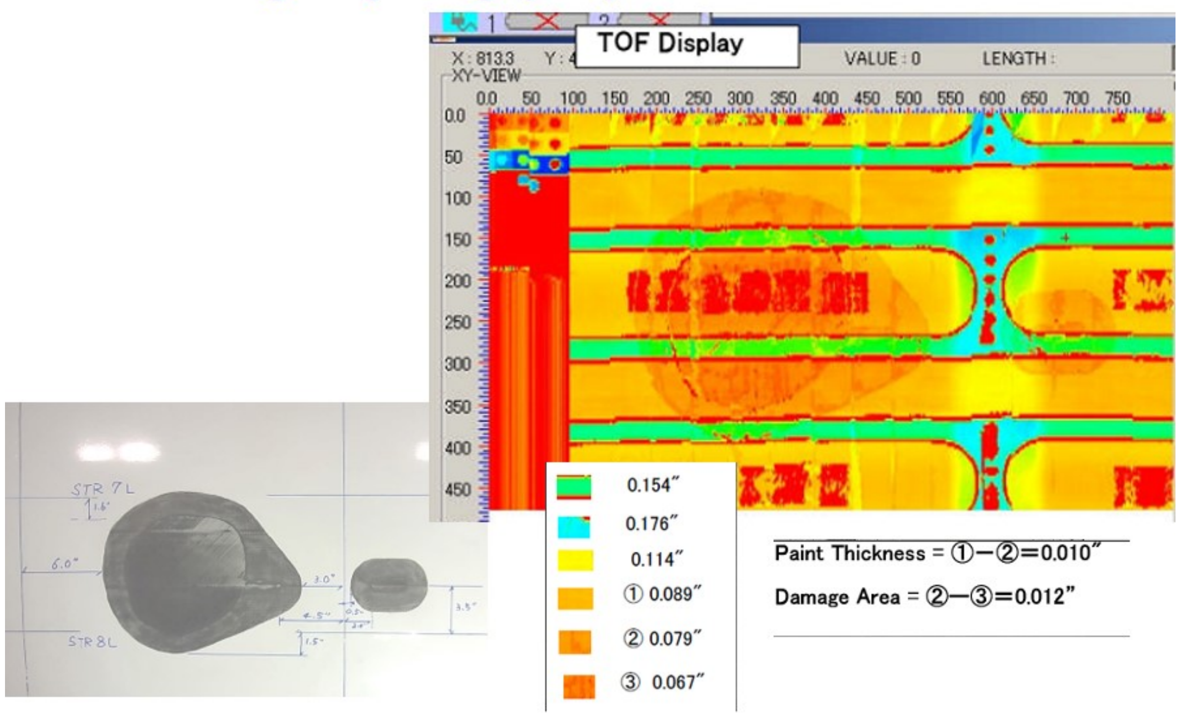
P	O	N	M	L	K	J	I	H	G	F	E	D	C	B	A		Location
0.149	0.124	0.124	0.124	0.127	0.127	0.128	0.151	0.151	0.150	0.151	0.150	0.151	0.151	0.156	0.174	STA	1
0.093	0.062	0.061	0.063	0.066	0.064	0.066	0.066	0.066	0.096	0.096	0.095	0.095	0.093	0.102	XX	921	2
XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	0.086	0.086	0.085	0.090	0.111		3
0.069	0.067	0.068	0.067	0.070	0.068	0.067	0.056	0.070	0.078	0.093	0.075	0.087	0.075	0.098	XX		4
0.147	0.138	0.127	0.127	0.126	0.126	0.128	0.127	0.121	0.149	0.148	0.149	0.147	0.149	0.152	0.169		5
							STRINGER - 8L										
0.148	0.149	0.148	0.149	0.148	0.148	0.149	0.150	0.150	0.150	0.150	0.150	XX	XX	XX	XX		6
0.077	0.079	0.091	0.092	0.094	0.093	0.090	0.090	0.092	0.092	0.090	0.091	XX	XX	XX	XX		7
XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX		8
0.076	0.075	0.075	0.075	0.075	0.074	0.075	0.075	0.077	0.078	0.077	0.076	XX	XX	XX	XX		9
0.149	0.149	0.150	0.149	0.147	0.148	0.147	0.148	0.148	0.148	0.149	0.148	XX	XX	XX	XX		10

Measurement Data

Measurement from Inside the Aircraft

Figure 201. Lightning damage assessment

Lightning Damage (cont.)



C-Scan Image Produced After Removal of the Damage

Figure 202. Lightning damage inspection after scarfing

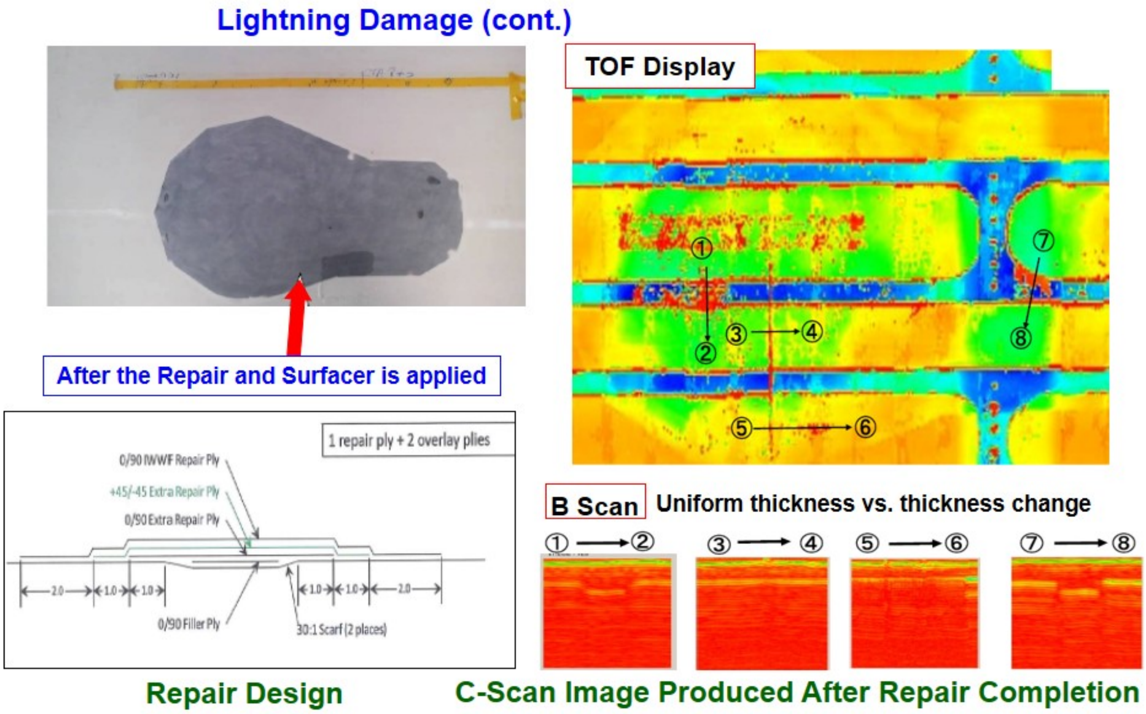


Figure 203. Lightning damage inspection after repair

Figure 204. Lightning damage next to window cutout

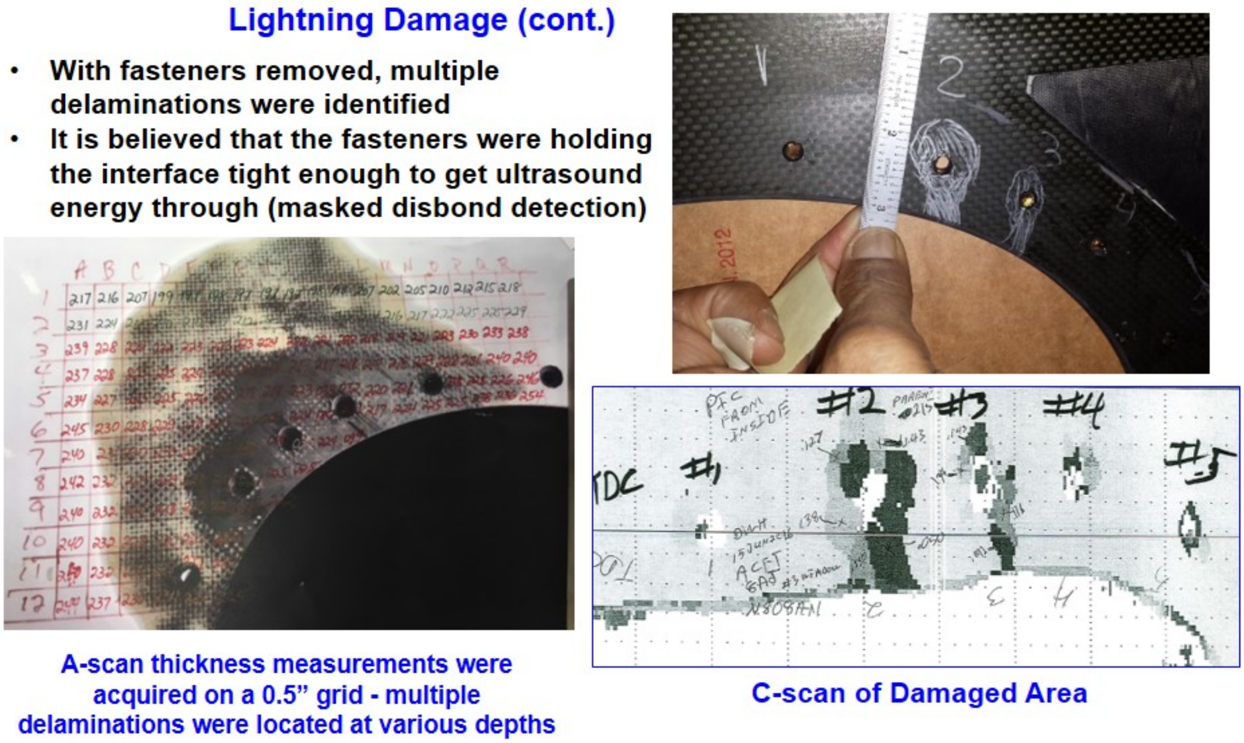


Figure 205. Inspection of lightning damage at window

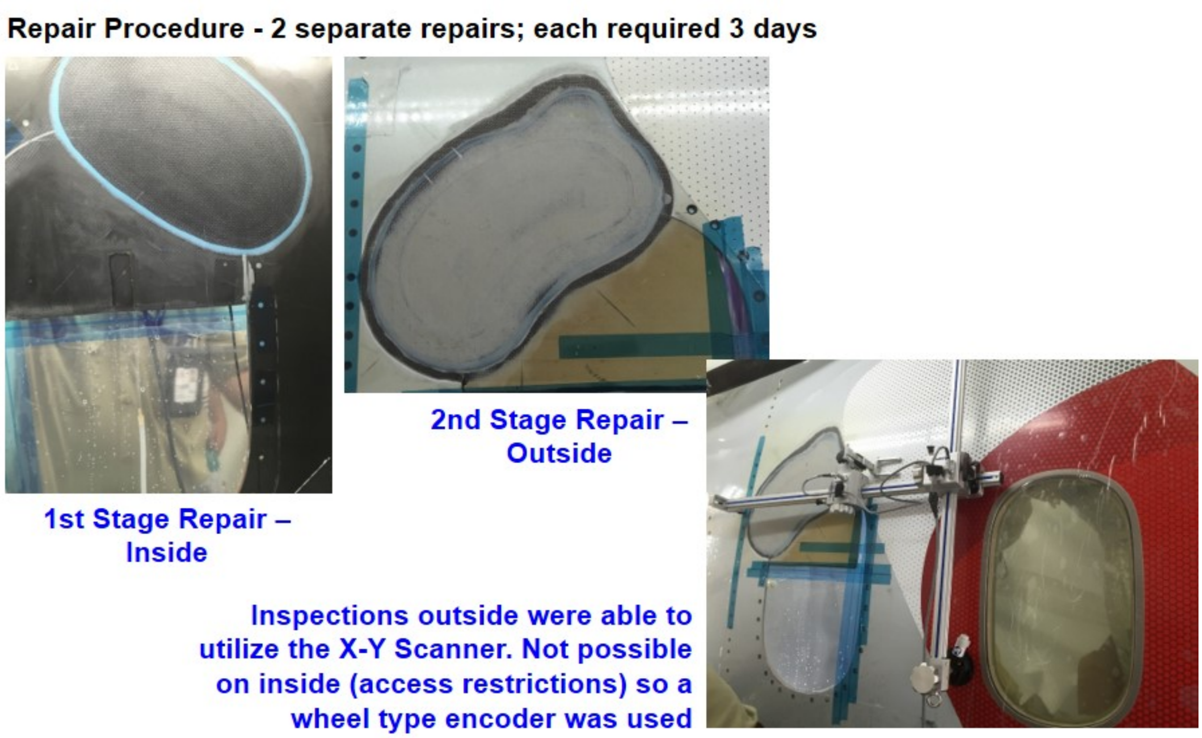


Figure 206. Repair of lightning damage at window

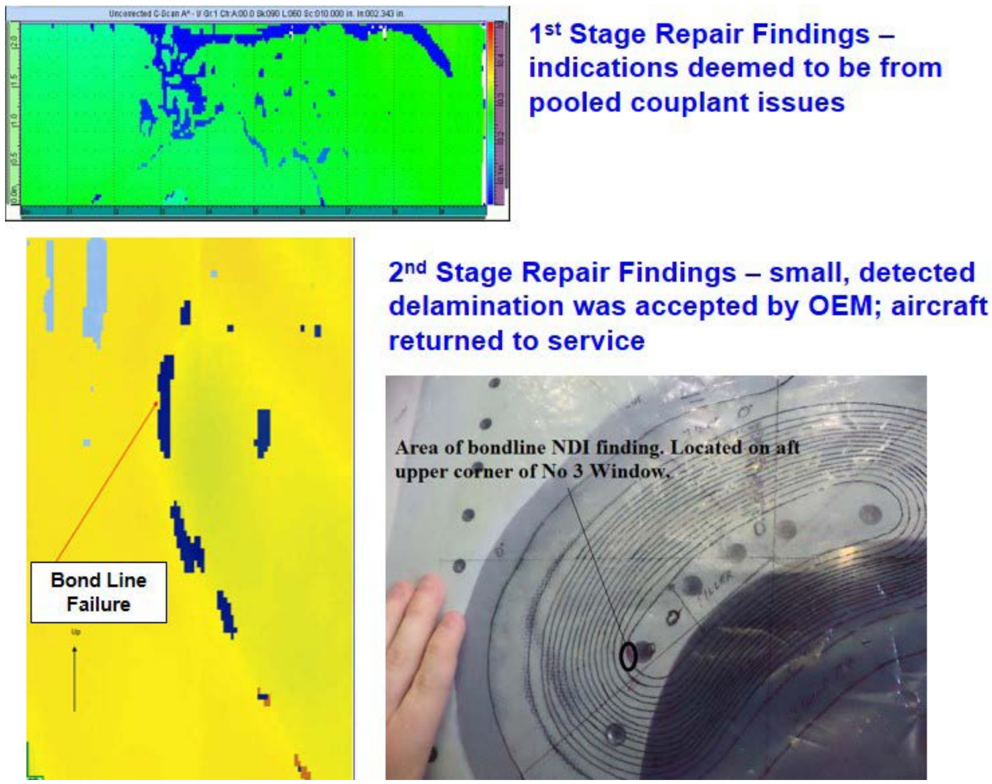


Figure 207. Post-repair NDI of lightning damage at window

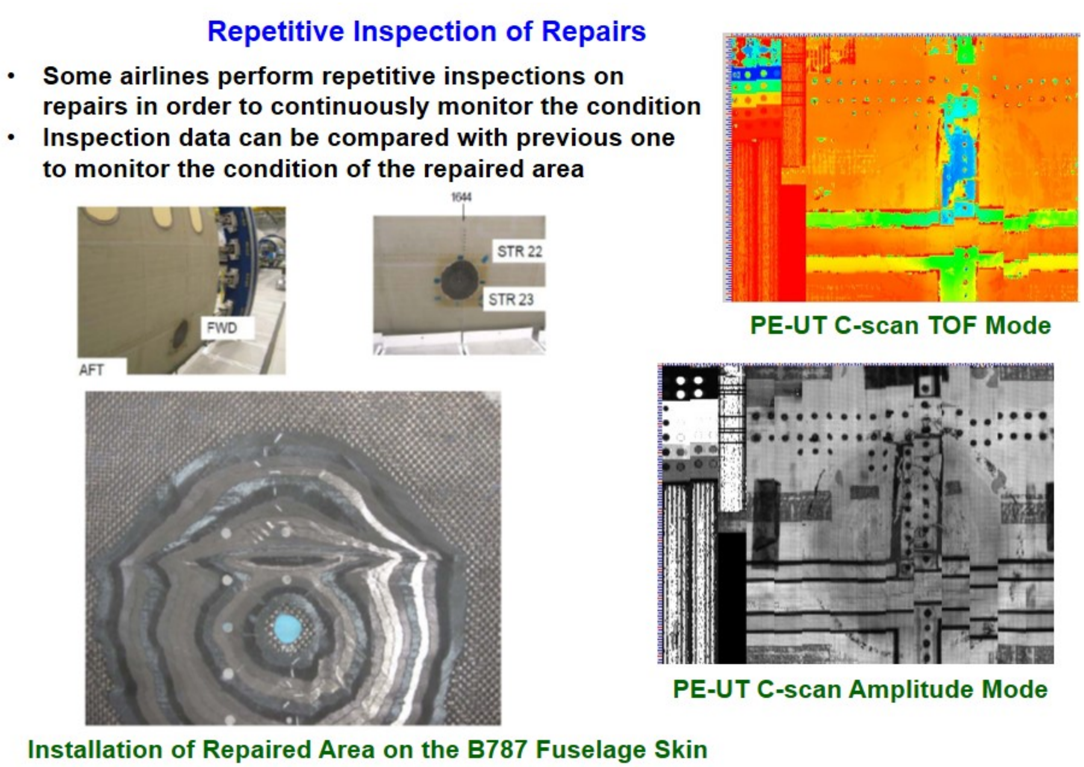


Figure 208. Repetitive inspection of repairs

Post Repair Inspection

1. Interply or bondline delaminations.
2. Porosity in the repair plies
   NOTE: A porosity analysis is only necessary for repairs with four or more repair plies.
3. Porosity in the bondlines
4. Inclusions (foreign material) in the repair.

Figure 209. Post-repair inspection

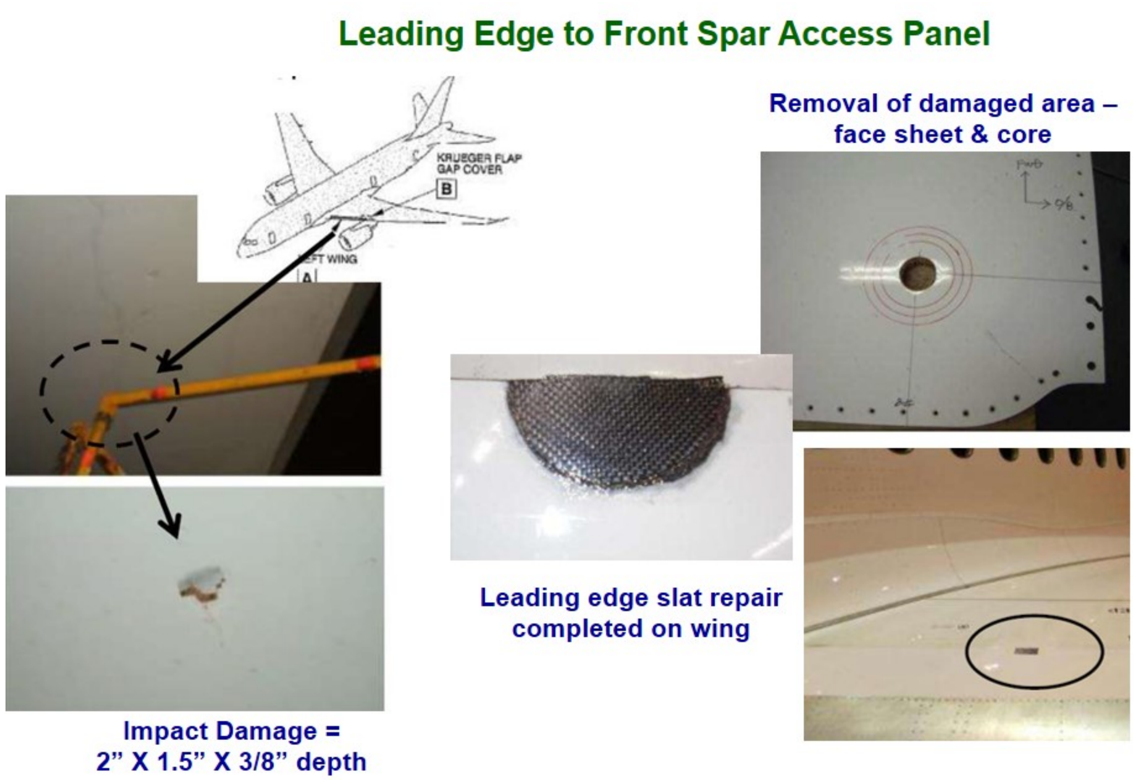


Figure 210. Accidental damage from ground handling equipment

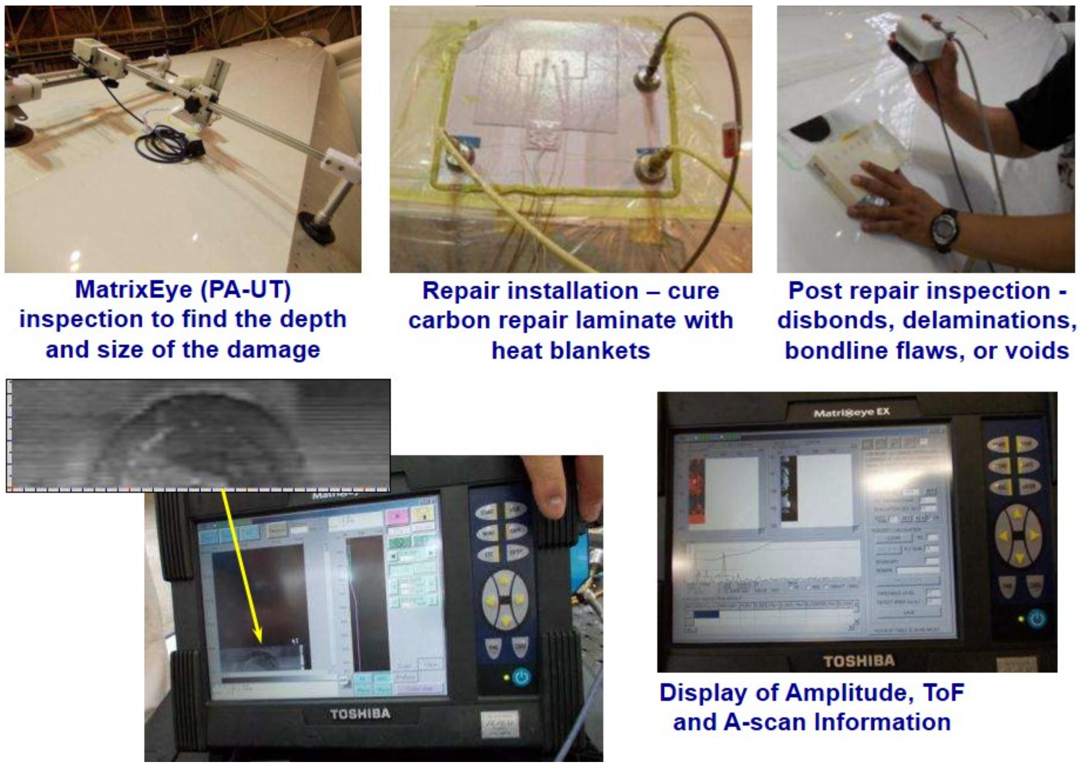


Figure 211. Inspection of leading-edge slat

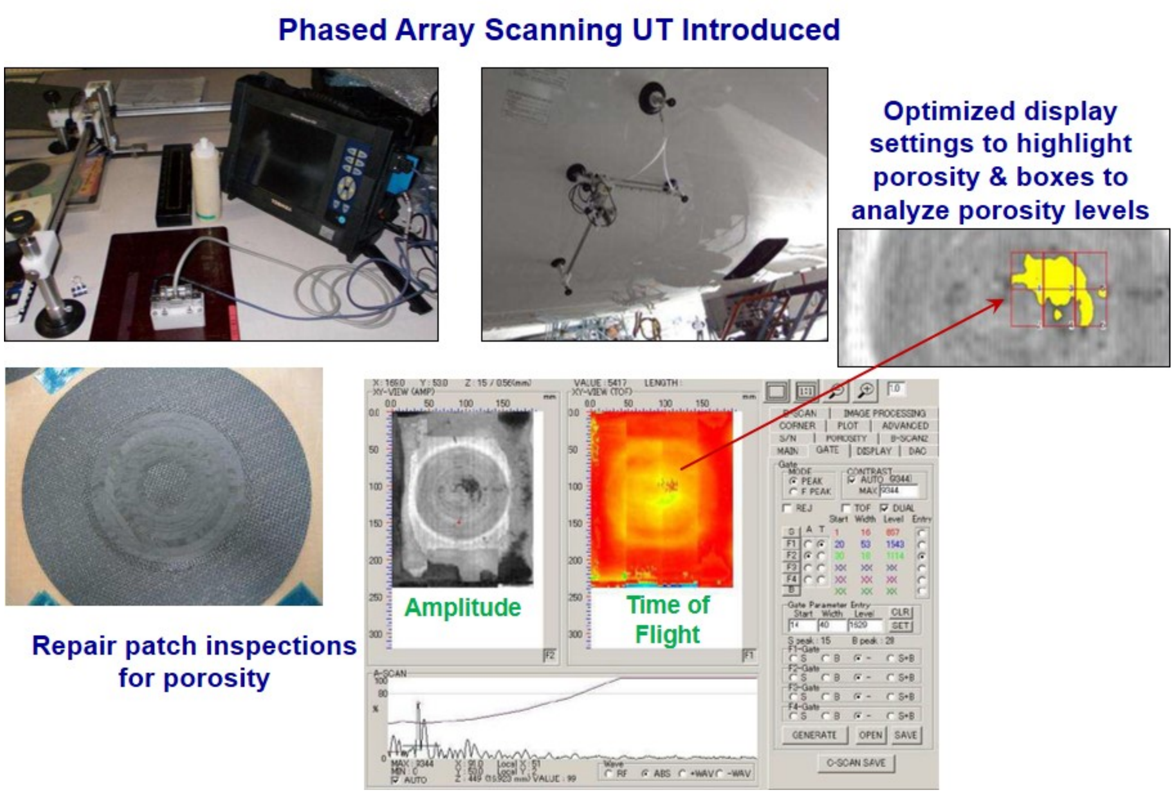


Figure 212. Post-repair inspection—Addressing challenging inspections

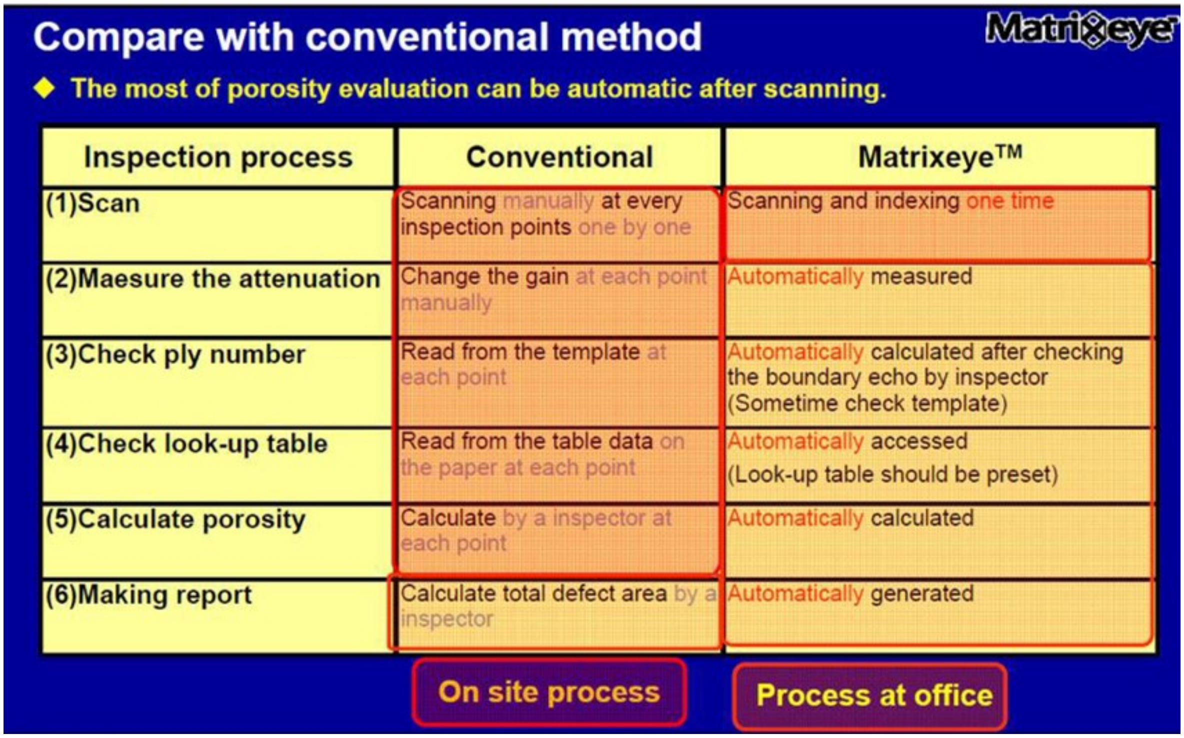


Figure 213. Advantages of scanning pulse echo ultrasonic transmission (PE UT)

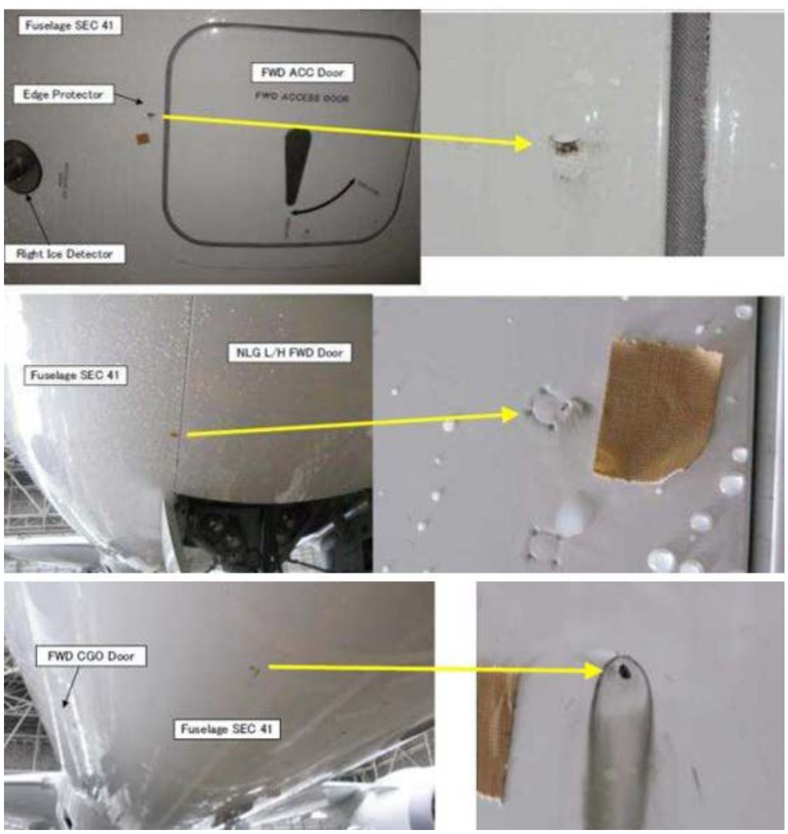


Figure 214. Field experience—Lightning strike adjacent to forward access door

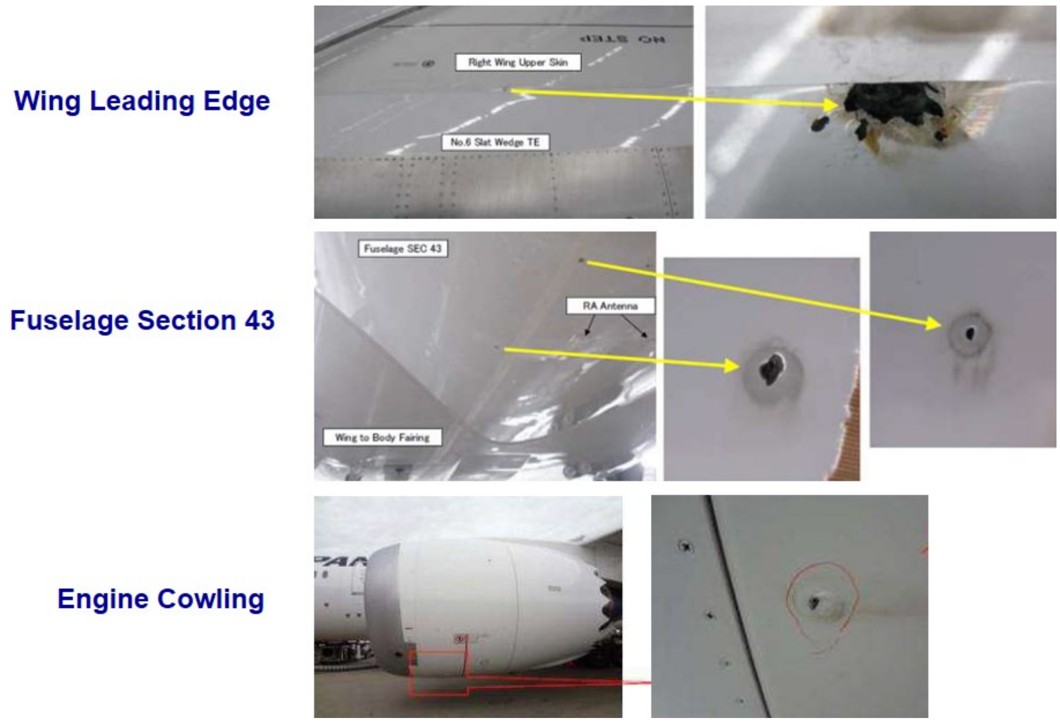


Figure 215. Field experience—Lightning strike examples

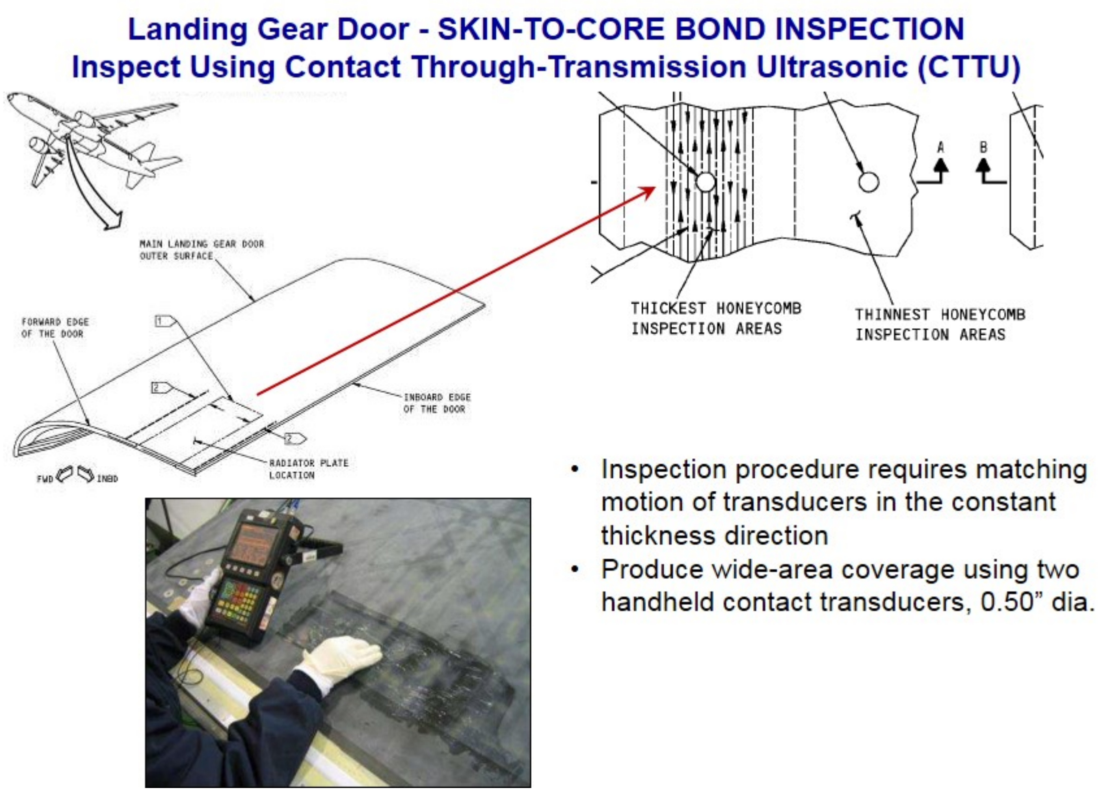


Figure 216. Sample challenging inspection—Landing gear door

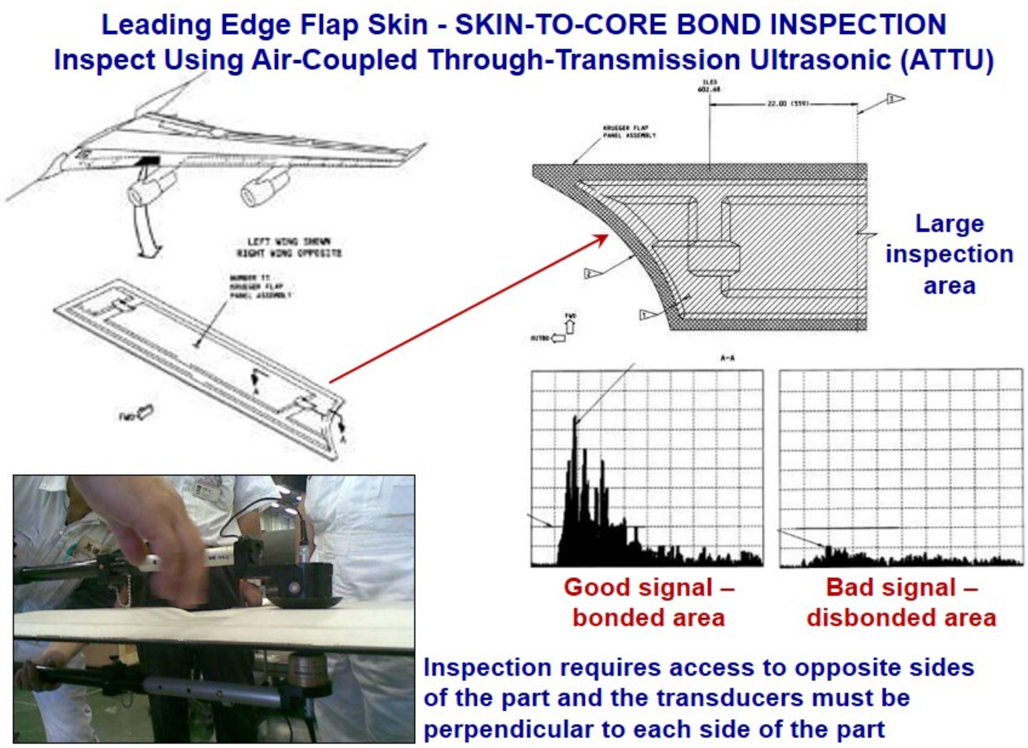


Figure 217. Sample challenging inspections—Leading-edge flap skin

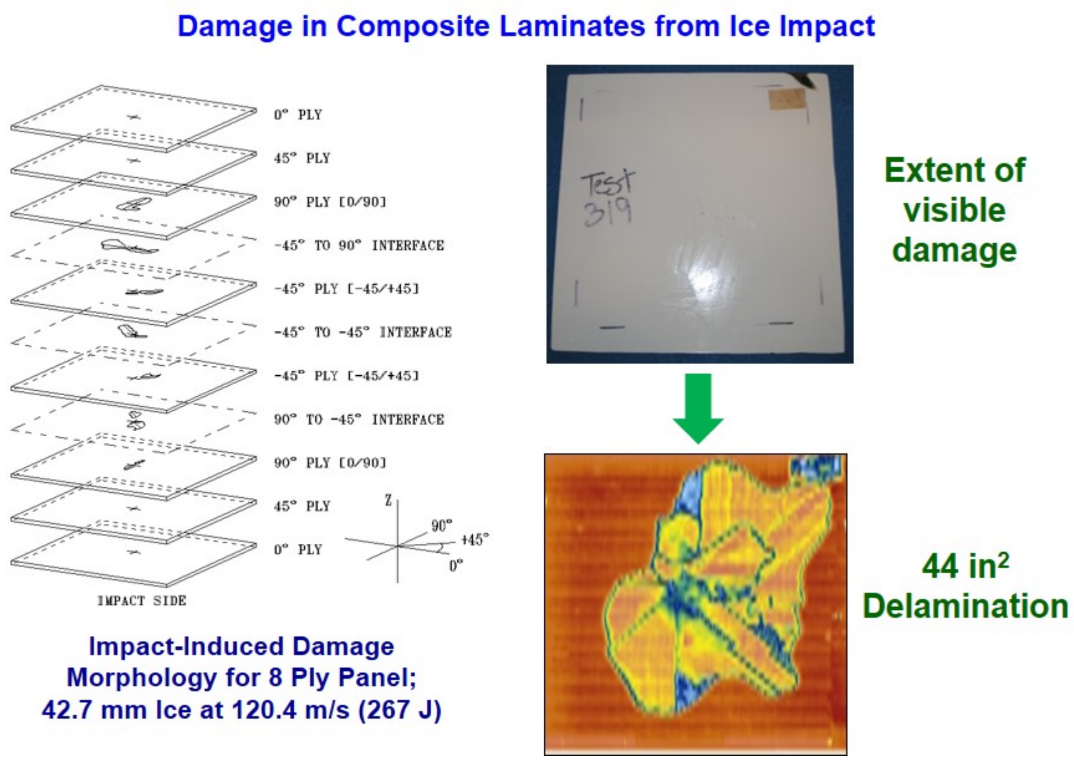


Figure 218. Inspection challenge—Hidden impact damage

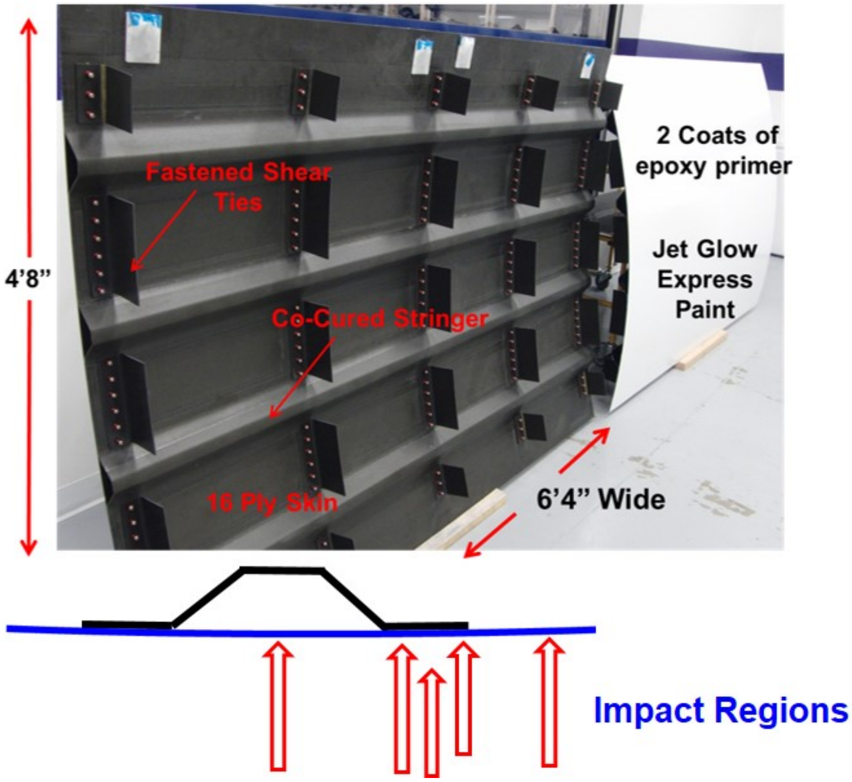


Figure 219. Composite impact damage assessment on full-scale fuselage test panels

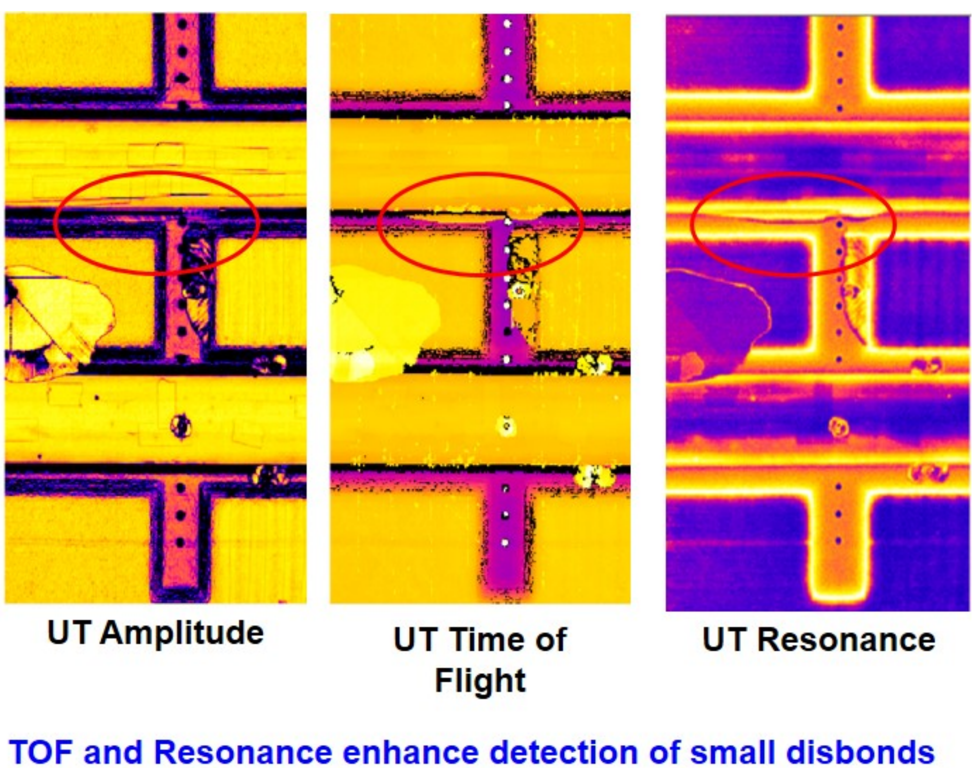


Figure 220. Comparison of NDI techniques for detecting composite impact damage

+

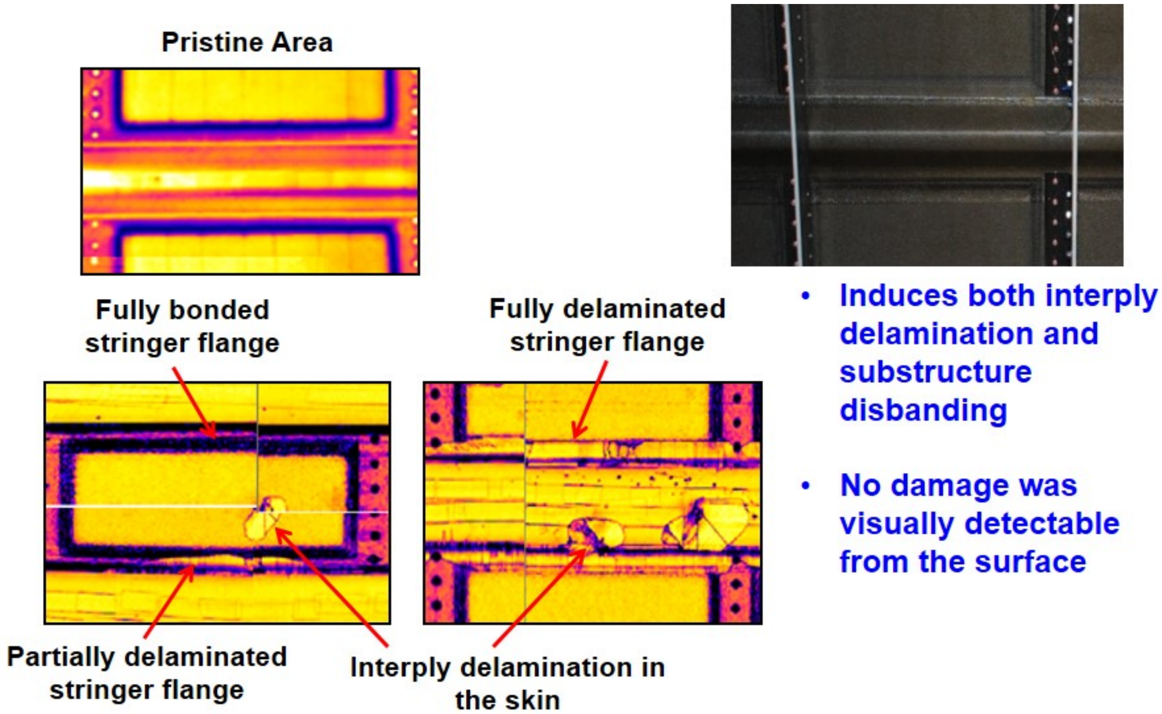


Figure 221. C-scan inspection interpretation—Impact damage

Lesson #1 – Read and Follow the Procedures

Lesson #2 – Embrace New Technology – It Can Be Helpful

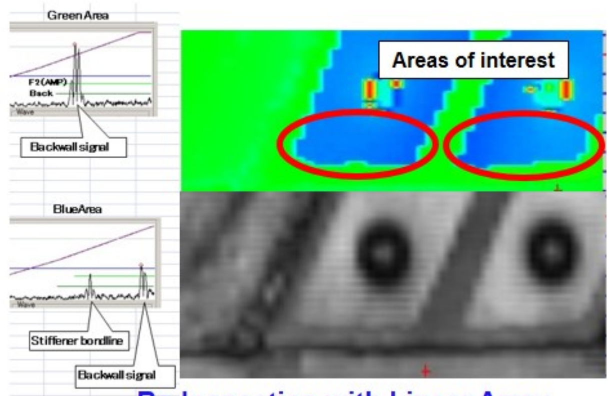
Lesson #3 – Composite Damage Tolerance is Good – NDI will Tell

Lesson #4 – Follow OEM Documentation

Figure 222. Four challenges and lessons learned to improve in-service inspections

- Inspection of thick composite structure
- Went from A-scan to C-scan roller probe
- Inspector mistakenly used saved settings in the instrument with incorrect TCG

Initial Roller Probe Linear Array C-scan
(TCG does not linearly increase with depth)



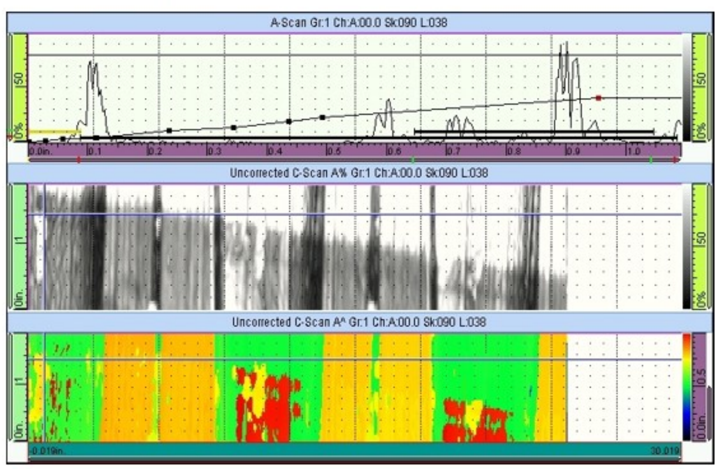
Re-Inspection with Linear Array
and Correct TGC

- Procedure made it clear to set TCG for entire inspection thickness
- "I've done this before" attitude caused the procedure to be ignored (didn't double-check TGC settings)
- Correct TCG could have prevented false indications and all of the associated follow-up

Figure 223. Lesson 1—Read and follow the procedures

- Challenging to locate stiffeners & back-up structure on wing
- Large areas are difficult to map

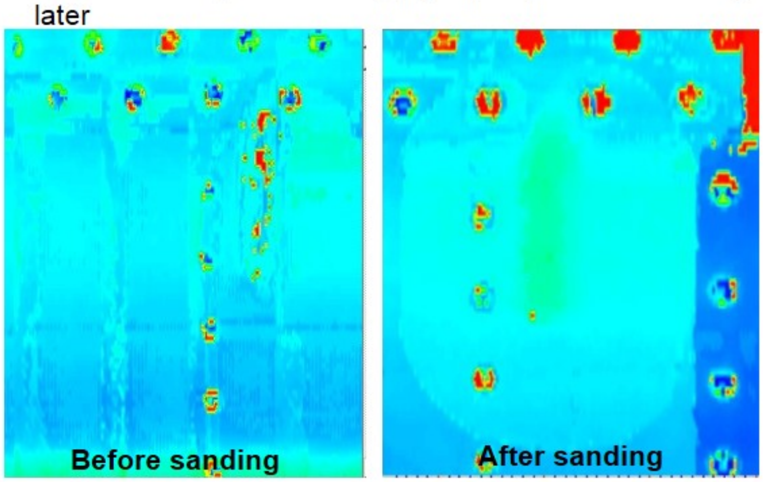
2-D data display
shows substructure
and aids flaw
detection



- A-scan is difficult & time consuming
- C-scan roller probe is a simpler & faster option

Figure 224. Lesson 2—Embrace new technology, it can be helpful

- Ramp Damage - forward cargo door area was hit during loading, 5” X 1” damage to the fuselage skin. NDI determined that the depth of damage was limited to 3 plies and that there was no underlying delamination.
- Temporary Repair: Based on NDI insight, evaluation by Boeing determined Speed Tape acceptable
- Permanent Repair: Overlay ply repair performed 13 days later



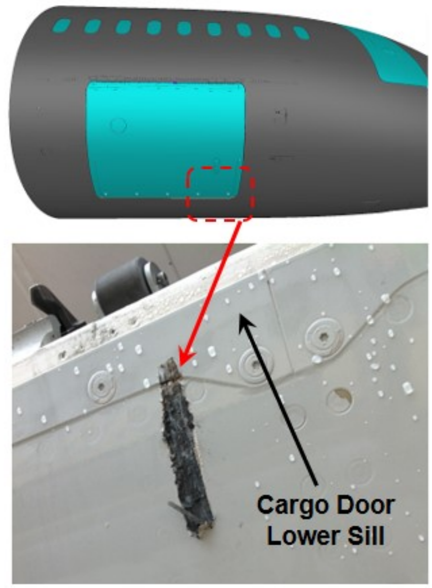
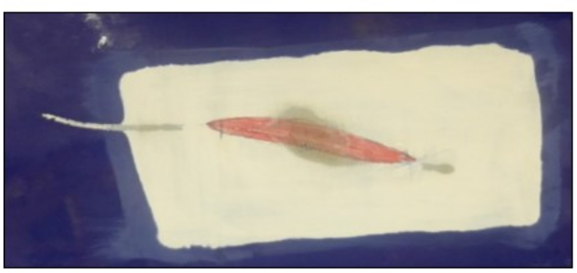
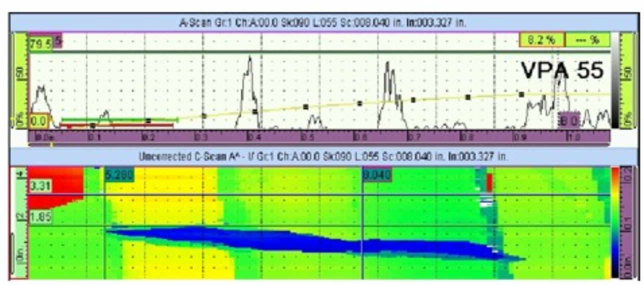
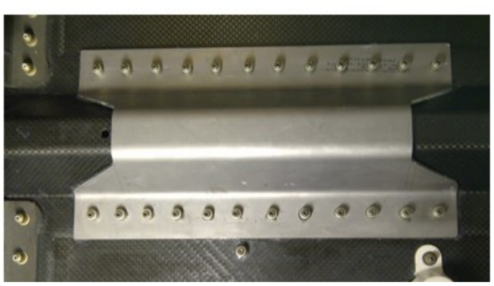


Figure 225. Lesson 3—Composite damage tolerance is good; NDI will tell

- Airplane was hit by air stand during a windstorm
- Caused delamination which was found externally
- Aircraft maintenance documentation required internal inspection







- Internal inspection revealed cracked stringer
- Damage repaired with a metallic bolted repair
- Internal damage would not have been found without internal inspection

Figure 226. Lesson 4—Follow OEM documentation

7. MODULE 5: NDI PROFICIENCY SPECIMENS

In addition to the development of the lecture portion of the Composite Inspector Training Course, a set of NDI proficiency specimens were designed and fabricated. The purpose of the NDI proficiency specimen set was to reinforce teaching points from the classroom, provide representative components used for hands-on training, support recurrent training and composite NDI exposure, and allow for use in both the blind mode and in feedback mode with templates. The overall configuration, thickness, defect size, and material selection was determined with industry feedback gathered at the first composite inspector training workshop held at Delta Air Lines in Atlanta, GA in August 2014. A list of workshop participants can be seen in the acknowledgements section of this report.

Module 5 in the Composite Inspector Training Course lecture series presents the NDI proficiency specimens in figures 227–263. The course instructor can use these materials and provide immediate, visual feedback to the inspectors after they have conducted an inspection on one of the specimens. It contains drawings, a flaw profile, a picture, and C-scan inspection results of each specimen. Appendix B contains the fabrication drawings for the specimen set and details regarding assembly and materials acquisition. If the airline conducting the training course decides to fabricate a set of NDI proficiency specimens following the drawings and guidelines in this report, it is recommended that similar C-scan inspection results be generated on the specific specimens that the inspectors will be using.

A summary of the NDI proficiency specimen set is shown in figure 229. The specimen set consists of eight specimens, each containing a different flaw profile. There are three different configurations of specimen. Configuration 1 contains a taper with two different taper ratios (10:1 and 20:1) and a secondarily bonded substructure element. Configuration 1 specimens contain areas of uniform thickness ranging from 16 to 64 plies. Configuration 2 specimens contain built-up pads, fastened shear ties, sealant, and acoustic tiles. Configuration 3 specimens are simple, flat laminate specimens without substructure and contain impact-induced delamination.
Configuration 1 and 2 specimens contain various types of engineered defects that when inspected with ultrasonics induce varying levels of signal reflection. The engineered defects used in the proficiency specimens are shown in figures 230–231. A spreadsheet was used for each specimen to distribute defect size and spacing evenly throughout the laminate thickness. An example flaw profile spreadsheet is shown in figure 232. After fabrication of the specimens, a second workshop was conducted in August 2015, also at Delta Air Lines in Atlanta, GA, where in addition to providing feedback on course content, participants conducted inspections on the specimen set and provided additional feedback regarding specific hands-on exercises.

The purpose of the NDI proficiency specimens is:

- Reinforce teaching points from the classroom
- Representative components used for hands-on training
- Support recurrent training and composite NDI exposure
- Used with templates or blind mode

Figure 227. Purpose of the composite NDI proficiency specimens

Class Definition - NDI Proficiency Specimens

Configuration/Design

- Total laminate/sub thickness = up to 0.5”
- Bonded and co-cured substructure
- Pad-ups with tapers around perimeter
- Fastened shear ties
- Uniform thickness and complex geometry - use of fasteners, bond lines, adhesive layers, sealant layers
- Uses – a) blind mode & b) use of Mylar templates for training

- Material - Pre-preg uniaxial tape
- Fab cycle – 85 psi autoclave cure
- Flaw sizes – 0.25” through 2.0” dia.; including irregular shapes
- Taper ratio = 10:1 and 20:1
- Cost - ~$10K range for sub-set

Figure 228. Description of the composite NDI proficiency specimens

Eight panels:

- **3 variations of configuration 1 panel**
- **2 variations of configuration 2 panel**
- **3 configuration 3 panels**

Panel Configuration	Structure	Test Specimen	Primary Variation
Configuration 1	24"x18" Panel with complex taper (10:1 and 20:1) and secondary bond	1a	Standard configuration 1
		1b	Additional Secondary bond and more subtle flaws (different flaw profile)
		1c	Additional thickness (up to 64 plies) and different flaw profile
Configuration 2	24"x18" Panel with pads, fasteners, co-cured bonds, sealant, sound dampers	2a	Standard configuration 2
		2b	Different flaw profile
Configuration 3	16ply solid lamiante skin	3a	Standard configuration 3
		3b	Subtle impact
		3c	Large impact

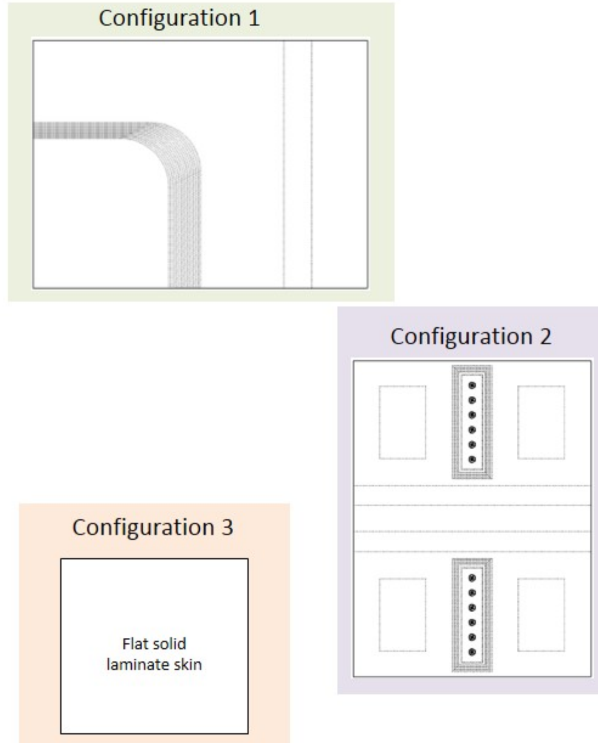


Figure 229. Proficiency specimen configuration summary

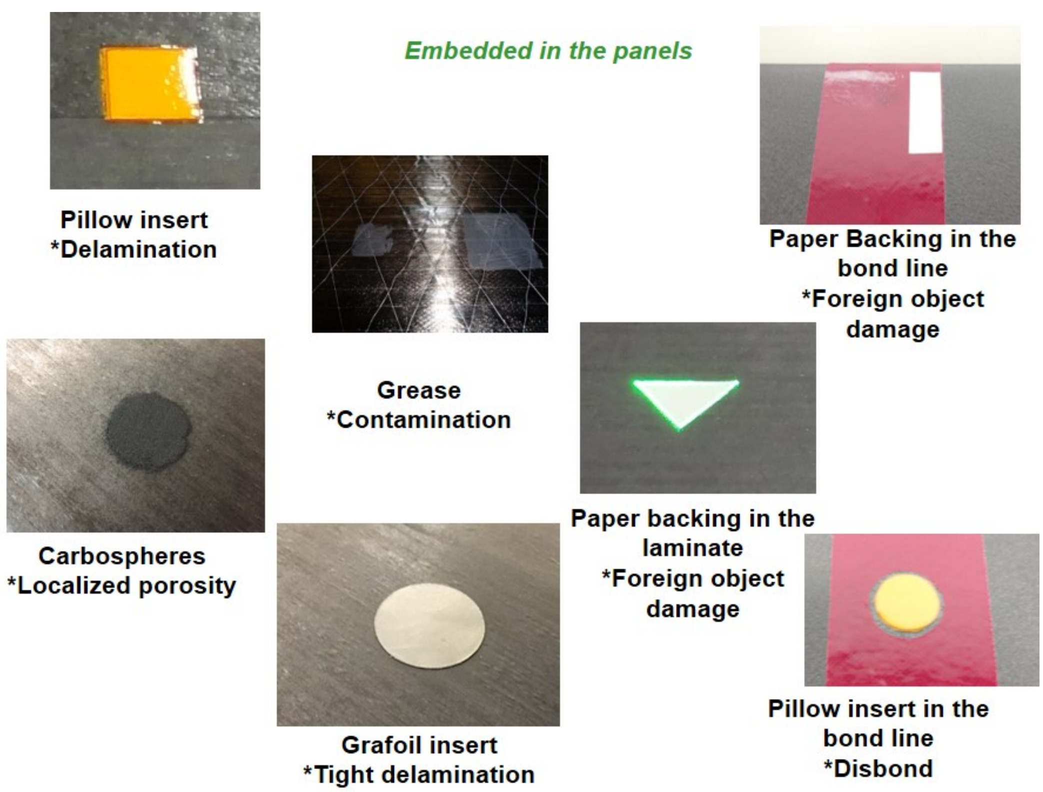


Figure 230. Pictures of engineered flaws embedded in the specimens

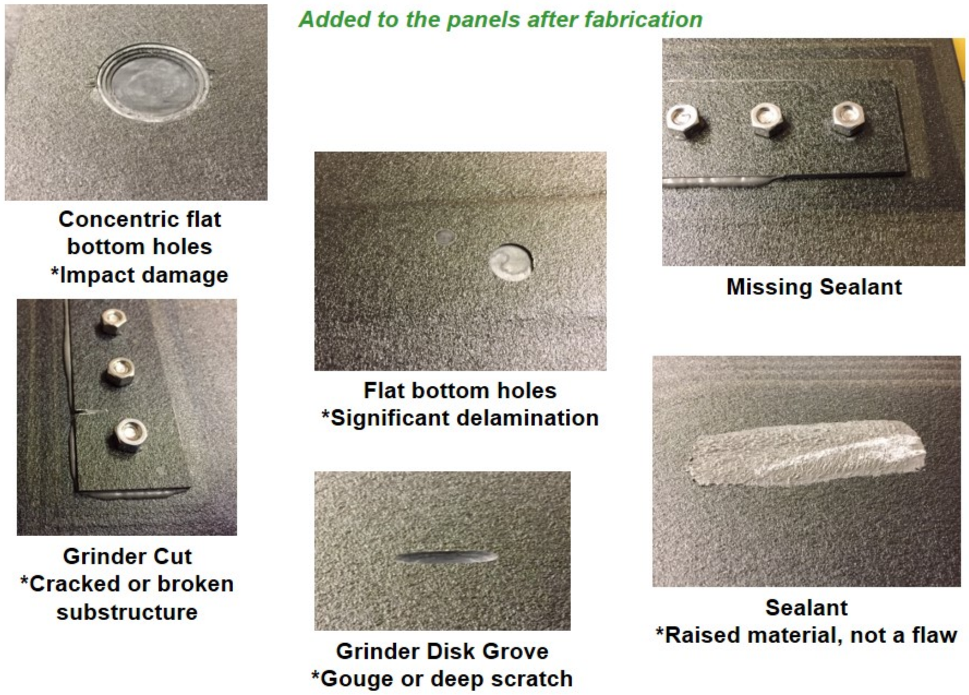


Figure 231. Engineered flaws added to the specimens after fabrication

Composite NDI Proficiency Specimens - **Panel 1c (32-64 Ply 2nd Bonded SS)**								
Flaw Size (Dia. In Inches)	0.25	0.50	0.75	1.00	1.25	1.50	2.00	Total
Number of Flaws	1	2	3	5	1	5	2	19
Lam - PI - Pillow Insert				1	1	1		3
Lam - FBH - Flat Bottom Hole		1				1		2
Lam - IMP - Impact						1		1
Lam - GF - Graphite Foil				1				1
Lam - GR - Grease								0
Lam - POR - Carbospheres							1	1
Lam - PB - Pre Preg Backing			1				1	2
Lam - Grind Disk Groove	1							1
Total Flaws in Lam. Each Size	1	1	1	2	1	3	2	11
Tap - PI - Pillow Insert			1	1		1		3
Tap - FBH - Flat Bottom Hole								0
Tap - IMP - Impact								0
Tap - GF - Graphite Foil								0
Tap - GR - Grease				1				1
Tap - POR - Carbospheres								0
Tap - PB - Pre Preg Backing								0
Total Flaws in Taper Each Size	0	0	1	2	0	1	0	4
Bond - PI - Pillow Insert			1					1
Bond -FBH - Flat Bottom Hole								0
Bond - POR - Glass Microball.								0
Bond - GF - Graphite Foil								0
Bond - PB - Adh. Paper Backing						1		1
Total Flaws in Bond Each Size	0	0	1	0	0	1	0	2
Second Str - PI - Pillow Insert				1				1
Second Str -FBH - Flat Bottom Hole		1						1
Total Flaws in 2ndary Bonded SS	0	1	0	1	0	0	0	2
Total Flaw Area (in^2)	0.05	0.39	1.33	3.93	1.23	8.84	6.28	22.04
Total Insp.Area = 23.5" X 17.5" = 411.25 in^2			Ratio - unflawed / flawed area =				18.66	

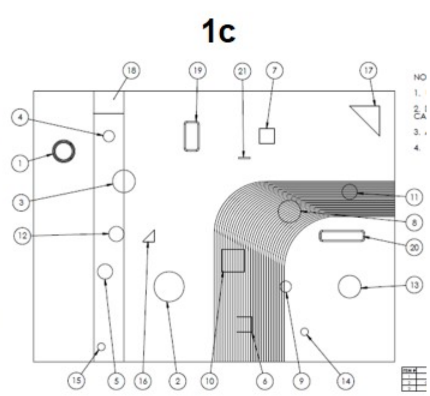


Figure 232. Example flaw profile for specimen 1C

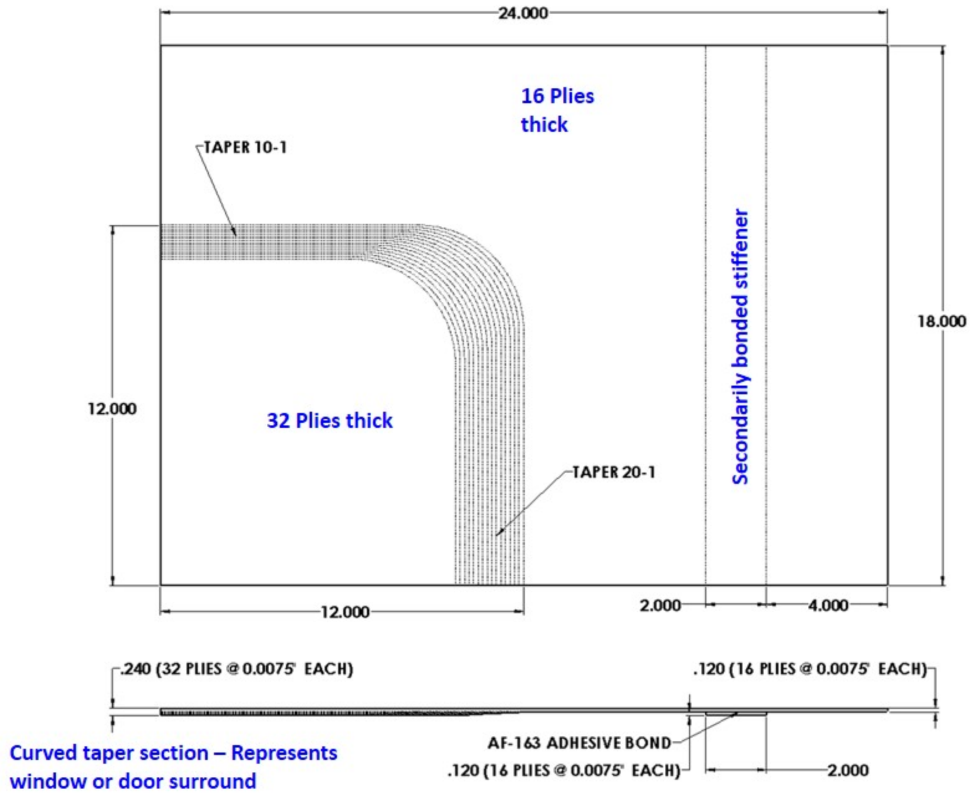
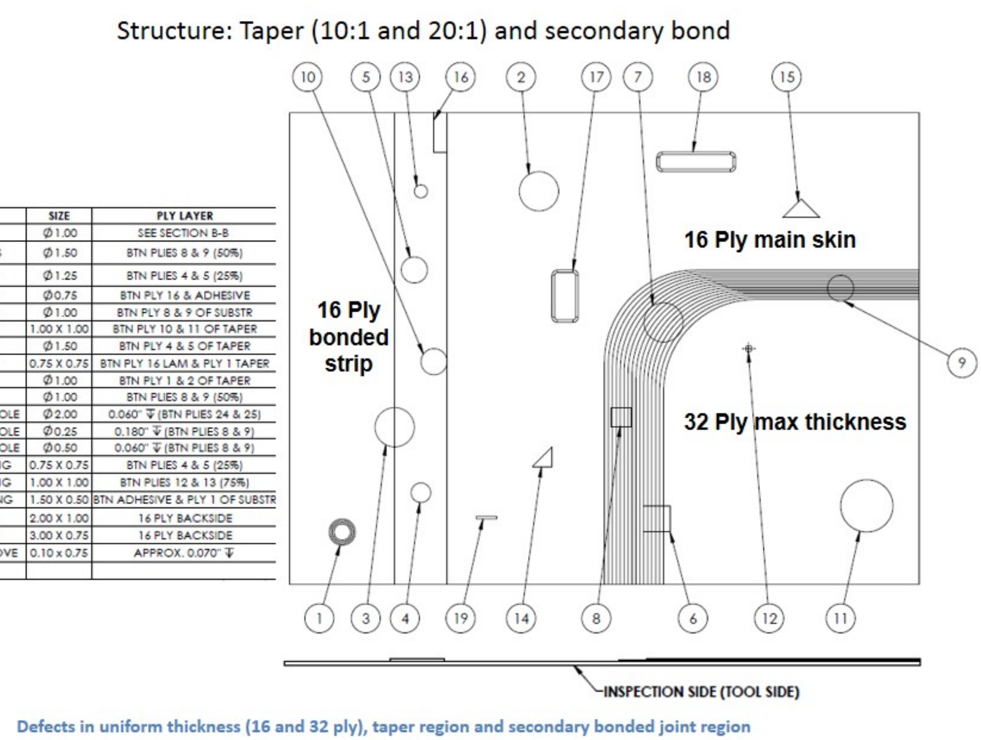


Figure 233. Configuration 1A—Schematic



ITEM #	FLAW TYPE	SIZE	PLY LAYER
1	IMPACT	Ø1.00	SEE SECTION B-B
2	CARBOSPHERES	Ø1.50	BTN PLIES 8 & 9 (50%)
3	PILLOW INSERT	Ø1.25	BTN PLIES 4 & 5 (25%)
4	PILLOW INSERT	Ø0.75	BTN PLY 16 & ADHESIVE
5	PILLOW INSERT	Ø1.00	BTN PLY 8 & 9 OF SUBSTR
6	PILLOW INSERT	1.00 X 1.00	BTN PLY 10 & 11 OF TAPER
7	PILLOW INSERT	Ø1.50	BTN PLY 4 & 5 OF TAPER
8	PILLOW INSERT	0.75 X 0.75	BTN PLY 16 LAM & PLY 1 TAPER
9	GREASE	Ø1.00	BTN PLY 1 & 2 OF TAPER
10	GRAFOIL	Ø1.00	BTN PLIES 8 & 9 (50%)
11	FLAT BOTTOMED HOLE	Ø2.00	0.060" ↧ (BTN PLIES 24 & 25)
12	FLAT BOTTOMED HOLE	Ø0.25	0.180" ↧ (BTN PLIES 8 & 9)
13	FLAT BOTTOMED HOLE	Ø0.50	0.060" ↧ (BTN PLIES 8 & 9)
14	PREPREG BACKING	0.75 X 0.75	BTN PLIES 4 & 5 (25%)
15	PREPREG BACKING	1.00 X 1.00	BTN PLIES 12 & 13 (75%)
16	ADHESIVE BACKING	1.50 X 0.50	BTN ADHESIVE & PLY 1 OF SUBSTR
17	SEALANT BLOB	2.00 X 1.00	16 PLY BACKSIDE
18	SEALANT BLOB	3.00 X 0.75	16 PLY BACKSIDE
19	GRIND DISK GROOVE	0.10 x 0.75	APPROX. 0.070" ↧

Figure 234. Specimen 1A—Flaw profile

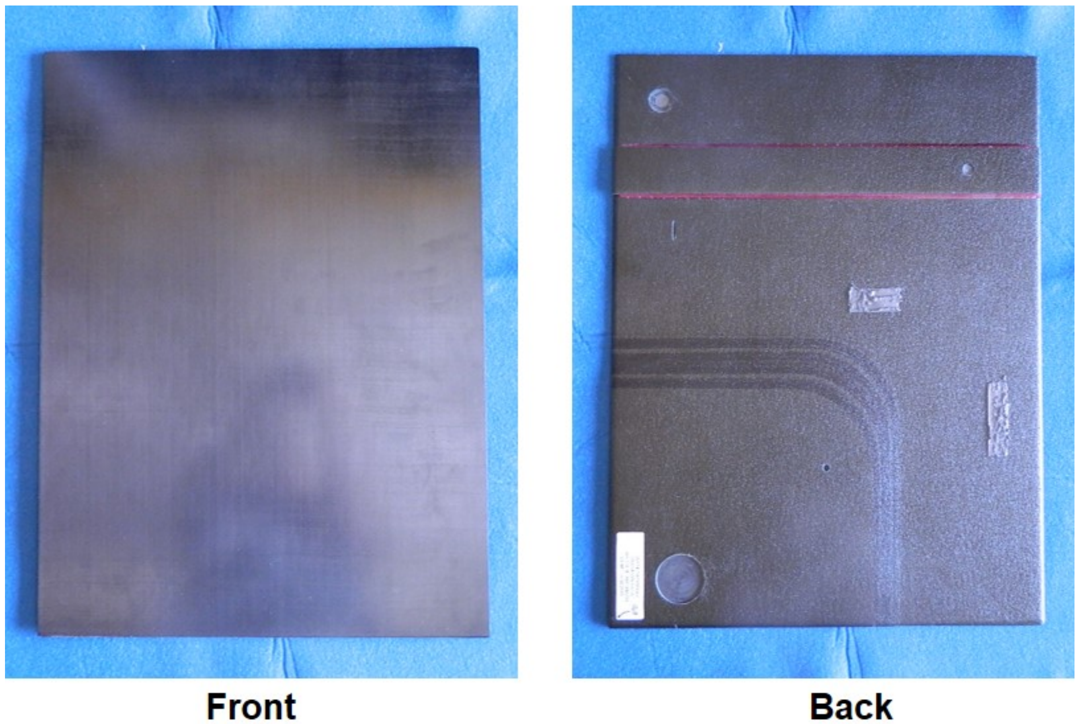


Figure 235. Specimen 1A—Picture

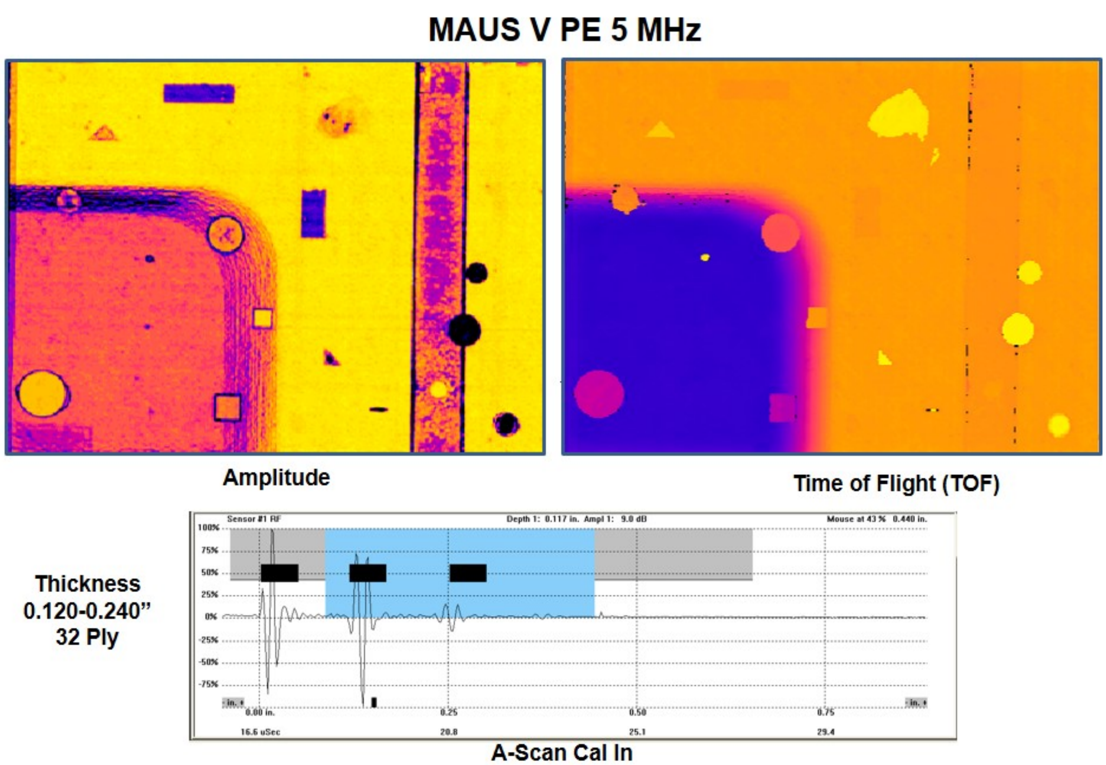


Figure 236. Specimen 1A—Pulse Econ (PU) UT inspection results

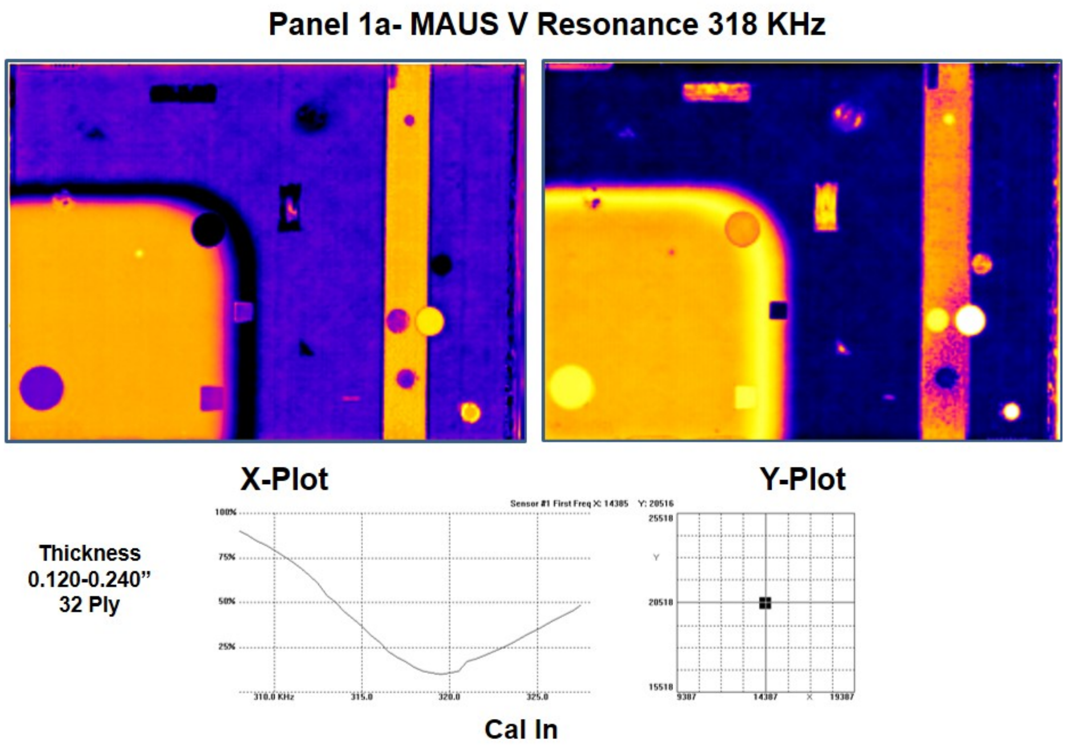


Figure 237. Specimen 1A—Resonance inspection results

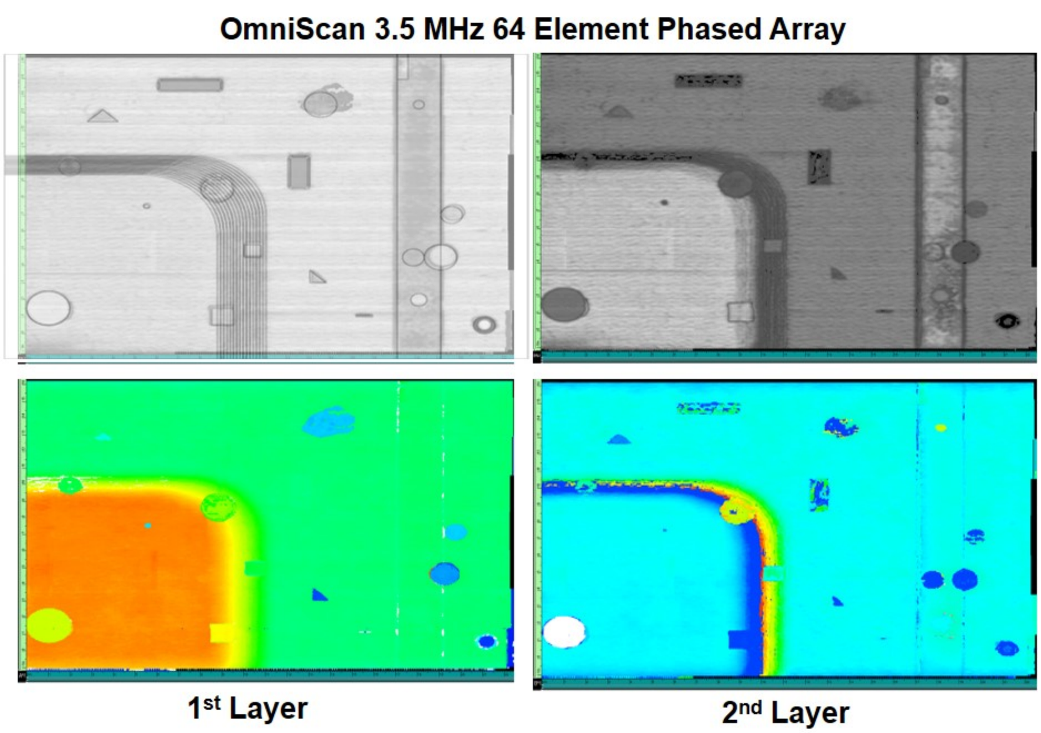


Figure 238. Specimen 1A—Phased array ultrasonic transmission (PA UT) inspection results

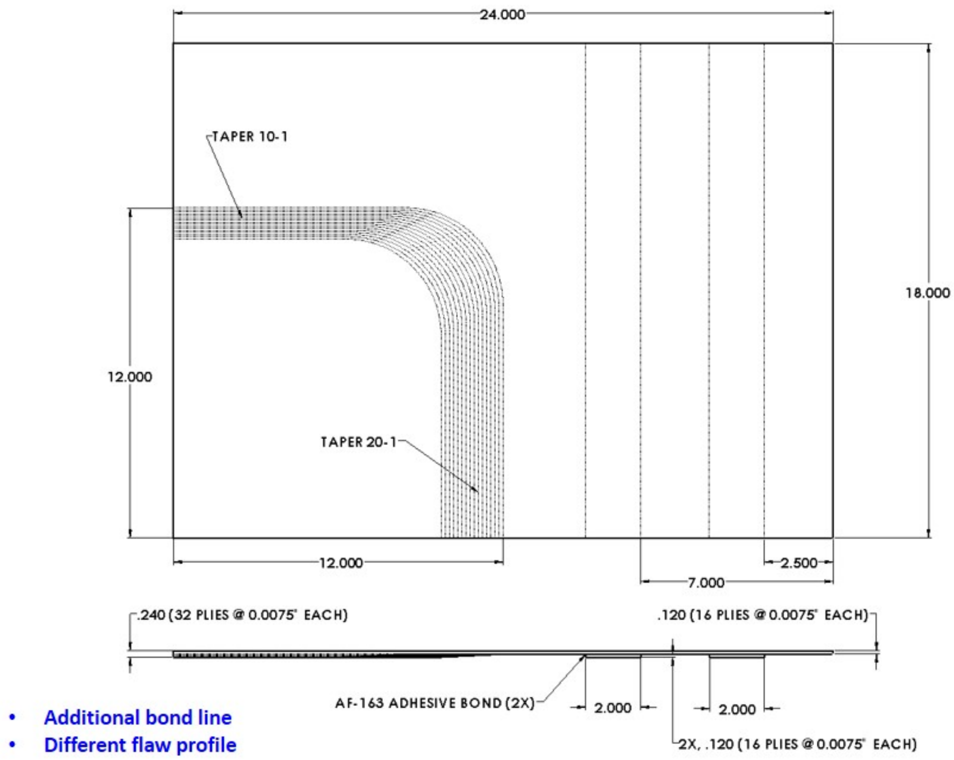
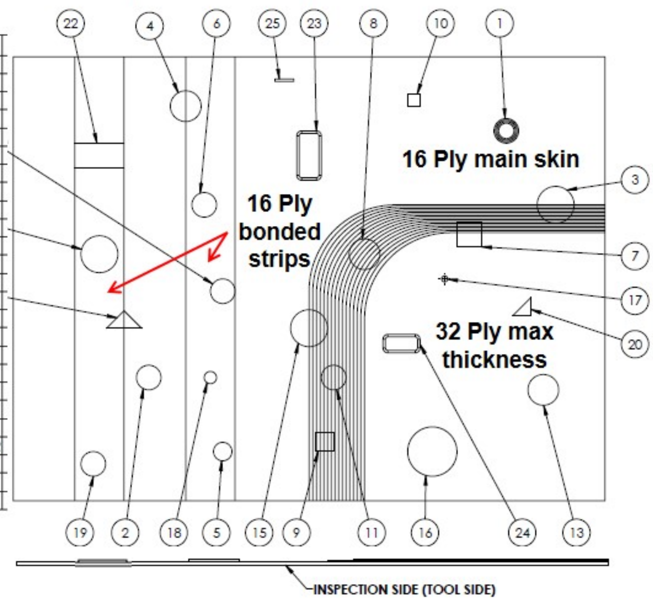


Figure 239. Configuration 1B—Schematic

Structure: Taper (10:1 and 20:1) and secondary bond

ITEM #	FLAW TYPE	SIZE	PLY LAYER
1	IMPACT	Ø1.00	SEE SECTION B-B
2	CARBOSPHERES	Ø1.00	BTN PLIES 8 & 9 (50%)
3	CARBOSPHERES	Ø1.50	BTN PLIES 4 & 5 (25%)
4	PILLOW INSERT	Ø1.25	BTN PLIES 4 & 5 (25%)
5	PILLOW INSERT	Ø 0.75	BTN PLIES 16 & ADHESIVE
6	PILLOW INSERT	Ø1.00	BTN PLIES 8 & 9 OF SUBSTR
7	PILLOW INSERT	1.00 X 1.00	BTN PLIES 10 & 11 OF TAPER
8	PILLOW INSERT	Ø1.25	BTN PLIES 4 & 5 OF TAPER
9	PILLOW INSERT	0.75 X 0.75	BTN PLIES 12 & 13 (75%)
10	PILLOW INSERT	0.50 X 0.50	BTN PLIES 8 & 9 (50%)
11	GREASE	Ø1.00	BTN PLIES 1 & 2 OF TAPER
12	GREASE	Ø1.50	BTN PLIES 12 & 13 (75%)
13	GREASE	Ø1.25	BTN PLIES 8 & 9 OF TAPER
14	GRAFOIL	Ø1.00	BTN PLIES 8 & 9 (50%)
15	GRAFOIL	Ø1.50	BTN PLIES 4 & 5 (25%)
16	FLAT BOTTOMED HOLE	Ø2.00	0.060" ↧ (BTN PLIES 24 & 25)
17	FLAT BOTTOMED HOLE	Ø 0.25	0.180" ↧ (BTN PLIES 8 & 9)
18	FLAT BOTTOMED HOLE	Ø0.50	0.060" ↧ (BTN PLIES 8 & 9)
19	FLAT BOTTOMED HOLE	Ø1.00	0.030" ↧ (BTN PLIES 4 & 5)
20	PREPREG BACKING	0.75 X 0.75	BTN PLIES 4 & 5
21	PREPREG BACKING	1.00 X 1.00	BTN PLIES 12 & 13 (75%)
22	ADHESIVE BACKING	2.00 X 1.00	BTN ADHESIVE & PLY 1 OF SUBSTR
23	SEALANT BLOB	2.00 X 1.00	16 PLY BACKSIDE
24	SEALANT BLOB	1.50X 0.75	32 PLY BACKSIDE
25	GRIND DISK GROOVE	0.10 X 0.75	APPROX. 0.070" ↧



Note: Specimen 1b is the same as 1a with an additional secondary bond and a different flaw profile with more subtle flaws

Figure 240. Specimen 1B—Flaw profile

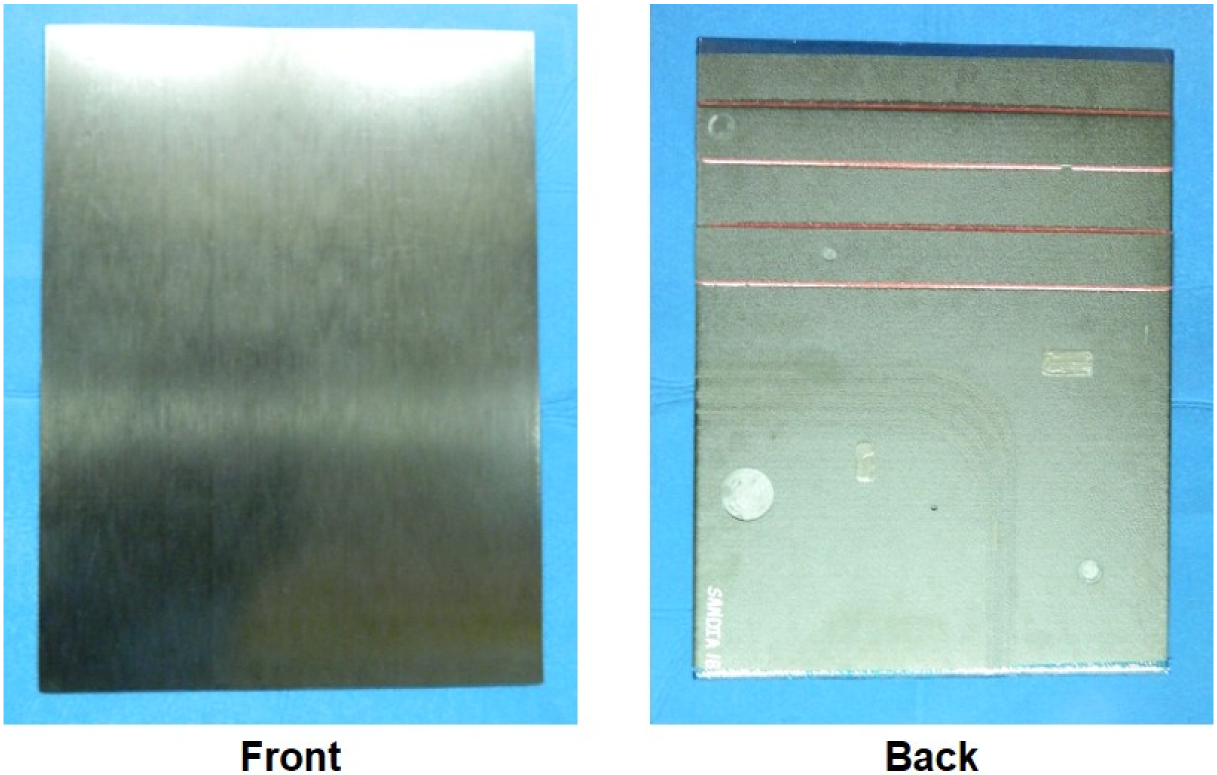


Figure 241. Specimen 1B—Picture

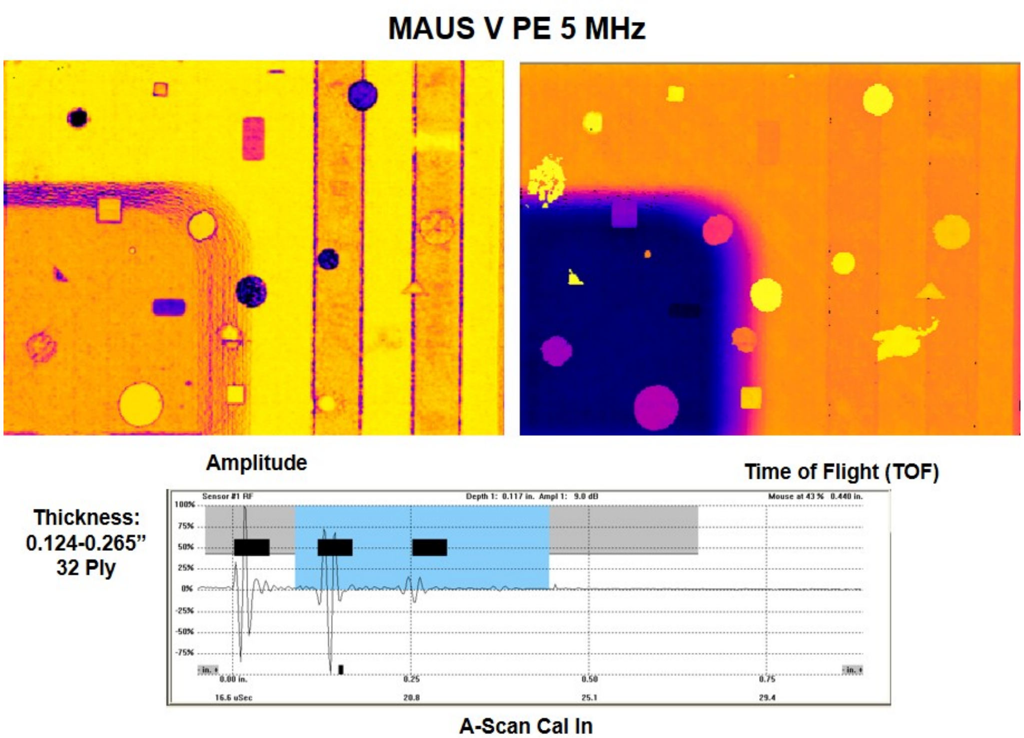


Figure 242. Specimen 1B—PE UT inspection results

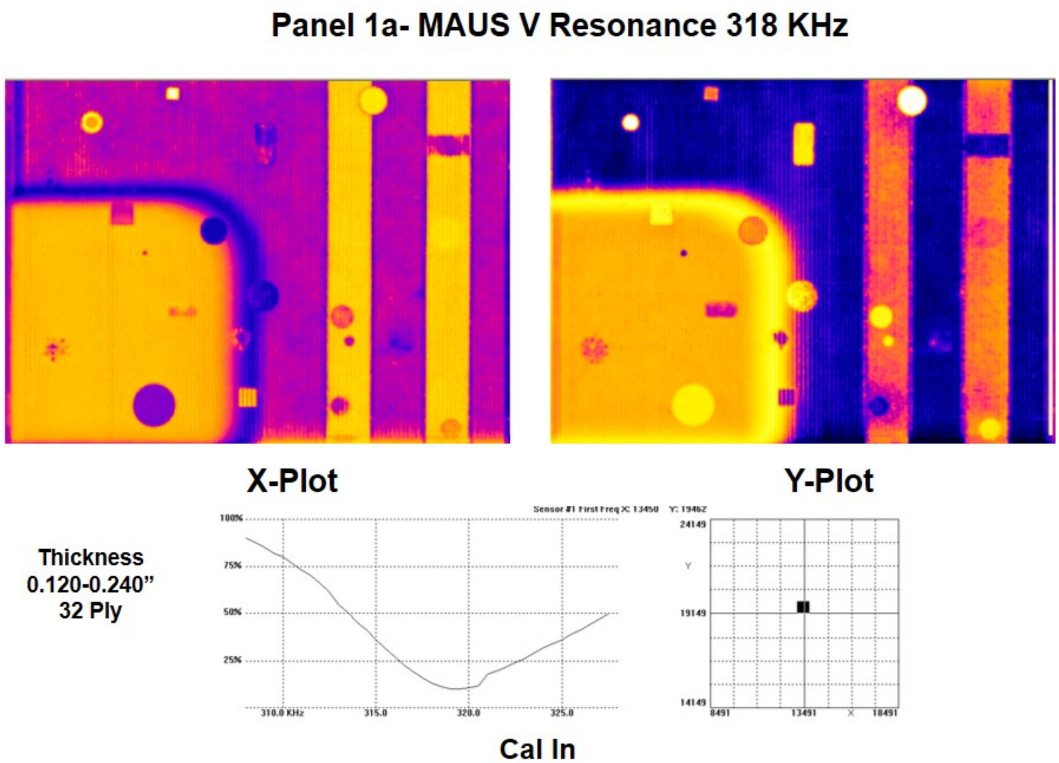


Figure 243. Specimen 1B—Resonance inspection results

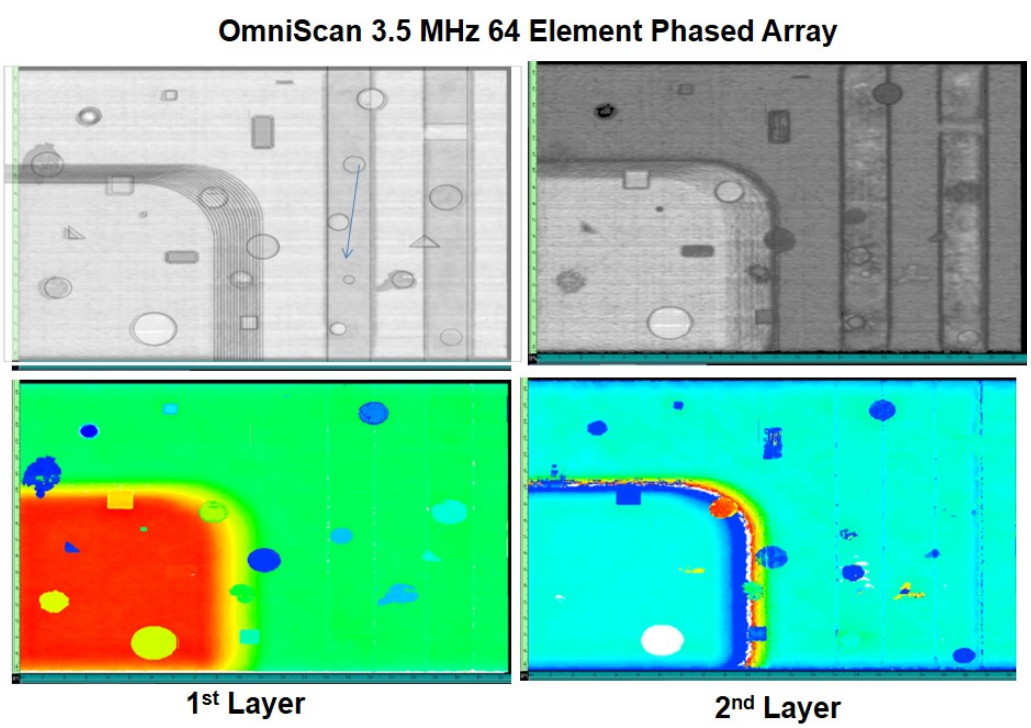
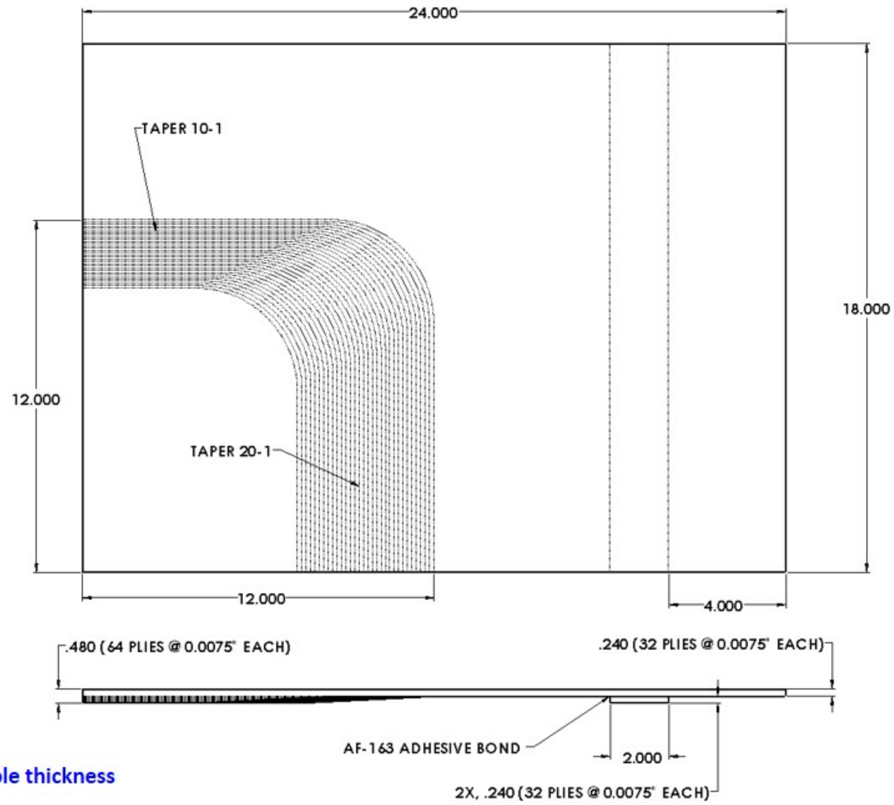


Figure 244. Specimen 1B—PA UT inspection results

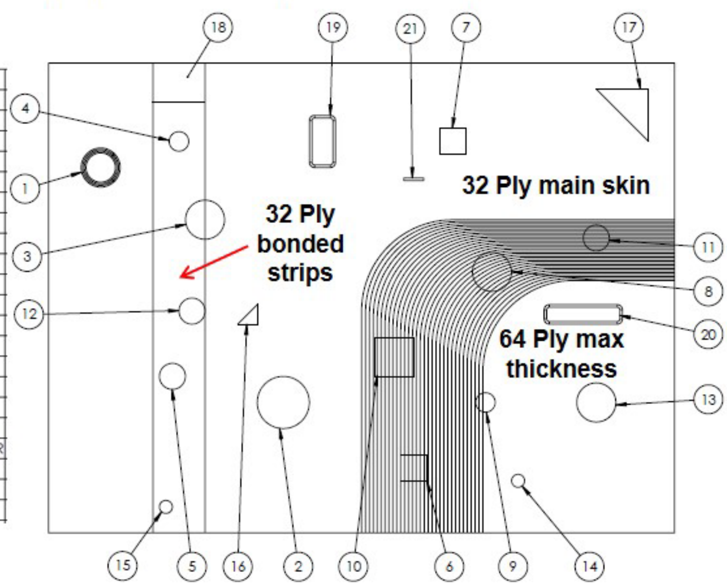


- Double thickness

Figure 245. Configuration 1C—Schematic

Structure: *Thick Specimen* - Taper (10:1 and 20:1) and secondary bond

ITEM #	FLAW TYPE	SIZE	PLY LAYER
1	IMPACT	Ø1.50	SEE SECTION B-B
2	CARBOSPHERES	Ø2.00	BTN PLIES 16 & 17 (50%)
3	PILLOW INSERT	Ø1.50	BTN PLIES 8 & 9 (25%)
4	PILLOW INSERT	Ø0.75	BTN PLY 32 & ADHESIVE
5	PILLOW INSERT	Ø1.00	BTN PLY 16 & 17 OF SUBSTR
6	PILLOW INSERT	1.00 X 1.00	BTN PLY 10 & 11 OF TAPER
7	PILLOW INSERT	1.00 X 1.00	BTN PLY 24 & 25 (75%)
8	PILLOW INSERT	Ø1.50	BTN PLY 4 & 5 OF TAPER
9	PILLOW INSERT	Ø0.75	BTN PLY 26 & 27 OF TAPER
10	PILLOW INSERT	1.50 X 1.50	BTN PLY 24 & 25 (75%)
11	GREASE	Ø1.00	BTN PLY 1 & 2 OF TAPER
12	GRAFOIL	Ø1.00	BTN PLIES 16 & 17 (50%)
13	FLAT BOTTOMED HOLE	Ø1.50	0.120" ↧ (BTN PLIES 16 & 17)
14	FLAT BOTTOMED HOLE	Ø0.50	0.360" ↧ (BTN PLIES 48 & 49)
15	FLAT BOTTOMED HOLE	Ø0.50	0.060" ↧ (BTN PLIES 24 & 25)
16	PREPREG BACKING	0.75 X 0.75	BTN PLIES 8 & 9 (25%)
17	PREPREG BACKING	2.00 X 2.00	BTN PLIES 24 & 25 (75%)
18	ADHESIVE BACKING	2.00 X 1.50	BTN ADHESIVE & PLY 1 OF SUBSTR
19	SEALANT BLOB	2.00 X 1.00	32 PLY BACKSIDE
20	SEALANT BLOB	3.00 X 0.75	64 PLY BACKSIDE
21	GRIND DISK GROOVE	0.10 x 0.75	APPROX. 0.070" ↧



Note: Specimen1c has similar features to 1a and 1b but is twice as thick (~ 0.5").

Figure 246. Specimen 1C—Flaw profile

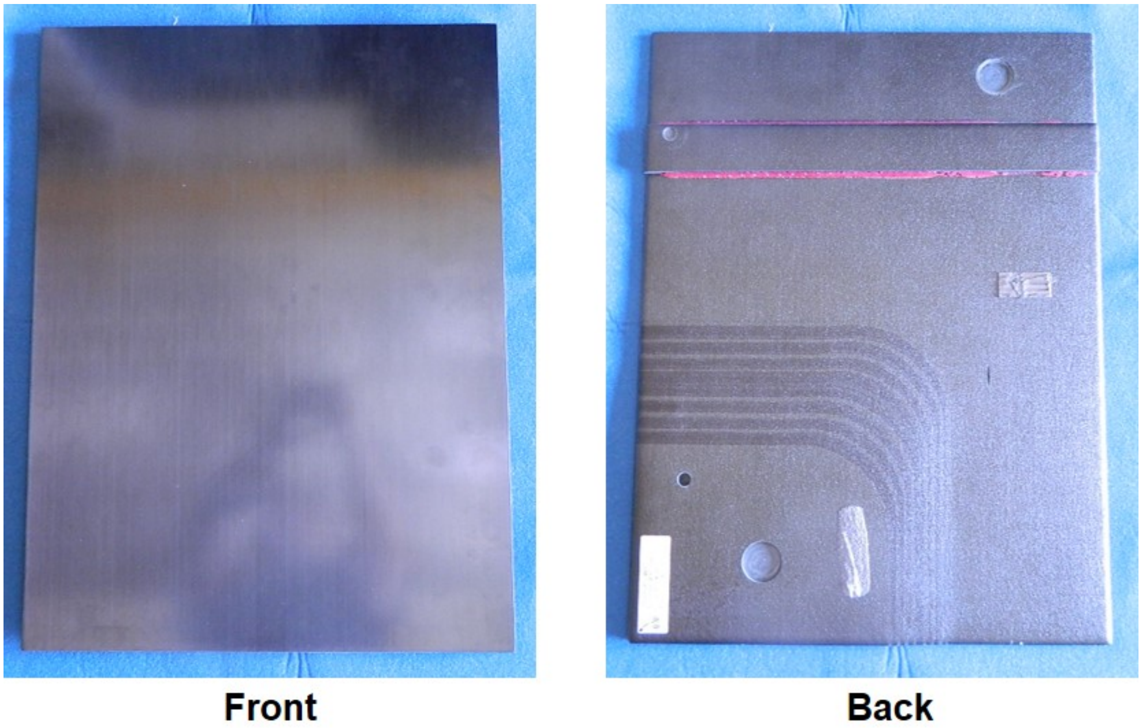


Figure 247. Specimen 1C—Picture

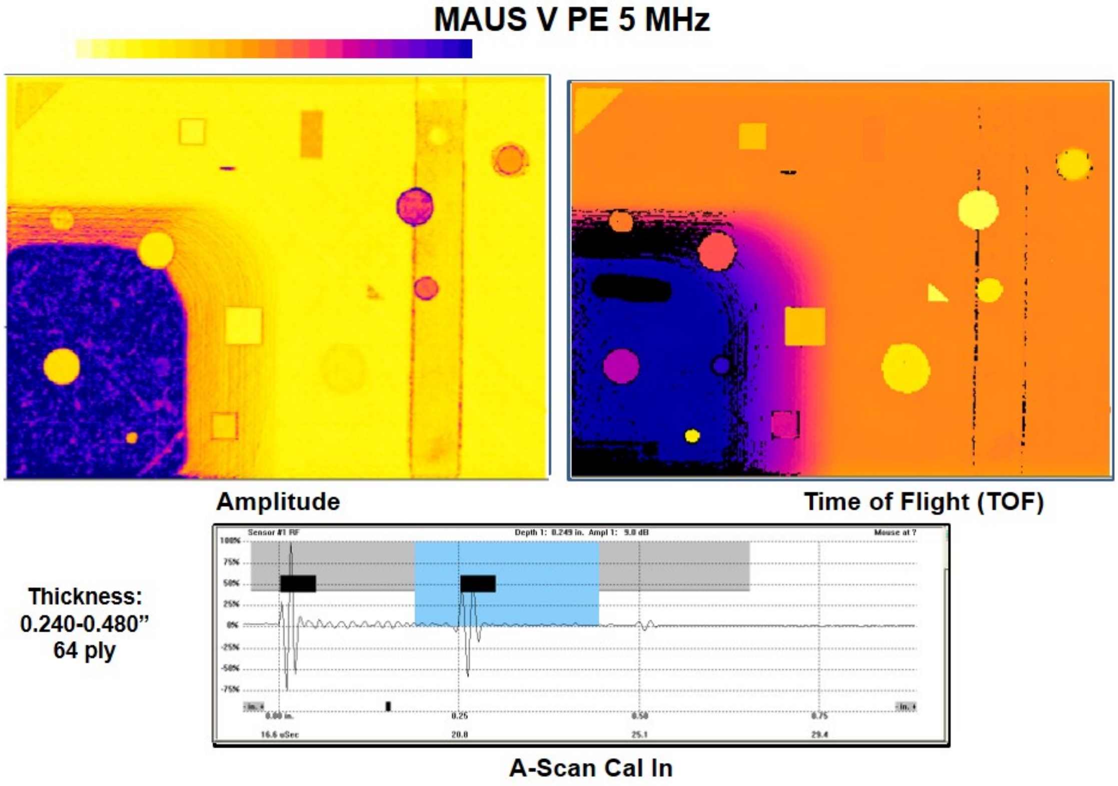


Figure 248. Specimen 1C—PE UT inspection results

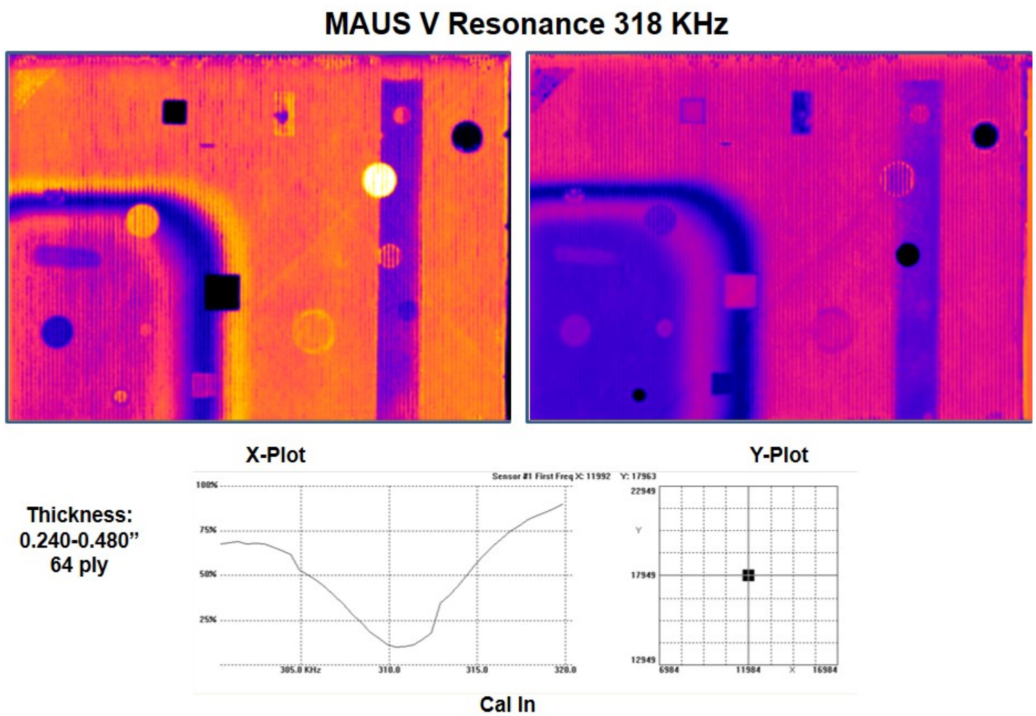


Figure 249. Specimen 1C—Resonance inspection results

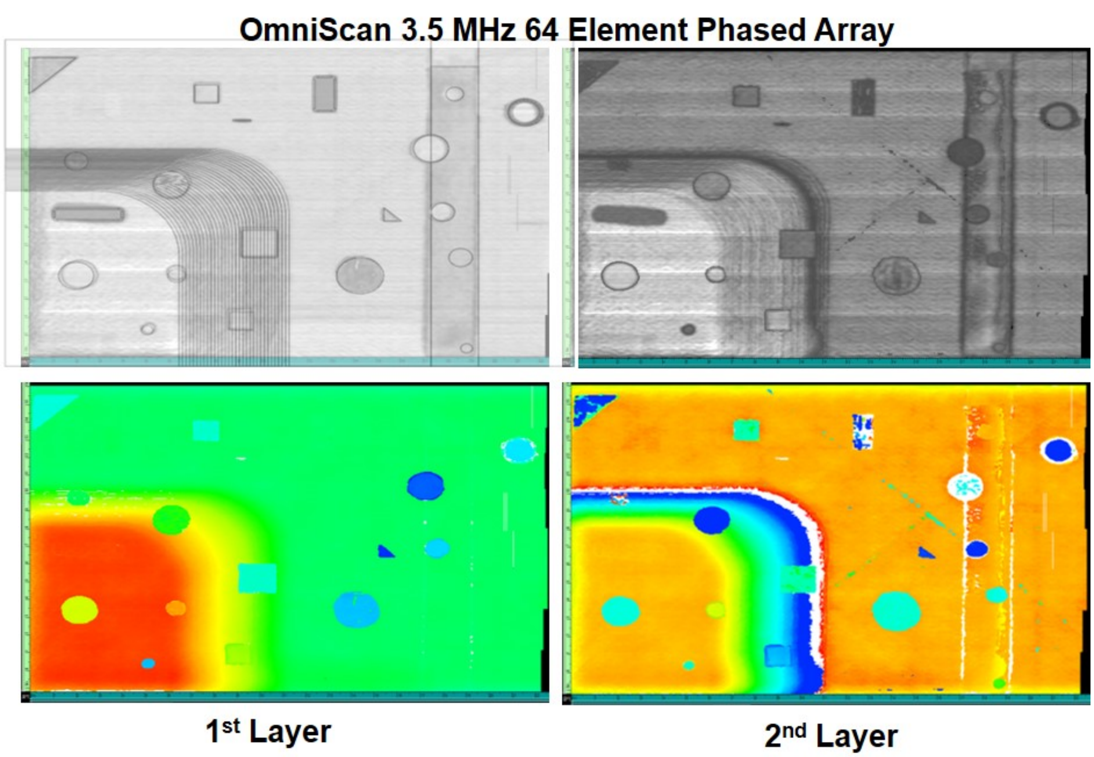


Figure 250. Specimen 1C—PA UT inspection results

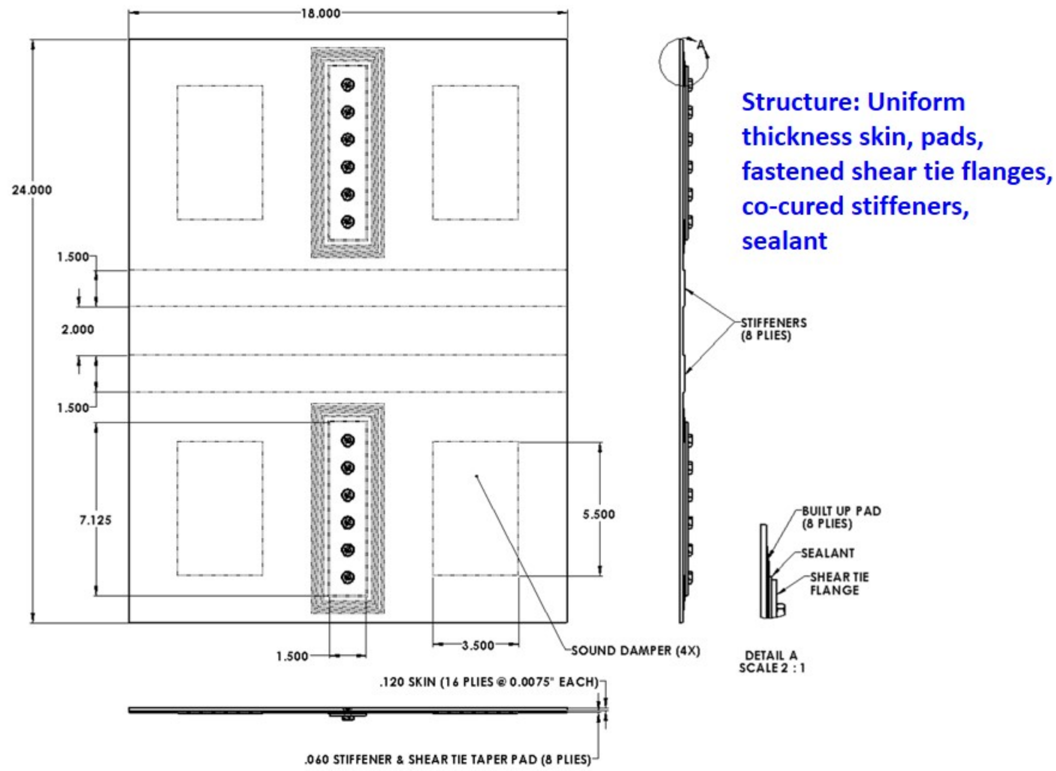


Figure 251. Configuration 2—Schematic

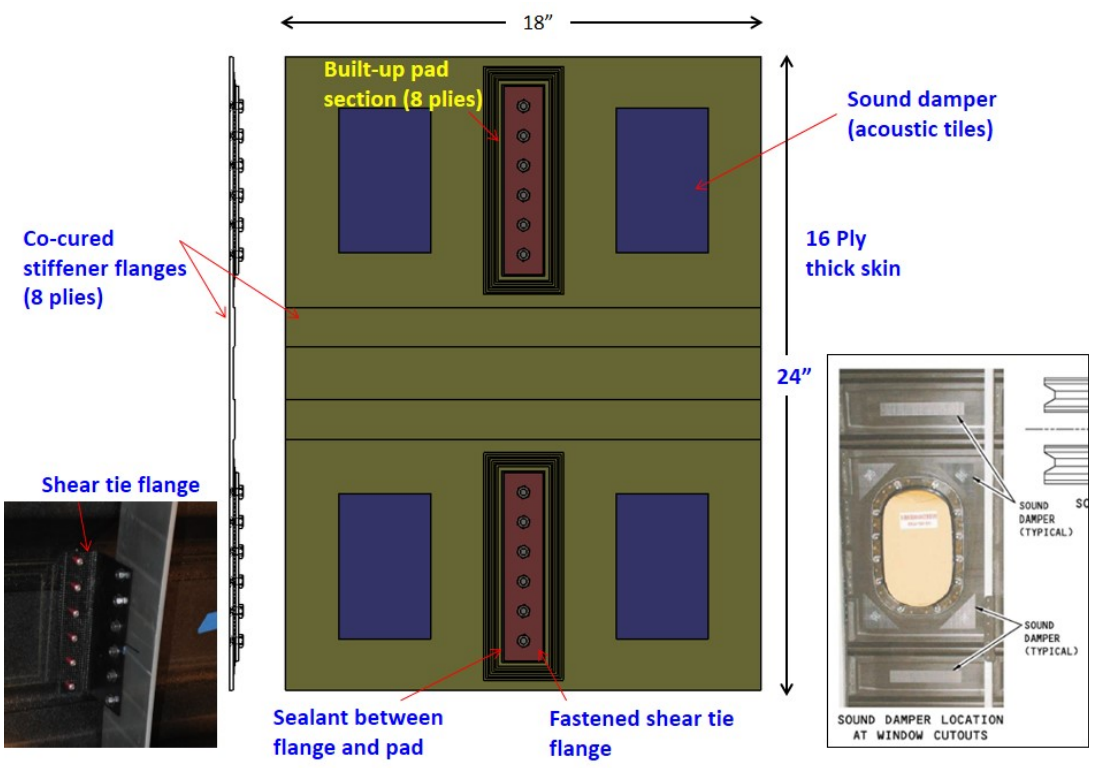


Figure 252. Configuration 2—Substructure description

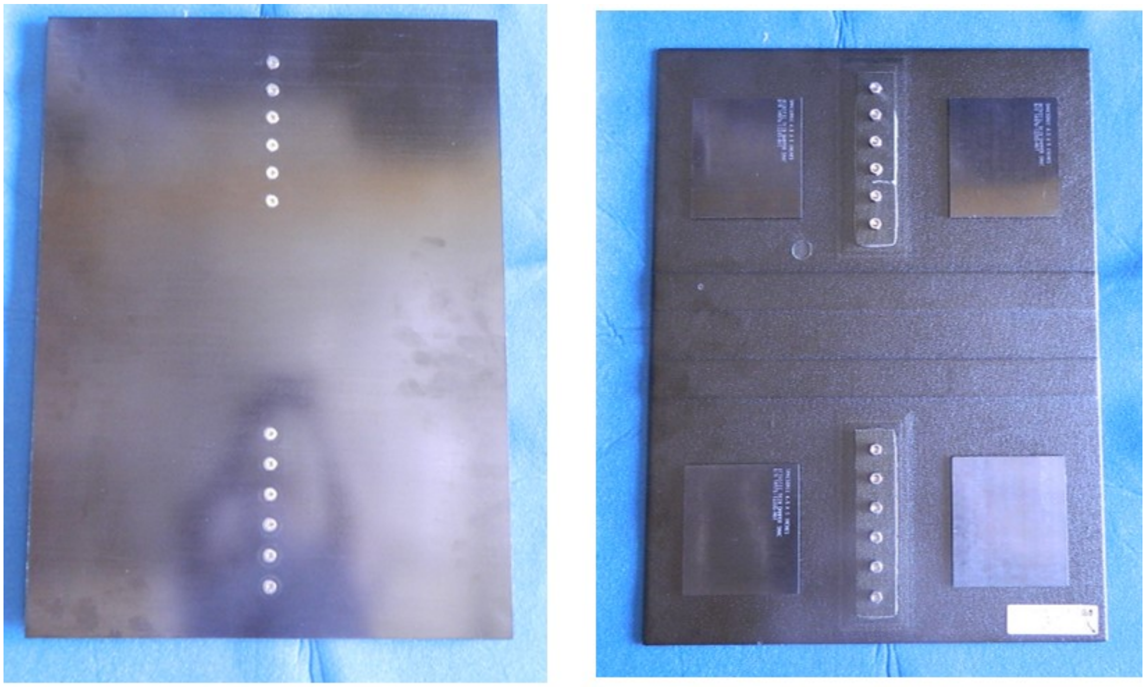
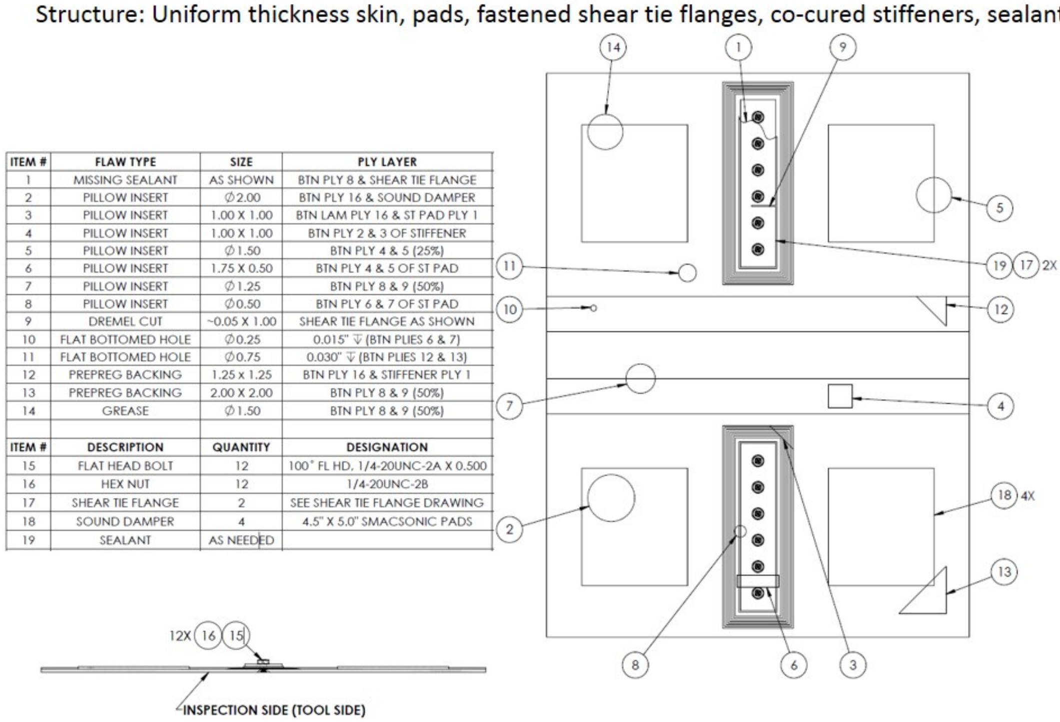


Figure 253. Specimen 2A—Picture

Structure: Uniform thickness skin, pads, fastened shear tie flanges, co-cured stiffeners, sealant



ITEM #	FLAW TYPE	SIZE	PLY LAYER
1	MISSING SEALANT	AS SHOWN	BTN PLY 8 & SHEAR TIE FLANGE
2	PILLOW INSERT	Ø2.00	BTN PLY 16 & SOUND DAMPER
3	PILLOW INSERT	1.00 X 1.00	BTN LAM PLY 16 & ST PAD PLY 1
4	PILLOW INSERT	1.00 X 1.00	BTN PLY 2 & 3 OF STIFFENER
5	PILLOW INSERT	Ø1.50	BTN PLY 4 & 5 (25%)
6	PILLOW INSERT	1.75 X 0.50	BTN PLY 4 & 5 OF ST PAD
7	PILLOW INSERT	Ø1.25	BTN PLY 8 & 9 (50%)
8	PILLOW INSERT	Ø0.50	BTN PLY 6 & 7 OF ST PAD
9	DREMEL CUT	~0.05 X 1.00	SHEAR TIE FLANGE AS SHOWN
10	FLAT BOTTOMED HOLE	Ø0.25	0.015" ↧ (BTN PLIES 6 & 7)
11	FLAT BOTTOMED HOLE	Ø0.75	0.030" ↧ (BTN PLIES 12 & 13)
12	PREPREG BACKING	1.25 x 1.25	BTN PLY 16 & STIFFENER PLY 1
13	PREPREG BACKING	2.00 X 2.00	BTN PLY 8 & 9 (50%)
14	GREASE	Ø1.50	BTN PLY 8 & 9 (50%)
ITEM #	DESCRIPTION	QUANTITY	DESIGNATION
15	FLAT HEAD BOLT	12	100° FL HD, 1/4-20UNC-2A X 0.500
16	HEX NUT	12	1/4-20UNC-2B
17	SHEAR TIE FLANGE	2	SEE SHEAR TIE FLANGE DRAWING
18	SOUND DAMPER	4	4.5" X 5.0" SMACSONIC PADS
19	SEALANT	AS NEEDED	

Figure 254. Specimen 2A—Flaw profile

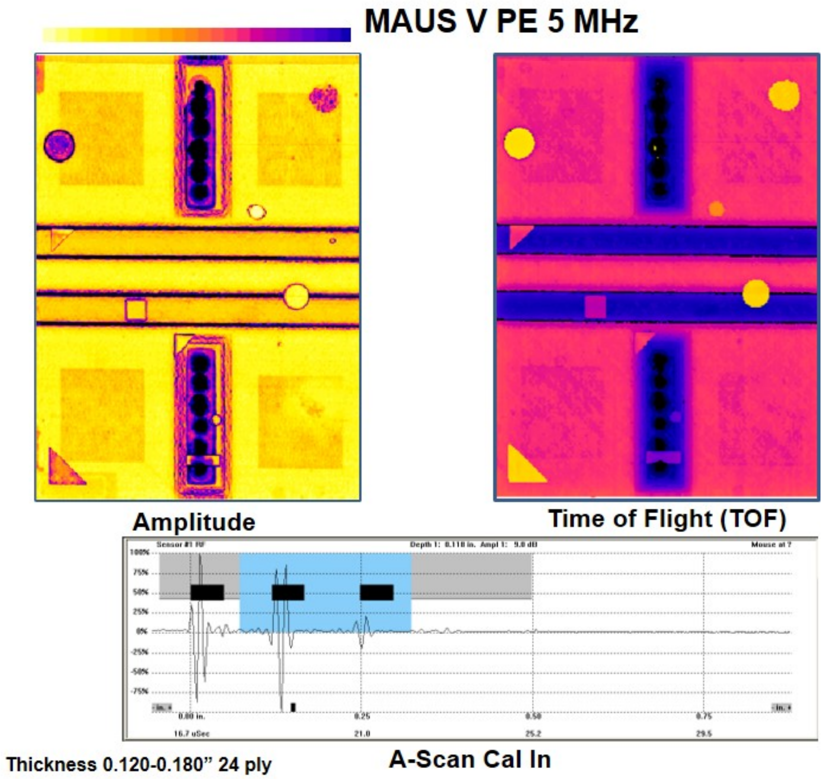


Figure 255. Specimen 2A—PE UT inspection results

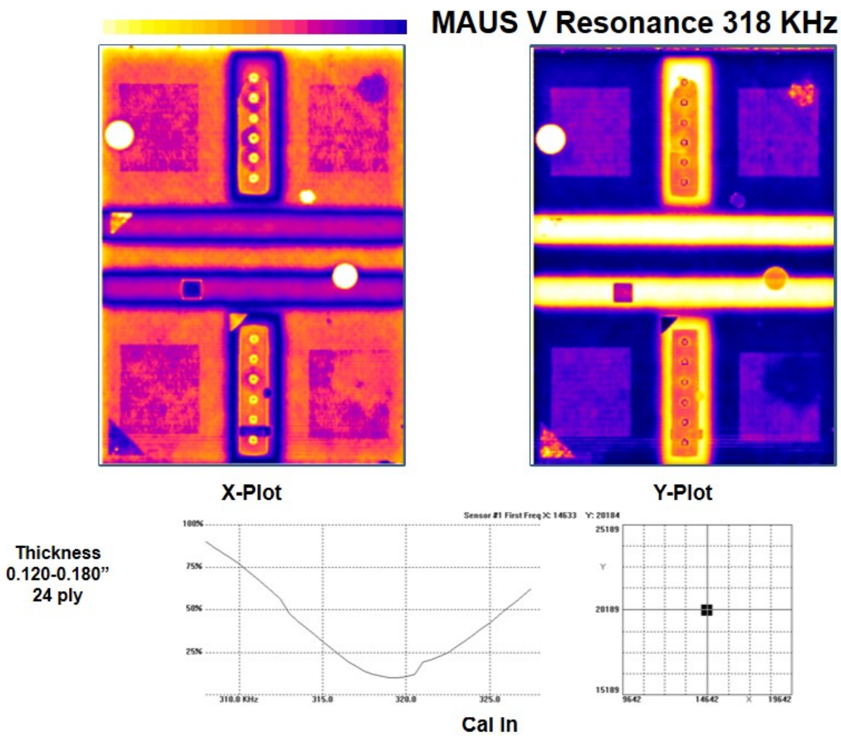


Figure 256. Specimen 2A—Resonance inspection results

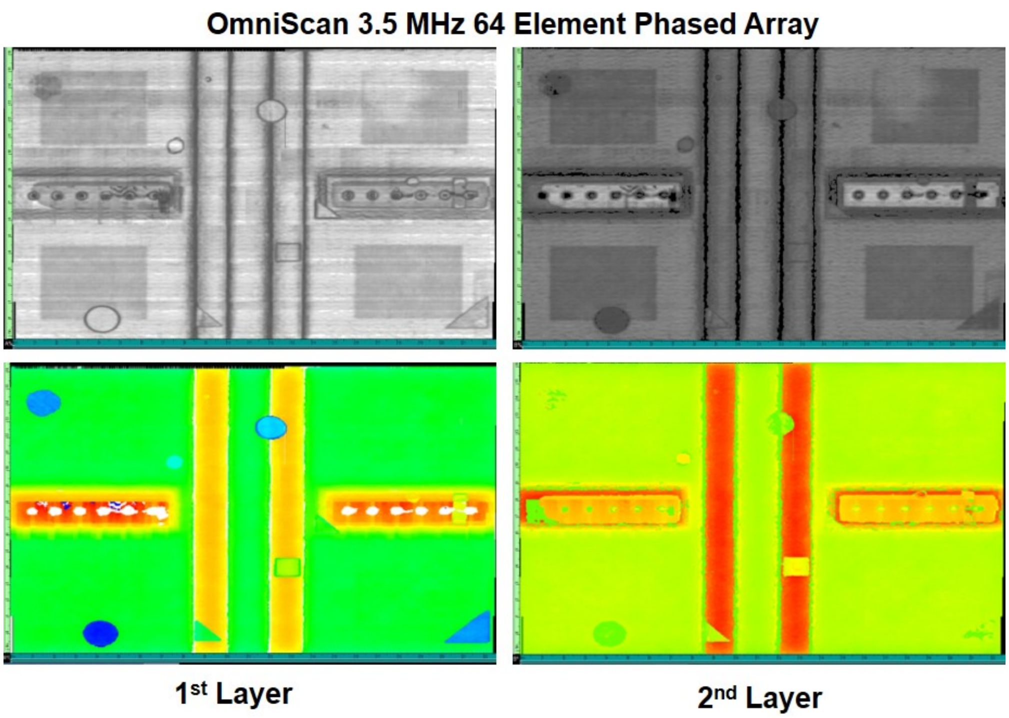
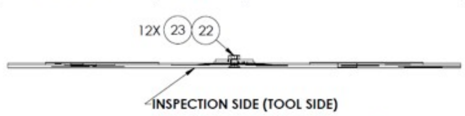
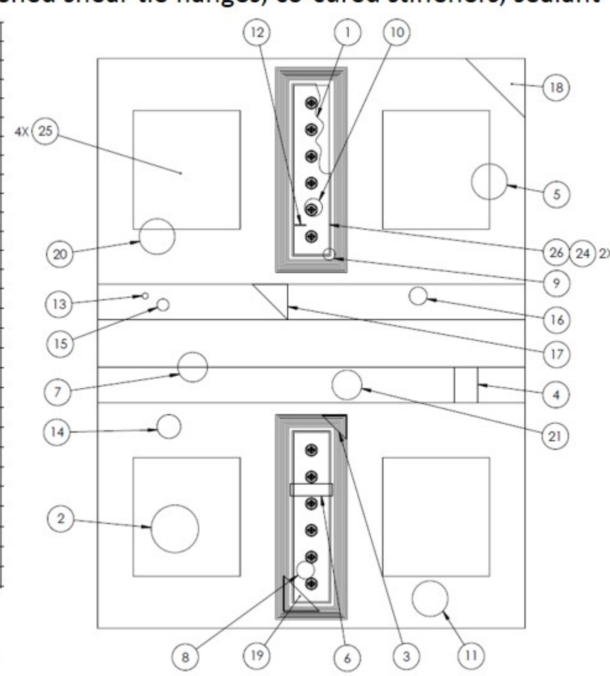


Figure 257. Specimen 2A—PA UT inspection results

Structure: Uniform thickness skin, pads, fastened shear tie flanges, co-cured stiffeners, sealant

ITEM #	FLAW TYPE	SIZE	PLY LAYER
1	MISSING SEALANT	AS SHOWN	BTN PLY 8 & SHEAR TIE FLANGE
2	PILLOW INSERT	Ø2.00	BTN PLY 16 & SOUND DAMPER
3	PILLOW INSERT	1.00 X 1.00	BTN LAM PLY 16 & ST PAD PLY 1
4	PILLOW INSERT	1.50 X 1.00	BTN PLY 2 & 3 OF STIFFENER
5	PILLOW INSERT	Ø1.50	BTN PLY 4 & 5 (25%)
6	PILLOW INSERT	1.75 X 0.50	BTN PLY 7 & 8 OF ST PAD
7	PILLOW INSERT	Ø1.25	BTN PLY 8 & 9 (50%)
8	PILLOW INSERT	Ø0.75	BTN PLY 2 & 3 OF ST PAD
9	PILLOW INSERT	Ø0.50	BTN PLY 6 & 7 OF ST PAD
10	PILLOW INSERT	Ø0.75	BTN PLY 6 & 7 OF ST PAD
11	CARBOSPHERES	Ø1.50	BTN PLY 12 & 13 (75%)
12	DREMEL CUT	~0.05 X 0.50	SHEAR TIE FLANGE AS SHOWN
13	FLAT BOTTOMED HOLE	Ø0.25	0.015" ↧ (BTN PLIES 6 & 7)
14	FLAT BOTTOMED HOLE	Ø1.00	0.030" ↧ (BTN PLIES 12 & 13)
15	FLAT BOTTOMED HOLE	Ø0.50	0.045" ↧ (BTN PLIES 2 & 3)
16	FLAT BOTTOMED HOLE	Ø0.75	0.165" ↧ (BTN PLIES 2 & 3)
17	PREPREG BACKING	1.50 x 1.50	BTN PLY 16 & STIFFENER PLY 1
18	PREPREG BACKING	2.50 X 2.50	BTN PLY 8 & 9 (50%)
19	PREPREG BACKING	1.50 x 1.50	BTN PLY 5 & 6 OF ST PAD
20	GREASE	Ø1.50	BTN PLY 8 & 9 (50%)
21	GRAFOIL	Ø1.25	BTN PLY 12 & 13 (75%)
ITEM #	**DESCRIPTION**	**QUANTITY**	**DESIGNATION**
22	FLAT HEAD BOLT	12	100° FL HD. 1/4-20UNC-2A X 0.500
23	HEX NUT	12	1/4-20UNC-2B
24	SHEAR TIE FLANGE	2	SEE SHEAR TIE FLANGE DRAWING
25	SOUND DAMPER	4	4.5" X 5.0" SMACSONIC PADS
26	SEALANT	AS NEEDED	





Note: **Specimen 2b** is the same configuration as 2a with an additional secondary bond and a different flaw profile with more subtle flaws

Figure 258. Specimen 2B—Flaw profile

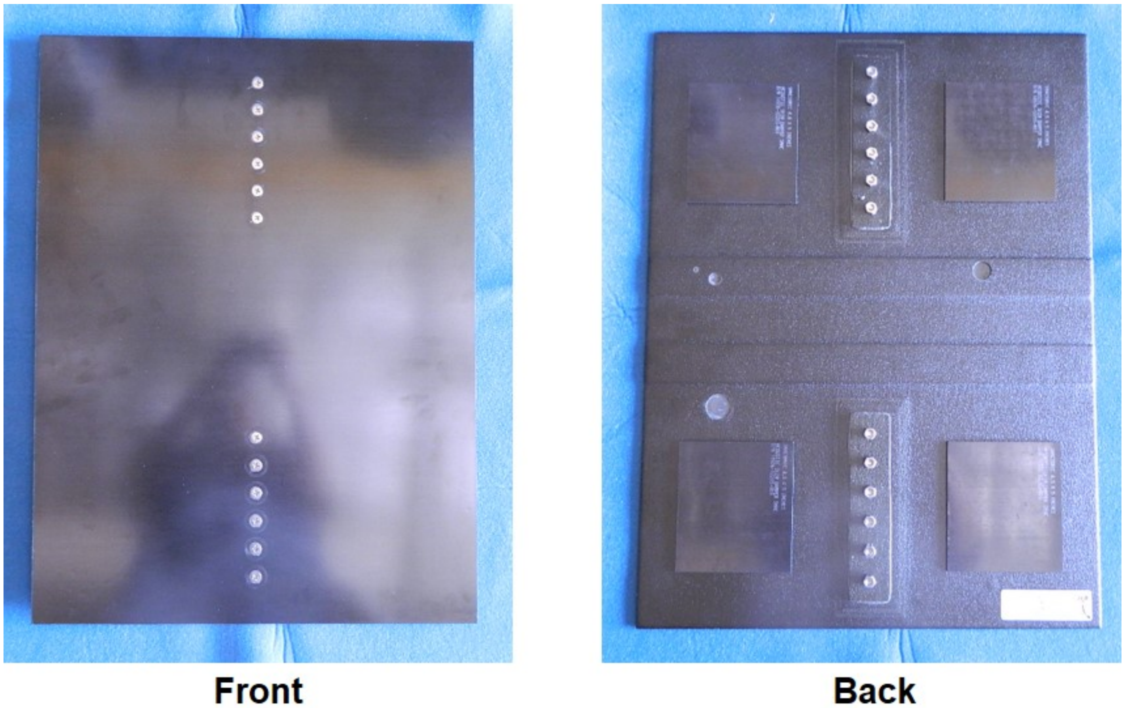


Figure 259. Specimen 2B—Picture

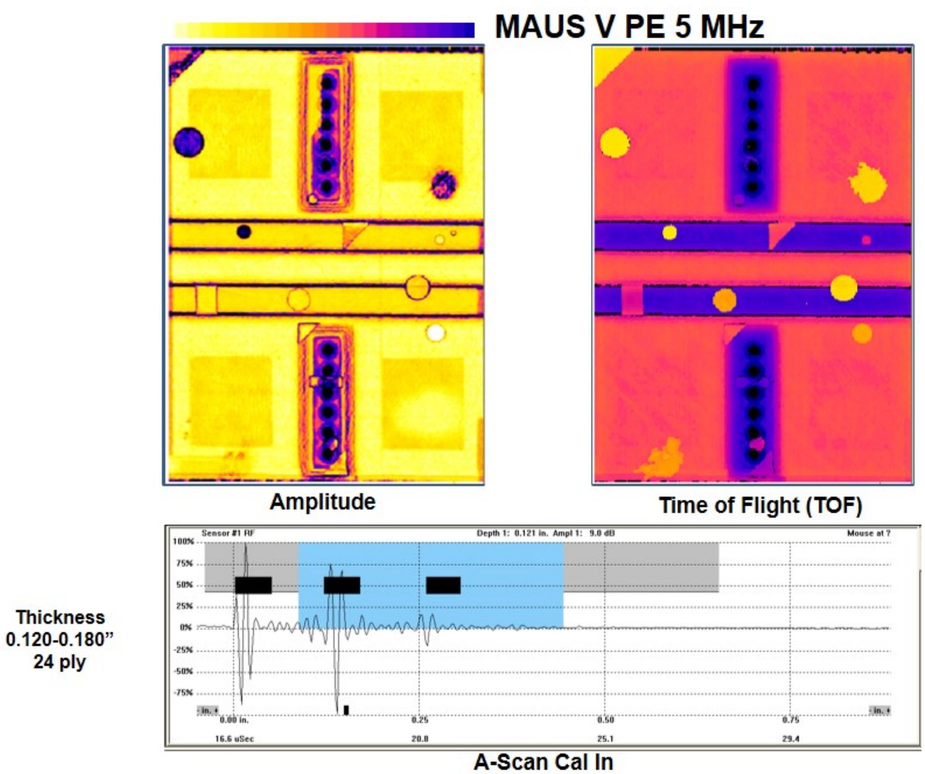


Figure 260. Specimen 2B—PE UT inspection results

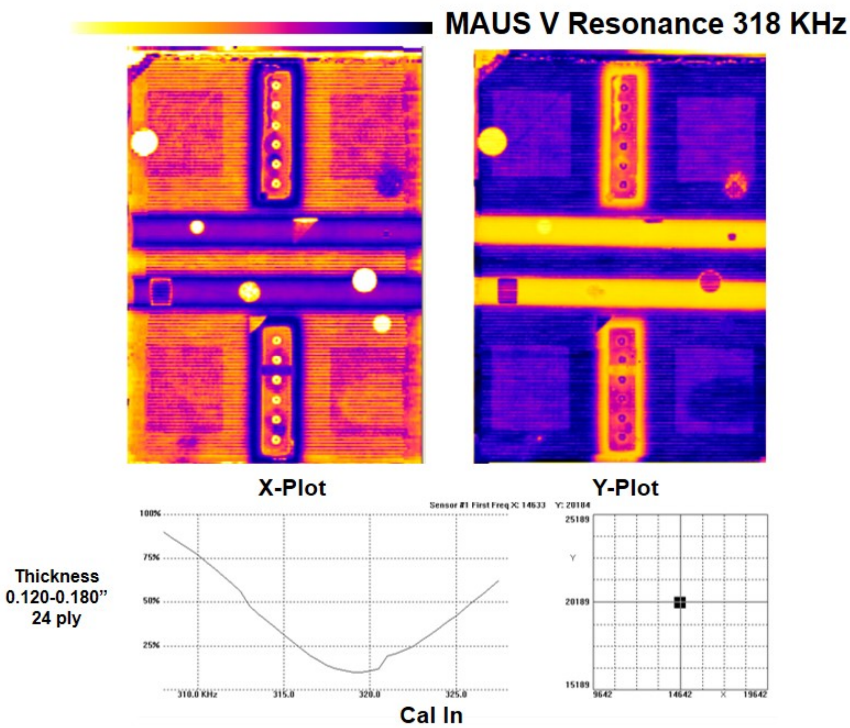


Figure 261. Specimen 2B—Resonance inspection results

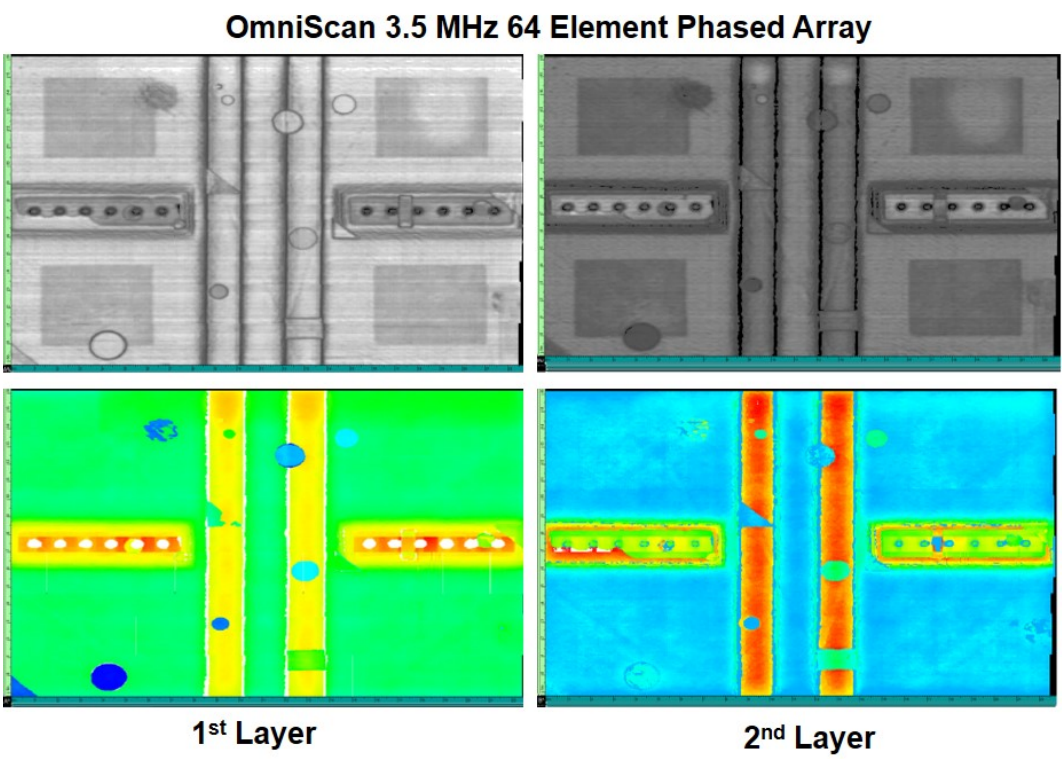


Figure 262. Specimen 2B—PA UT inspection results

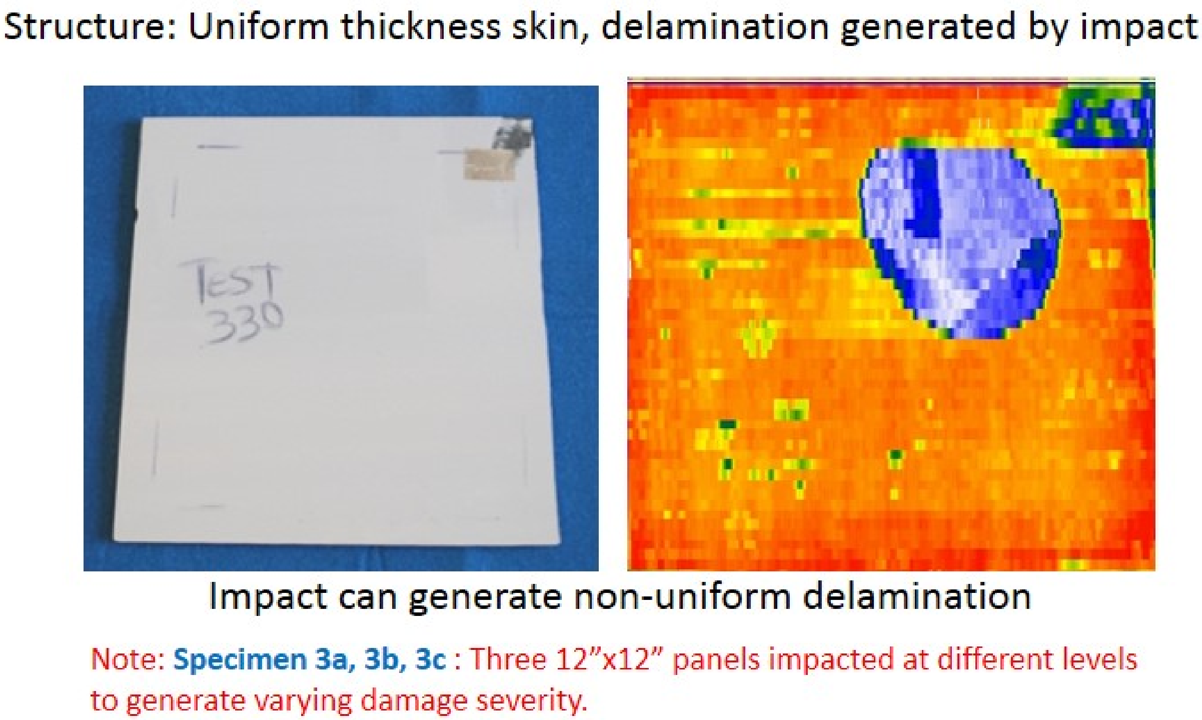


Figure 263. Specimen 3A, 3B, and 3C design

8. MODULE 6: COMPOSITE NDI—INTRODUCTION TO THE HANDS-ON EXERCISES

There are seven specific hands-on exercises developed for the Composite Inspector Training Course designed to be used with the NDI proficiency specimens. The exercises walk the inspector through a composite laminate inspection, reinforcing guidance presented in the lecture portion of the class. A general introduction to the hands-on exercises that can be presented to inspectors prior to the hands-on portion of the course is shown in figures 264–265. All seven of the hands-on exercises are included in this report and can be seen in Appendix C. These include:

1. Calibration—set the material velocity and TCG curve
2. Marking the substructure on the surface of the panel
3. Defect detection in uniform thickness skin
4. Defect detection in tapered skin
5. Inspection of bonded substructure
6. Inspection of co-cured substructure
7. Defect detection around other aircraft elements

The hands-on exercises reference two different composite laminate inspection procedures specifically developed for the course. They are the General Ultrasonic Carbon Laminate Guide (Conventional A-Scan) and the General Ultrasonic Carbon Laminate Guide (Conventional C-Scan). These two procedures are provided in Appendix D.

In practice, once the inspector has completed the exercises applicable to the specimen they are working on, templates are used to lay over the specimen and determine which defects were detected and which ones were missed. The inspection results provided in Module 5 can then be used to explore why the inspector may have missed specific defects.

- Conventional A-scan (2 groups)

A-Scan Exercises

- Inspection will be conducted in the blind mode
- Follow general inspection procedures
- Seven hands-on exercises developed based on guidance from best practices identified in the Solid Laminate POD

A-Scan Exercises	Panels
General A-Scan Inspection Procedure	All panels
1 - Calibration - Set Material Velocity and TCG Curve	Ref Std
2 - Mark substructure on surface	1a,1b,1c,2a,2b
3 - Defect detection in uniform thickness skin	1a,1b,1c,3a,3b,3c
4 - Defect detection in tapered skin	1a,1b,1c
5 - Inspection of bonded substructure	1a,1b,1c
6 - Inspection of co-cured substructure	2a,2b
7 - Defect detection around other aircraft elements	2a,2b

- Phased Array C-Scan (2 groups)

C-Scan Exercises

- Follow general inspection procedures
- Once the scan is complete, identify defects
  - Marking defects may be more difficult
- Test out different gate settings once the scan is saved

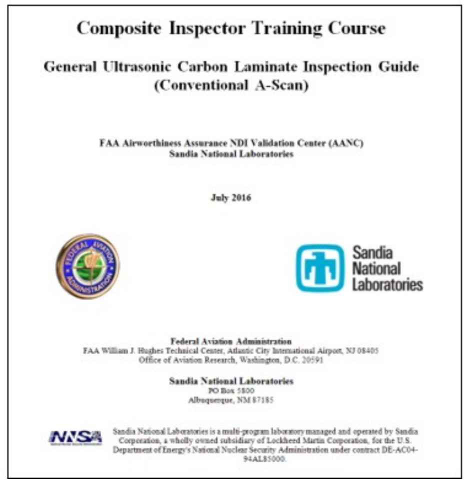
PA Exercises	Panels
General C-Scan Inspection Procedure	All panels
1 - PA Calibration	Ref Std
2 - Set up TCG Curve	ST8872
3 - Setting gates	All panels
4 - Analyzing C-Scan results	All panels

Figure 264. Introduction to hands-on exercises

Exercises

- Work through the exercises
- Keep track of timing on each panel
- Mark flaws on surface of panel following inspection procedures
- Instructor will provide feedback when you have completed each exercise or at the completion of each panel

- Clean markings off panel when complete
- Everyone should get a chance to work the equipment
- Each group should be able to complete all panels using both A-scan and C-scan.


Composite Inspector Training Course

General Ultrasonic Carbon Laminate Inspection Guide
(Conventional A-Scan)

FAA Airworthiness Assurance NDI Validation Center (AANC)
Sandia National Laboratories

July 2016

Federal Aviation Administration
FAA William J. Hughes Technical Center, Atlantic City International Airport, NJ 08405
Office of Aviation Research, Washington, D.C. 20591

Sandia National Laboratories
PO Box 5800
Albuquerque, NM 87185

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Figure 264. Introduction to hands-on exercises (continued)

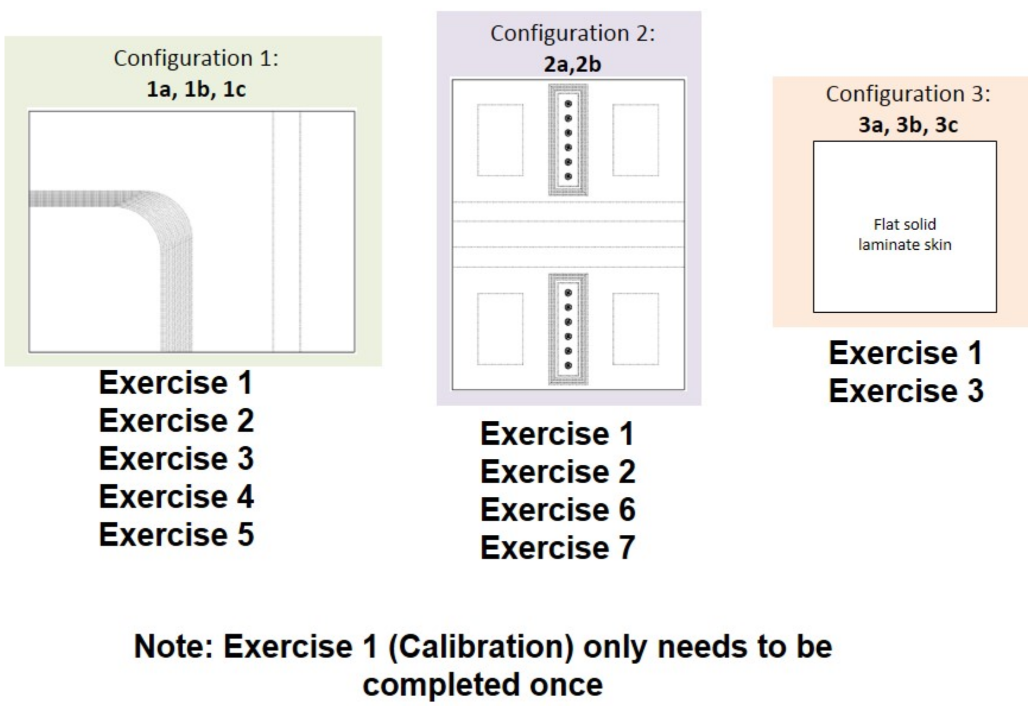

Figure 265. Hands-on exercises flow chart —Applicable exercises for each panel configuration

9. REFERENCES

1. Melton, A., Neidigk, S., Roach, D. (2016). *An Airline Perspective on Meeting the Training Needs for NDI of Composites*. Proceedings from the Airlines for America Nondestructive Testing Forum, San Diego, CA.

2. Neidigk, S., Roach, D., Melton, A. (2016). *A Composite NDI Training Course to Address the Growing Need for Extensive Composite Laminate Inspections*. Proceedings from the Airlines for America Nondestructive Testing Forum, San Diego, CA.

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4. Soutis, C.J. (2005). Fibre Reinforced Composites in Aircraft Construction. *Progress in Aerospace Sciences, 41*(2), 143–151.

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14. Roach, D., Neidigk, S., Roach, D, Rice, T., Duvall, R. (2014). *A Comparison of Inspection Methods for Assessing Composite Laminate Impact Damage in the Failure Threshold-to-Barely Visible Range*. Proceedings from the FAA/DOD/NASA Aircraft Airworthiness & Sustainment Conference, Baltimore, MD.

15. Department of Transportation Report. (2012). Safety Research Corporation of America (SRCA), ("FAA-H-8083-31, 2012). Aviation Maintenance Technician Handbook – Airframe, Volume1.

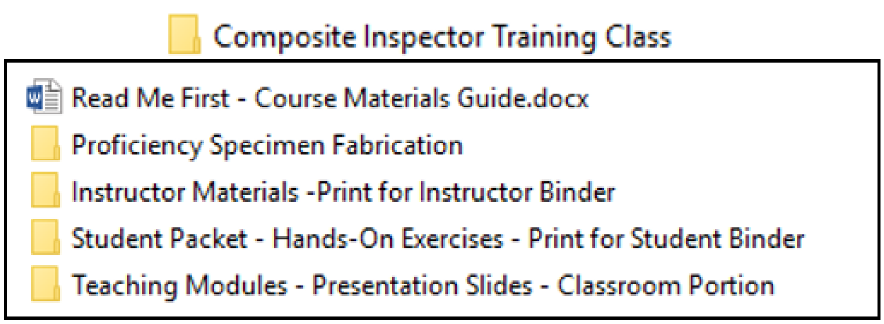
APPENDIX A—COMPOSITE INSPECTOR TRAINING COURSE MATERIAL GUIDE

Composite Inspector Training Course Materials Guide

Sandia National Laboratories
FAA Airworthiness Assurance Center
Stephen Neidigk
sneidig@sandia.gov; (505)284-2200
Dennis Roach
dproach@sandia.gov; (505)844-6078
Tom Rice
tmrice@sandia.gov; (505)844-7738

Thank you for your interest in the FAA/Sandia National Labs Composite Inspector Training Course. This document is a brief description of the file structure to help you navigate the classroom and hands-on materials developed for the class.

You have already located this "Read Me First" document within the Composite Inspector Training Class folder shown below. Also contained within this folder are four additional folders containing all of the course materials. They consist of the "Proficiency Specimen Fabrication," "Instructor Materials," "Student Packet," and "Teaching Modules" folders.



The "Teaching Modules" folder contains all six of the course modules for the class. This is the classroom lecture portion of the class and is intended to be taught in numerical order. It is recommended the hands-on portion follow the classroom portion of the class. The contents of the "Teaching Modules" folder are shown below.

Teaching Modules - Presentation Slides - Classroom Portion

- Module 1- Introduction.pptx
- Module 2 - Composite Materials Awareness.pptx
- Module 3 - Composite NDI Theory and Practice.pptx
- Module 4 - Special Cases and Lessons Learned.pptx
- Module 5 - NDI Proficiency Specimens.pptx
- Module 6 - Intro to Hands-On Exercises.pptx

The "Proficiency Specimen Fabrication" folder is used in combination with the hands-on exercises. It is critical that a set of the NDI proficiency specimens be fabricated and accessible to inspectors prior to teaching this course. The set of specimens can either be built to order from NORDAM Interiors and Structures (contact Daryl Graham, dgraham@nordam.com) or at another composite manufacturing facility using drawings provided in the "Proficiency Specimen Fabrication" folder.

Proficiency Specimen Fabrication

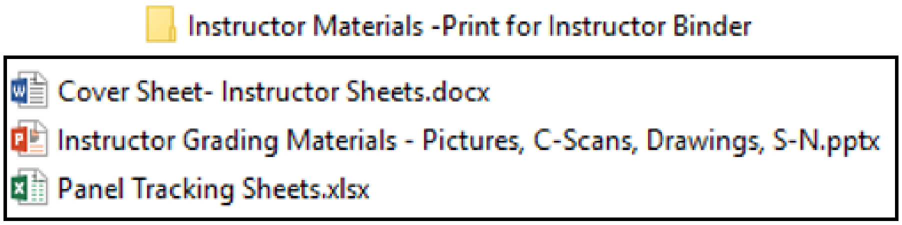
- Engineered Defect Descriptions.pptx
- Proficiency Specimen Fabrication Drawings.pptx

The "Student Packet" folder contains all seven of the hands-on exercises to be used with the proficiency specimens and is shown below. It also contains inspection procedures for A-scan and C-scan inspections, and structural configuration drawings used during the hands-on portion of the class. It is recommended that these materials be printed out and put in a binder for the inspector.

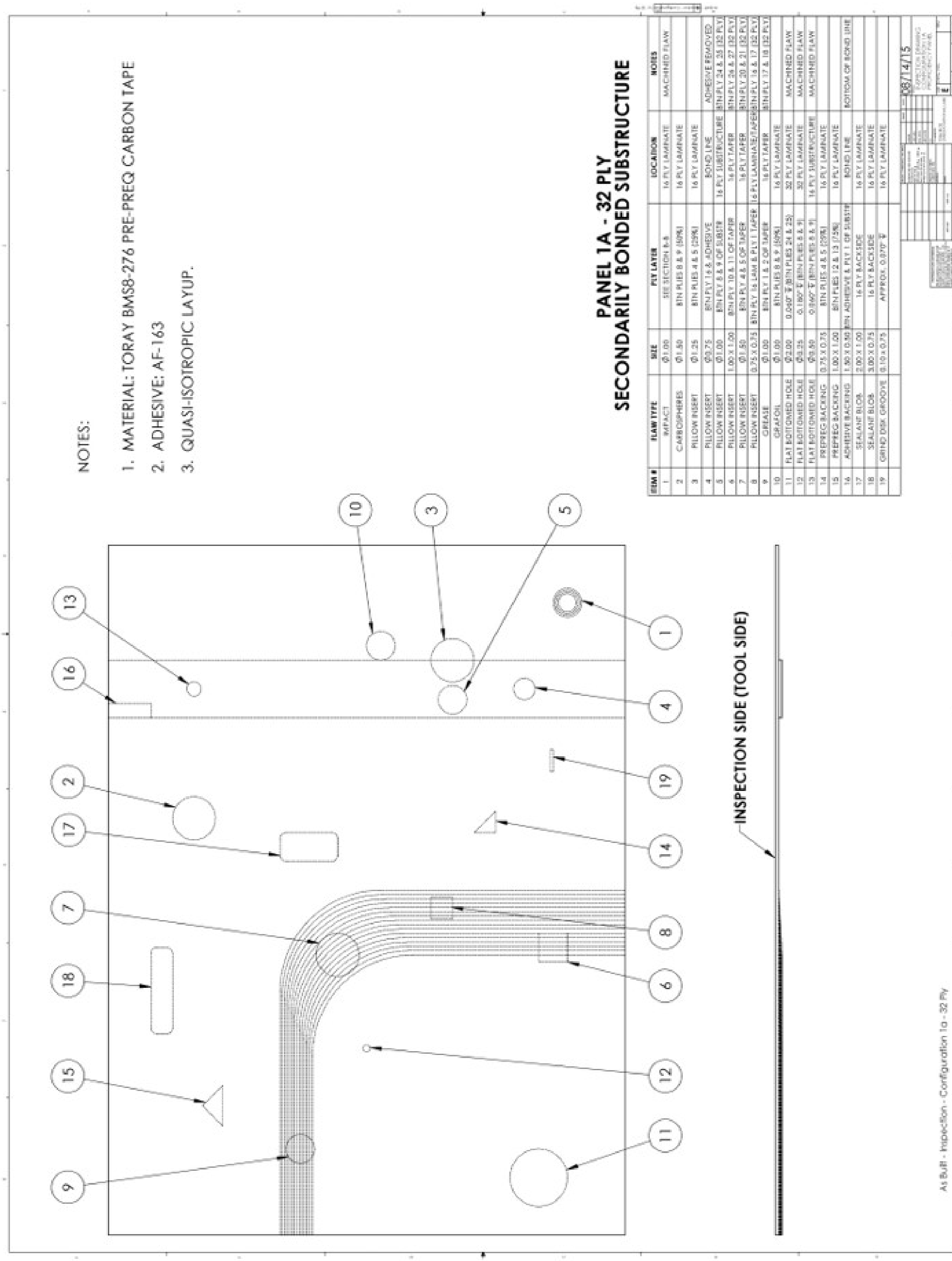
Student Packet - Hands-On Exercises - Print for Student Binder

- Cover Sheet- Hands-on Exercises.docx
- Exercise 1 - Calibration - Mat Velocity.doc
- Exercise 2 - Mark Substructure on Surface.doc
- Exercise 3 - Defect Detection in Uniform Thickness Skin.doc
- Exercise 4 - Defect Detection in Tapered Skin.doc
- Exercise 5 - Inspection of Bonded Substructures.doc
- Exercise 6 - Inspection of Co-Cured Substructure.doc
- Exercise 7 - Defect Detection Around Other Aircraft Elements.doc
- General A-Scan Procedure.docx
- General C-Scan Procedure.docx
- Proficiency Specimen Part Drawings for the Inspector.pptx

The "Instructor Materials" folder contains resources necessary for grading and providing inspectors feedback when they are finished inspecting each proficiency specimen. The "Instructor Grading Materials" file provides pictures of each proficiency specimen with the engineered defects marked onto the surface of each panel, C-scan inspection images, and signal-to-noise data for each defect contained in the proficiency specimens. This information is intended to be used for grading and providing inspectors feedback when they are finished inspecting each panel. It is recommended that these materials be printed out and put in a binder for the instructor. The contents of the “Instructor Materials” folder are shown below.



FAA Program Manager:
David Westlund
David.Westlund@faa.gov
(609) 485-4923



ITEM #	FLAW TYPE	SIZE	PLY LAYER	LOCATION	NOTES
1	IMPACT	Ø1.00	SEE SECTION B-B	16 PLY LAMINATE	MACHINED FLAW
2	CARBOSPHERES	Ø1.50	BTN PLIES 8 & 9 (50%)	16 PLY LAMINATE	
3	PILLOW INSERT	Ø1.25	BTN PLIES 4 & 5 (25%)	16 PLY LAMINATE	
4	PILLOW INSERT	Ø0.75	BTN PLY 16 & ADHESIVE	BOND LINE	ADHESIVE REMOVED
5	PILLOW INSERT	Ø1.00	BTN PLY 8 & 9 OF SUBSTR	16 PLY SUBSTRUCTURE	BTN PLY 24 & 25 (32 PLY)
6	PILLOW INSERT	1.00 X 1.00	BTN PLY 10 & 11 OF TAPER	16 PLY TAPER	BTN PLY 26 & 27 (32 PLY)
7	PILLOW INSERT	Ø1.50	BTN PLY 4 & 5 OF TAPER	16 PLY TAPER	BTN PLY 20 & 21 (32 PLY)
8	PILLOW INSERT	0.75 X 0.75	BTN PLY 16 LAM & PLY 1 TAPER	16 PLY LAMINATE/TAPER	BTN PLY 16 & 17 (32 PLY)
9	GREASE	Ø1.00	BTN PLY 1 & 2 OF TAPER	16 PLY TAPER	BTN PLY 17 & 18 (32 PLY)
10	GRAFOIL	Ø1.00	BTN PLIES 8 & 9 (50%)	16 PLY LAMINATE	
11	FLAT BOTTOMED HOLE	Ø2.00	0.060" ↧ (BTN PLIES 24 & 25)	32 PLY LAMINATE	MACHINED FLAW
12	FLAT BOTTOMED HOLE	Ø0.25	0.180" ↧ (BTN PLIES 8 & 9)	32 PLY LAMINATE	MACHINED FLAW
13	FLAT BOTTOMED HOLE	Ø0.50	0.060" ↧ (BTN PLIES 8 & 9)	16 PLY SUBSTRUCTURE	MACHINED FLAW
14	PREPREG BACKING	0.75 X 0.75	BTN PLIES 4 & 5 (25%)	16 PLY LAMINATE	
15	PREPREG BACKING	1.00 X 1.00	BTN PLIES 12 & 13 (75%)	16 PLY LAMINATE	
16	ADHESIVE BACKING	1.50 X 0.50	BTN ADHESIVE & PLY 1 OF SUBSTR	BOND LINE	BOTTOM OF BOND LINE
17	SEALANT BLOB	2.00 X 1.00	16 PLY BACKSIDE	16 PLY LAMINATE	
18	SEALANT BLOB	3.00 X 0.75	16 PLY BACKSIDE	16 PLY LAMINATE	
19	GRIND DISK GROOVE	0.10 x 0.75	APPROX. 0.070" ↧	16 PLY LAMINATE	

Figure B-1. Panel 1A—Schematic and defect descriptions

PANEL 1A - 32 PLY
SECONDARILY BONDED SUBSTRUCTURE

ITEM #	FLAW TYPE	SIZE	PLY LAYER	LOCATION	NOTES
1	IMPACT	∅1.00	SEE SECTION B-B	16 PLY LAMINATE	MACHINED FLAW
2	CARBOSPHERES	∅1.50	BTN PLIES 8 & 9 (50%)	16 PLY LAMINATE	
3	PILLOW INSERT	∅1.25	BTN PLIES 4 & 5 (25%)	16 PLY LAMINATE	
4	PILLOW INSERT	∅0.75	BTN PLY 16 & ADHESIVE	BOND LINE	ADHESIVE REMOVED
5	PILLOW INSERT	∅1.00	BTN PLY 8 & 9 OF SUBSTR	16 PLY SUBSTRUCTURE	BTN PLY 24 & 25 (32 PLY)
6	PILLOW INSERT	1.00 X 1.00	BTN PLY 10 & 11 OF TAPER	16 PLY TAPER	BTN PLY 26 & 27 (32 PLY)
7	PILLOW INSERT	∅1.50	BTN PLY 4 & 5 OF TAPER	16 PLY TAPER	BTN PLY 20 & 21 (32 PLY)
8	PILLOW INSERT	0.75 X 0.75	BTN PLY 16 LAM & PLY 1 TAPER	16 PLY LAMINATE/TAPER	BTN PLY 16 & 17 (32 PLY)
9	GREASE	∅1.00	BTN PLY 1 & 2 OF TAPER	16 PLY TAPER	BTN PLY 17 & 18 (32 PLY)
10	GRAFOIL	∅1.00	BTN PLIES 8 & 9 (50%)	16 PLY LAMINATE	
11	FLAT BOTTOMED HOLE	∅2.00	0.060" ⫫ (BTN PLIES 24 & 25)	32 PLY LAMINATE	MACHINED FLAW
12	FLAT BOTTOMED HOLE	∅0.25	0.180" ⫫ (BTN PLIES 8 & 9)	32 PLY LAMINATE	MACHINED FLAW
13	FLAT BOTTOMED HOLE	∅0.50	0.060" ⫫ (BTN PLIES 8 & 9)	16 PLY SUBSTRUCTURE	MACHINED FLAW
14	PREPREG BACKING	0.75 X 0.75	BTN PLIES 4 & 5 (25%)	16 PLY LAMINATE	
15	PREPREG BACKING	1.00 X 1.00	BTN PLIES 12 & 13 (75%)	16 PLY LAMINATE	
16	ADHESIVE BACKING	1.50 X 0.50	BTN ADHESIVE & PLY 1 OF SUBSTR	BOND LINE	BOTTOM OF BOND LINE
17	SEALANT BLOB	2.00 X 1.00	16 PLY BACKSIDE	16 PLY LAMINATE	
18	SEALANT BLOB	3.00 X 0.75	16 PLY BACKSIDE	16 PLY LAMINATE	
19	GRIND DISK GROOVE	0.10 x 0.75	APPROX. 0.070" ⫫	16 PLY LAMINATE	

Figure B-2. Panel 1A—Defect type, size, layer, and location

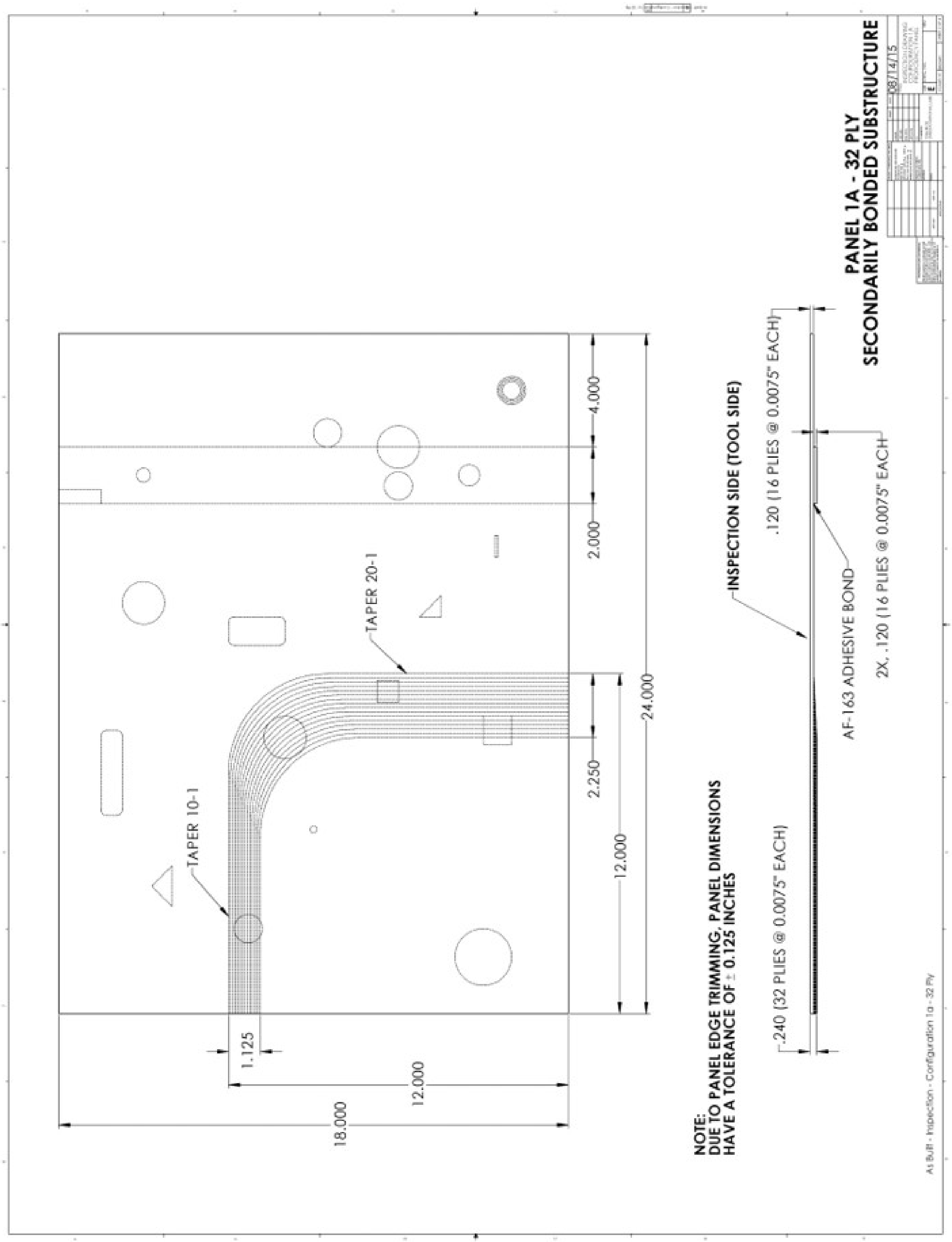


Figure B-3. Panel 1A—Substructure locations

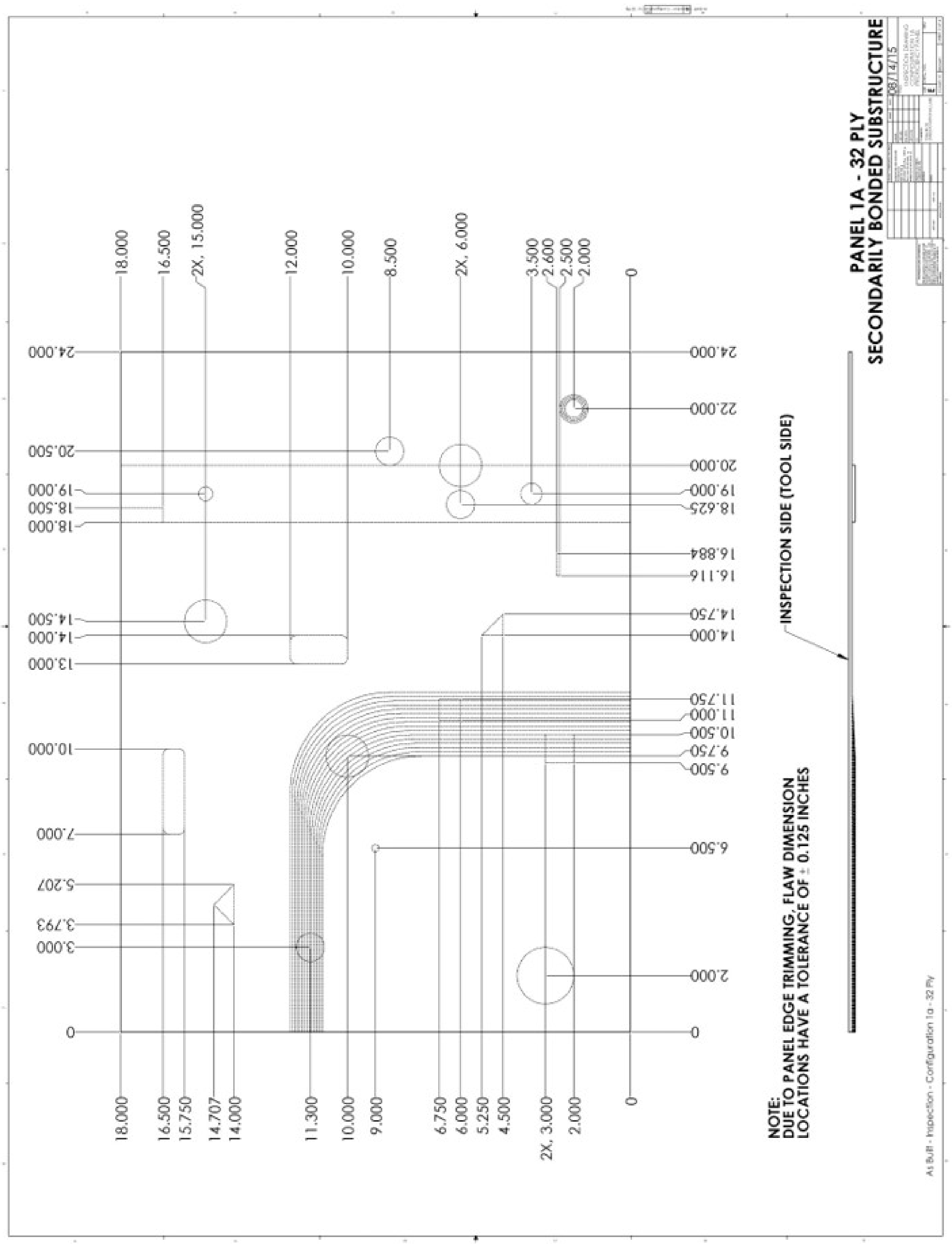


Figure B-4. Panel 1A—Defect locations

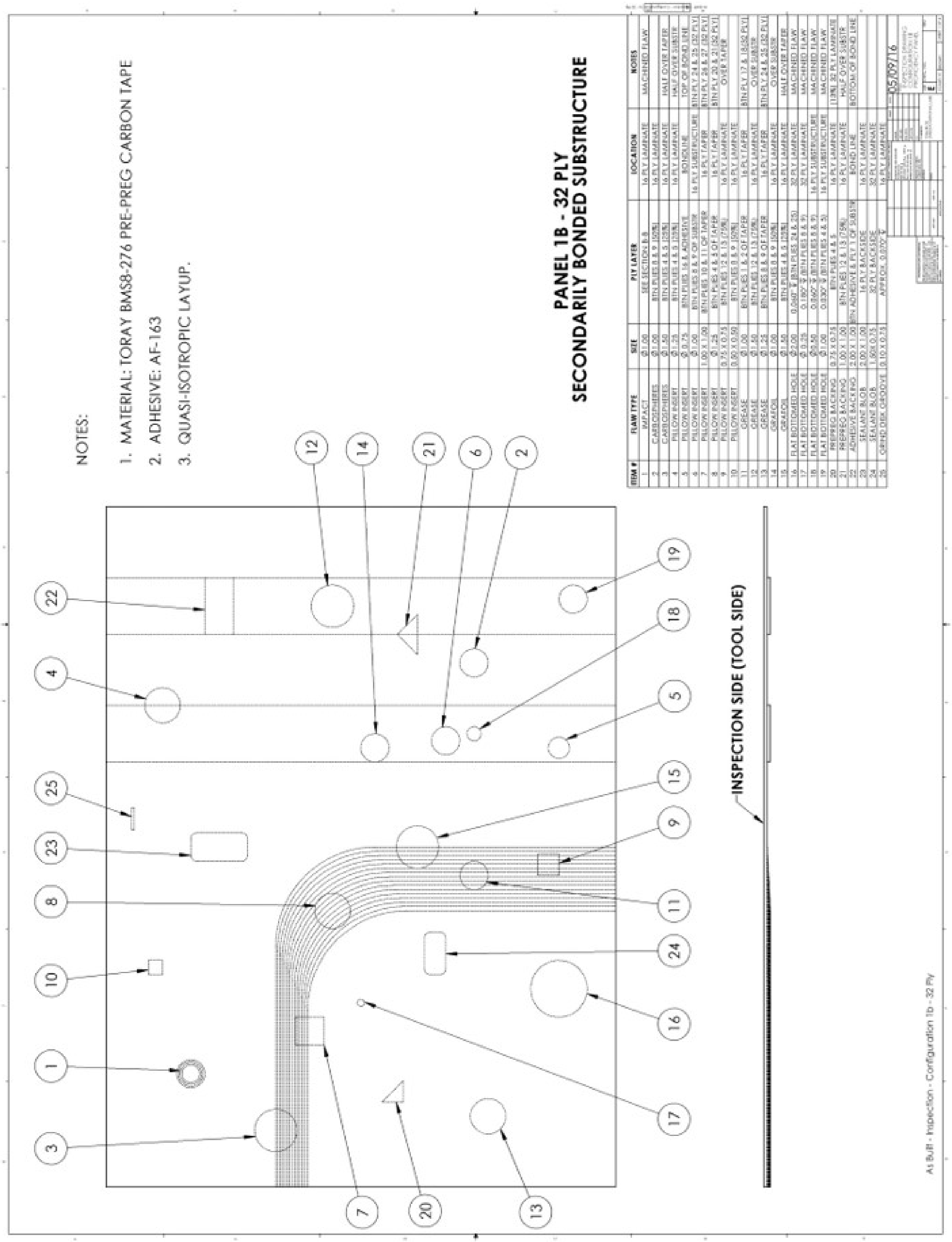


Figure B-5. Panel 1B—Schematic and defect descriptions

PANEL 1B - 32 PLY
SECONDARILY BONDED SUBSTRUCTURE

ITEM #	FLAW TYPE	SIZE	PLY LAYER	LOCATION	NOTES
1	IMPACT	∅1.00	SEE SECTION B-B	16 PLY LAMINATE	MACHINED FLAW
2	CARBOSPHERES	∅1.00	BTN PLIES 8 & 9 (50%)	16 PLY LAMINATE	
3	CARBOSPHERES	∅1.50	BTN PLIES 4 & 5 (25%)	16 PLY LAMINATE	HALF OVER TAPER
4	PILLOW INSERT	∅1.25	BTN PLIES 4 & 5 (25%)	16 PLY LAMINATE	HALF OVER SUBSTR
5	PILLOW INSERT	∅ 0.75	BTN PLIES 16 & ADHESIVE	BONDLINE	TOP OF BOND LINE
6	PILLOW INSERT	∅1.00	BTN PLIES 8 & 9 OF SUBSTR	16 PLY SUBSTRUCTURE	BTN PLY 24 & 25 (32 PLY)
7	PILLOW INSERT	1.00 X 1.00	BTN PLIES 10 & 11 OF TAPER	16 PLY TAPER	BTN PLY 26 & 27 (32 PLY)
8	PILLOW INSERT	∅1.25	BTN PLIES 4 & 5 OF TAPER	16 PLY TAPER	BTN PLY 20 & 21(32 PLY)
9	PILLOW INSERT	0.75 X 0.75	BTN PLIES 12 & 13 (75%)	16 PLY LAMINATE	OVER TAPER
10	PILLOW INSERT	0.50 X 0.50	BTN PLIES 8 & 9 (50%)	16 PLY LAMINATE	
11	GREASE	∅1.00	BTN PLIES 1 & 2 OF TAPER	16 PLY TAPER	BTN PLY 17 & 18(32 PLY)
12	GREASE	∅1.50	BTN PLIES 12 & 13 (75%)	16 PLY LAMINATE	OVER SUBSTR
13	GREASE	∅1.25	BTN PLIES 8 & 9 OF TAPER	16 PLY TAPER	BTN PLY 24 & 25 (32 PLY)
14	GRAFOIL	∅1.00	BTN PLIES 8 & 9 (50%)	16 PLY LAMINATE	OVER SUBSTR
15	GRAFOIL	∅1.50	BTN PLIES 4 & 5 (25%)	16 PLY LAMINATE	HALF OVER TAPER
16	FLAT BOTTOMED HOLE	∅2.00	0.060" ↧ (BTN PLIES 24 & 25)	32 PLY LAMINATE	MACHINED FLAW
17	FLAT BOTTOMED HOLE	∅ 0.25	0.180" ↧ (BTN PLIES 8 & 9)	32 PLY LAMINATE	MACHINED FLAW
18	FLAT BOTTOMED HOLE	∅0.50	0.060" ↧ (BTN PLIES 8 & 9)	16 PLY SUBSTRUCTURE	MACHINED FLAW
19	FLAT BOTTOMED HOLE	∅1.00	0.030" ↧ (BTN PLIES 4 & 5)	16 PLY SUBSTRUCTURE	MACHINED FLAW
20	PREPREG BACKING	0.75 X 0.75	BTN PLIES 4 & 5	16 PLY LAMINATE	(13%) 32 PLY LAMINATE
21	PREPREG BACKING	1.00 X 1.00	BTN PLIES 12 & 13 (75%)	16 PLY LAMINATE	HALF OVER SUBSTR
22	ADHESIVE BACKING	2.00 X 1.00	BTN ADHESIVE & PLY 1 OF SUBSTR	BOND LINE	BOTTOM OF BOND LINE
23	SEALANT BLOB	2.00 X 1.00	16 PLY BACKSIDE	16 PLY LAMINATE	
24	SEALANT BLOB	1.50X 0.75	32 PLY BACKSIDE	32 PLY LAMINATE	
25	GRIND DISK GROOVE	0.10 X 0.75	APPROX. 0.070" ↧	16 PLY LAMINATE	

Figure B-6. Panel 1B—Defect type, size, layer, and location

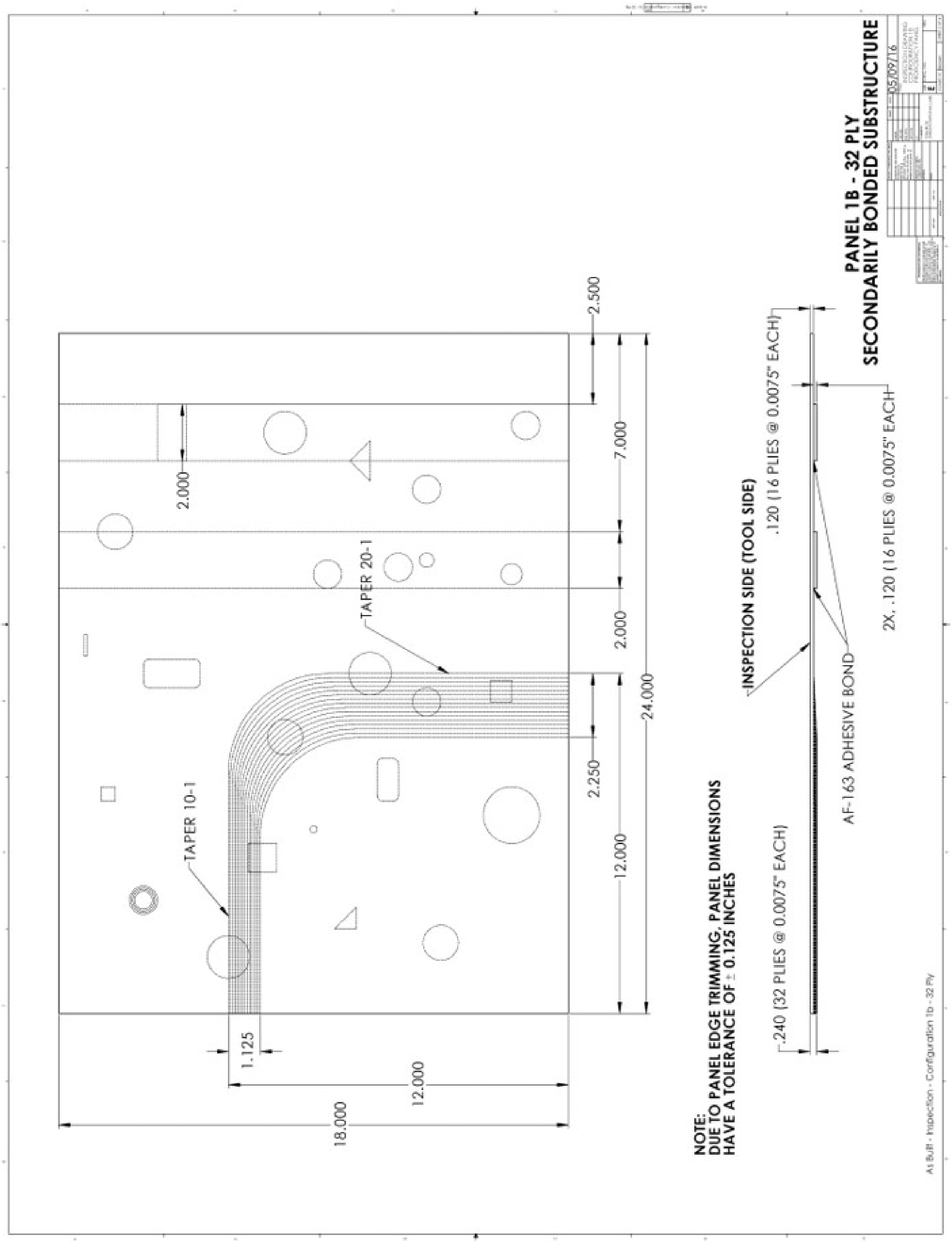


Figure B-7. Panel 1B—Substructure locations

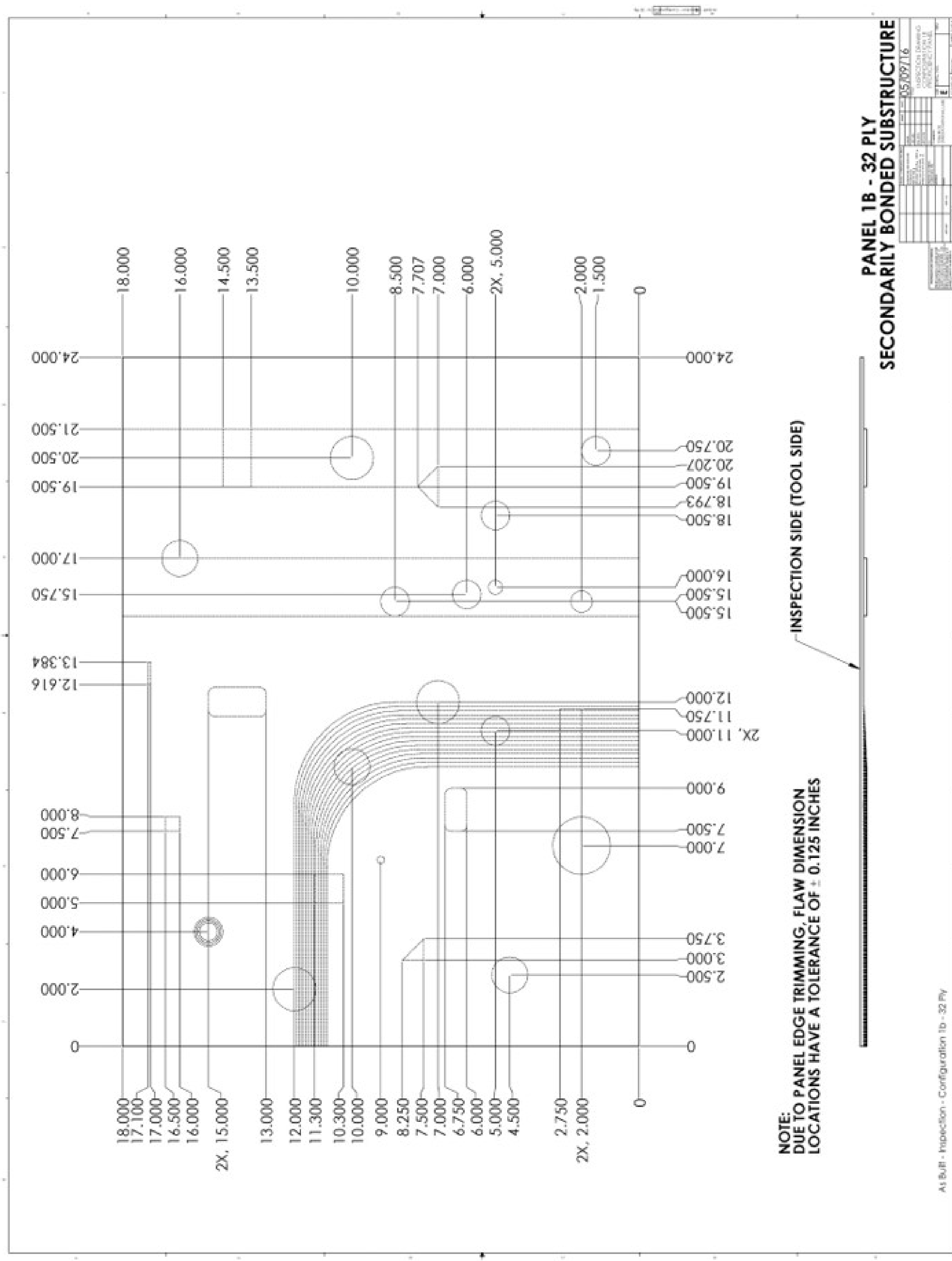
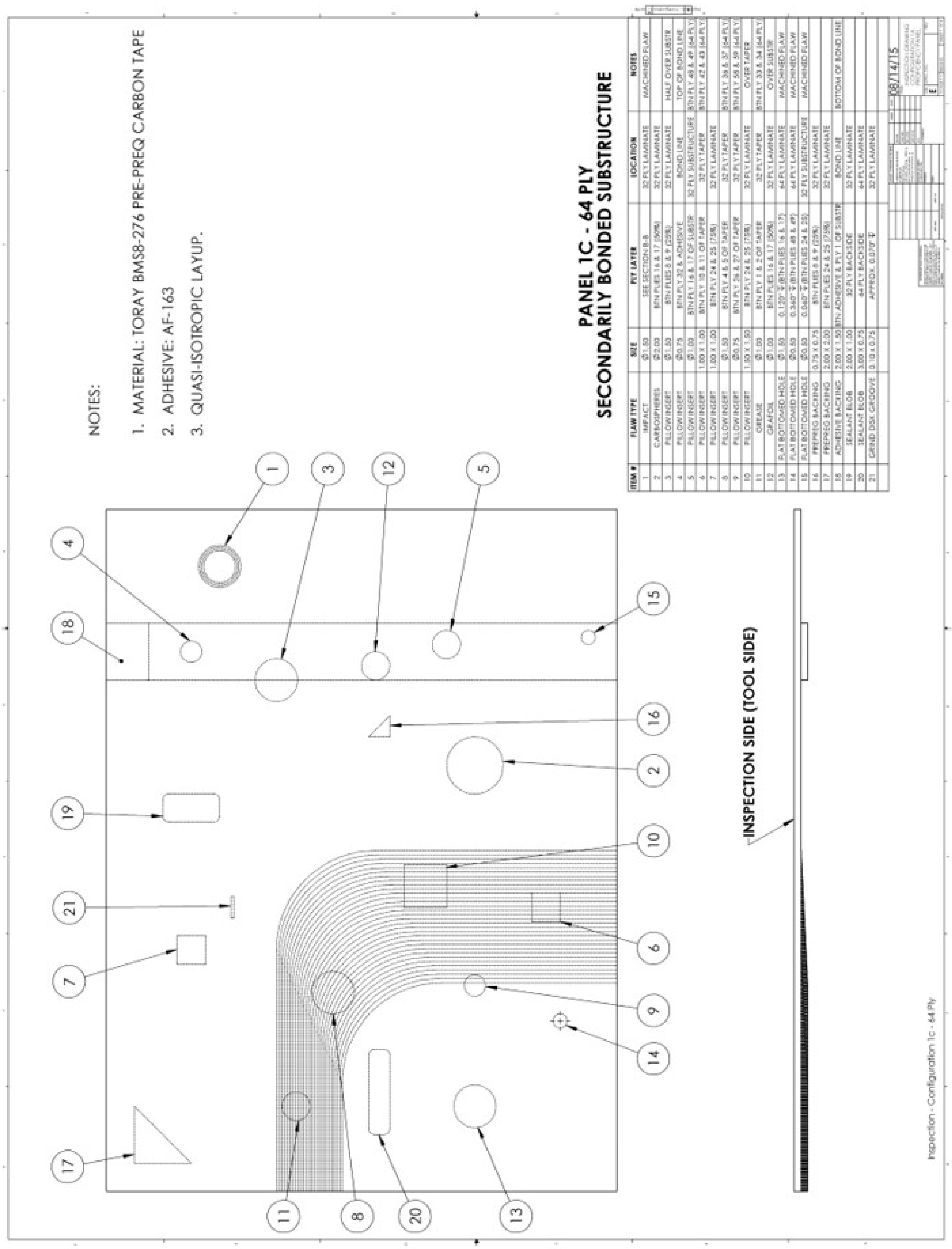


Figure B-8. Panel 1B—Defect locations



ITEM #	FLAW TYPE	SIZE	PLY LAYER	LOCATION	NOTES
1	IMPACT	Ø1.50	SEE SECTION B-B	32 PLY LAMINATE	MACHINED FLAW
2	CARBOSPHERES	Ø2.00	BTN PLIES 16 & 17 (50%)	32 PLY LAMINATE	
3	PILLOW INSERT	Ø1.50	BTN PLIES 8 & 9 (25%)	32 PLY LAMINATE	HALF OVER SUBSTR
4	PILLOW INSERT	Ø0.75	BTN PLY 32 & ADHESIVE	BOND LINE	TOP OF BOND LINE
5	PILLOW INSERT	Ø1.00	BTN PLY 16 & 17 OF SUBSTR	32 PLY SUBSTRUCTURE	BTN PLY 48 & 49 (64 PLY)
6	PILLOW INSERT	1.00 X 1.00	BTN PLY 10 & 11 OF TAPER	32 PLY TAPER	BTN PLY 42 & 43 (64 PLY)
7	PILLOW INSERT	1.00 X 1.00	BTN PLY 24 & 25 (75%)	32 PLY LAMINATE	
8	PILLOW INSERT	Ø1.50	BTN PLY 4 & 5 OF TAPER	32 PLY TAPER	BTN PLY 36 & 37 (64 PLY)
9	PILLOW INSERT	Ø0.75	BTN PLY 26 & 27 OF TAPER	32 PLY TAPER	BTN PLY 58 & 59 (64 PLY)
10	PILLOW INSERT	1.50 X 1.50	BTN PLY 24 & 25 (75%)	32 PLY LAMINATE	OVER TAPER
11	GREASE	Ø1.00	BTN PLY 1 & 2 OF TAPER	32 PLY TAPER	BTN PLY 33 & 34 (64 PLY)
12	GRAFOIL	Ø1.00	BTN PLIES 16 & 17 (50%)	32 PLY LAMINATE	OVER SUBSTR
13	FLAT BOTTOMED HOLE	Ø1.50	0.120" ↧(BTN PLIES 16 & 17)	64 PLY LAMINATE	MACHINED FLAW
14	FLAT BOTTOMED HOLE	Ø0.50	0.360" ↧(BTN PLIES 48 & 49)	64 PLY LAMINATE	MACHINED FLAW
15	FLAT BOTTOMED HOLE	Ø0.50	0.060" ↧(BTN PLIES 24 & 25)	32 PLY SUBSTRUCTURE	MACHINED FLAW
16	PREPREG BACKING	0.75 X 0.75	BTN PLIES 8 & 9 (25%)	32 PLY LAMINATE	
17	PREPREG BACKING	2.00 X 2.00	BTN PLIES 24 & 25 (75%)	32 PLY LAMINATE	
18	ADHESIVE BACKING	2.00 X 1.50	BTN ADHESIVE & PLY 1 OF SUBSTR	BOND LINE	BOTTOM OF BOND LINE
19	SEALANT BLOB	2.00 X 1.00	32 PLY BACKSIDE	32 PLY LAMINATE	
20	SEALANT BLOB	3.00 X 0.75	64 PLY BACKSIDE	64 PLY LAMINATE	
21	GRIND DISK GROOVE	0.10 x 0.75	APPROX. 0.070" ↧	32 PLY LAMINATE	

Figure B-9. Panel 1C—Schematic and defect descriptions

PANEL 1C - 64 PLY
SECONDARILY BONDED SUBSTRUCTURE

ITEM #	FLAW TYPE	SIZE	PLY LAYER	LOCATION	NOTES
1	IMPACT	Ø1.50	SEE SECTION B-B	32 PLY LAMINATE	MACHINED FLAW
2	CARBOSPHERES	Ø2.00	BTN PLIES 16 & 17 (50%)	32 PLY LAMINATE	
3	PILLOW INSERT	Ø1.50	BTN PLIES 8 & 9 (25%)	32 PLY LAMINATE	HALF OVER SUBSTR
4	PILLOW INSERT	Ø0.75	BTN PLY 32 & ADHESIVE	BOND LINE	TOP OF BOND LINE
5	PILLOW INSERT	Ø1.00	BTN PLY 16 & 17 OF SUBSTR	32 PLY SUBSTRUCTURE	BTN PLY 48 & 49 (64 PLY)
6	PILLOW INSERT	1.00 X 1.00	BTN PLY 10 & 11 OF TAPER	32 PLY TAPER	BTN PLY 42 & 43 (64 PLY)
7	PILLOW INSERT	1.00 X 1.00	BTN PLY 24 & 25 (75%)	32 PLY LAMINATE	
8	PILLOW INSERT	Ø1.50	BTN PLY 4 & 5 OF TAPER	32 PLY TAPER	BTN PLY 36 & 37 (64 PLY)
9	PILLOW INSERT	Ø0.75	BTN PLY 26 & 27 OF TAPER	32 PLY TAPER	BTN PLY 58 & 59 (64 PLY)
10	PILLOW INSERT	1.50 X 1.50	BTN PLY 24 & 25 (75%)	32 PLY LAMINATE	OVER TAPER
11	GREASE	Ø1.00	BTN PLY 1 & 2 OF TAPER	32 PLY TAPER	BTN PLY 33 & 34 (64 PLY)
12	GRAFOIL	Ø1.00	BTN PLIES 16 & 17 (50%)	32 PLY LAMINATE	OVER SUBSTR
13	FLAT BOTTOMED HOLE	Ø1.50	0.120" ↧ (BTN PLIES 16 & 17)	64 PLY LAMINATE	MACHINED FLAW
14	FLAT BOTTOMED HOLE	Ø0.50	0.360" ↧ (BTN PLIES 48 & 49)	64 PLY LAMINATE	MACHINED FLAW
15	FLAT BOTTOMED HOLE	Ø0.50	0.060" ↧ (BTN PLIES 24 & 25)	32 PLY SUBSTRUCTURE	MACHINED FLAW
16	PREPREG BACKING	0.75 X 0.75	BTN PLIES 8 & 9 (25%)	32 PLY LAMINATE	
17	PREPREG BACKING	2.00 X 2.00	BTN PLIES 24 & 25 (75%)	32 PLY LAMINATE	
18	ADHESIVE BACKING	2.00 X 1.50	BTN ADHESIVE & PLY 1 OF SUBSTR	BOND LINE	BOTTOM OF BOND LINE
19	SEALANT BLOB	2.00 X 1.00	32 PLY BACKSIDE	32 PLY LAMINATE	
20	SEALANT BLOB	3.00 X 0.75	64 PLY BACKSIDE	64 PLY LAMINATE	
21	GRIND DISK GROOVE	0.10 x 0.75	APPROX. 0.070" ↧	32 PLY LAMINATE	

Figure B-10. Panel 1C—Defect type, size, layer, and location

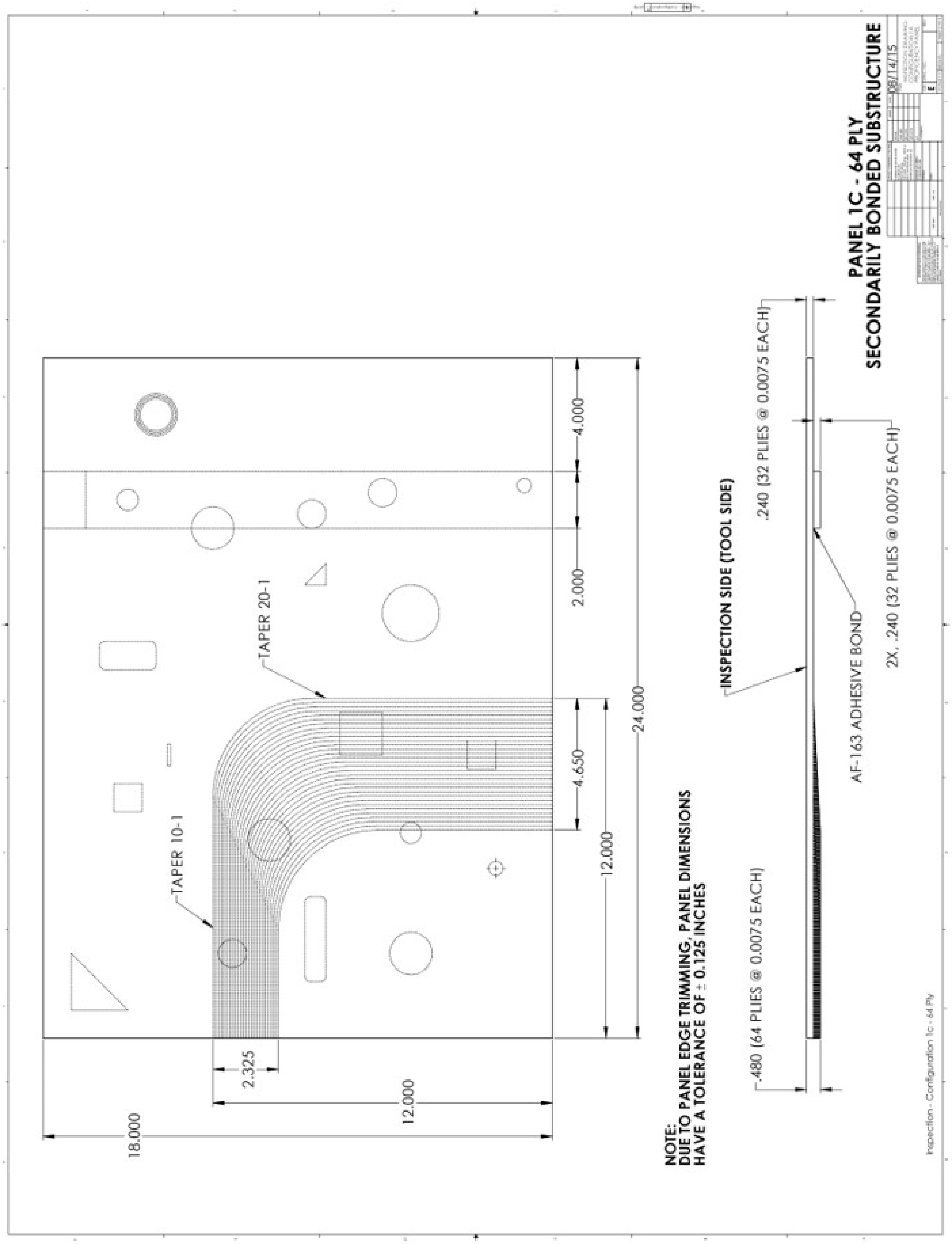


Figure B-11. Panel 1C—Substructure locations

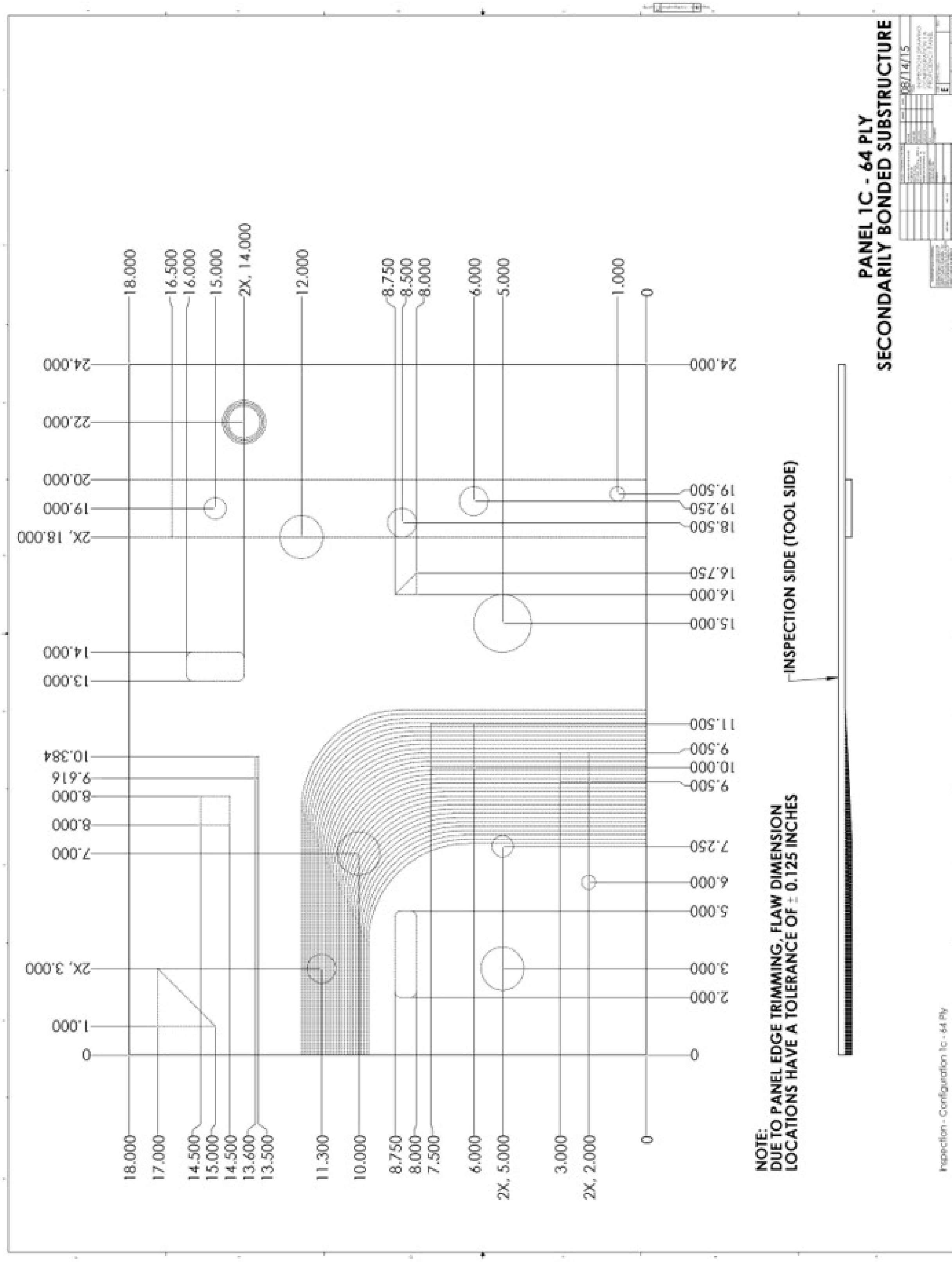
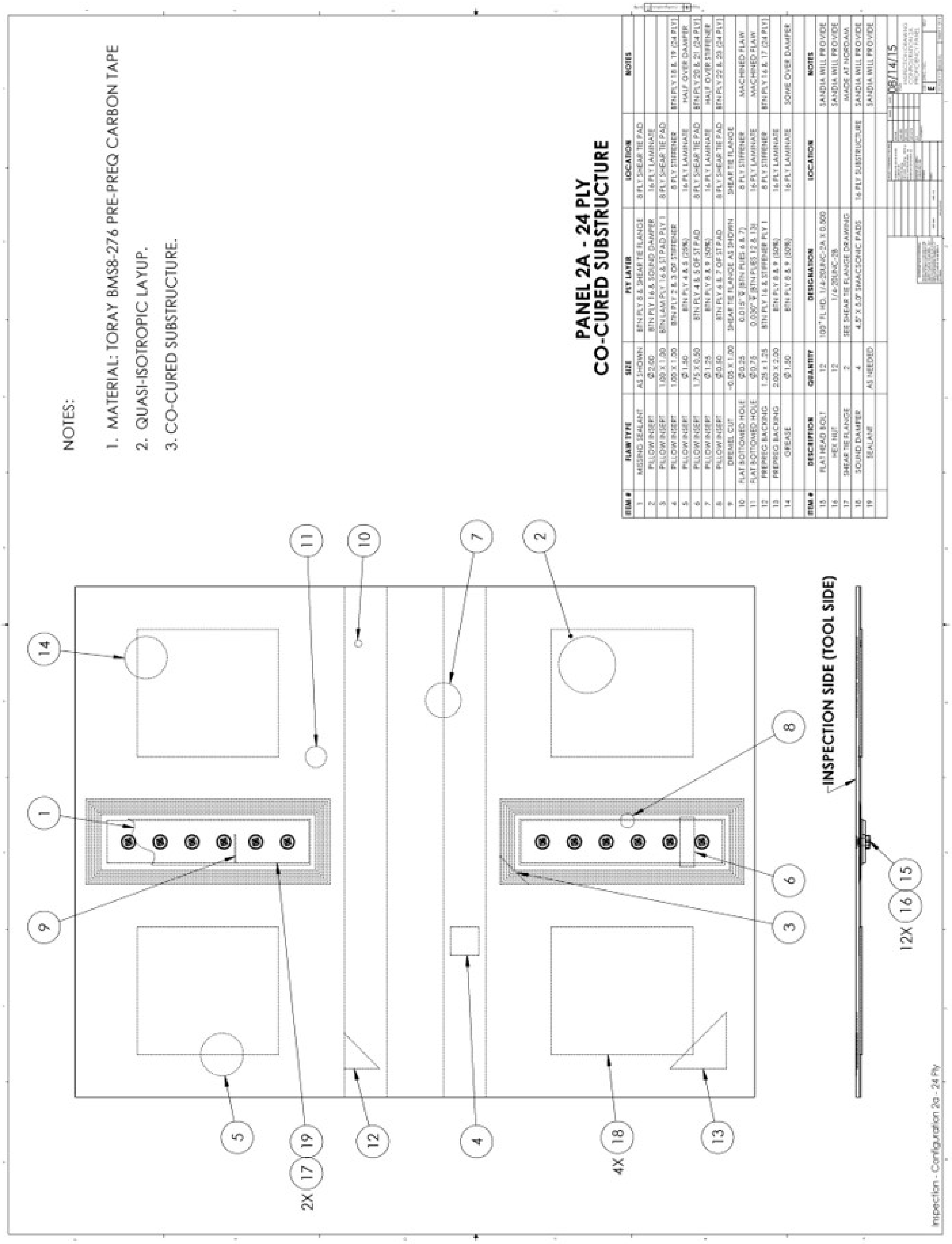


Figure B-12. Panel 1C—Defect locations



ITEM #	FLAW TYPE	SIZE	PLY LAYER	LOCATION	NOTES
1	MISSING SEALANT	AS SHOWN	BTN PLY 8 & SHEAR TIE FLANGE	8 PLY SHEAR TIE PAD	
2	PILLOW INSERT	Ø2.00	BTN PLY 16 & SOUND DAMPER	16 PLY LAMINATE	
3	PILLOW INSERT	1.00 X 1.00	BTN LAM PLY 16 & ST PAD PLY 1	8 PLY SHEAR TIE PAD	
4	PILLOW INSERT	1.00 X 1.00	BTN PLY 2 & 3 OF STIFFENER	8 PLY STIFFENER	BTN PLY 18 & 19 (24 PLY)
5	PILLOW INSERT	Ø1.50	BTN PLY 4 & 5 (25%)	16 PLY LAMINATE	HALF OVER DAMPER
6	PILLOW INSERT	1.75 X 0.50	BTN PLY 4 & 5 OF ST PAD	8 PLY SHEAR TIE PAD	BTN PLY 20 & 21 (24 PLY)
7	PILLOW INSERT	Ø1.25	BTN PLY 8 & 9 (50%)	16 PLY LAMINATE	HALF OVER STIFFENER
8	PILLOW INSERT	Ø0.50	BTN PLY 6 & 7 OF ST PAD	8 PLY SHEAR TIE PAD	BTN PLY 22 & 23 (24 PLY)
9	DREMEL CUT	~0.05 X 1.00	SHEAR TIE FLANGE AS SHOWN	SHEAR TIE FLANGE	
10	FLAT BOTTOMED HOLE	Ø0.25	0.015" ↧ (BTN PLIES 6 & 7)	8 PLY STIFFENER	MACHINED FLAW
11	FLAT BOTTOMED HOLE	Ø0.75	0.030" ↧ (BTN PLIES 12 & 13)	16 PLY LAMINATE	MACHINED FLAW
12	PREPREG BACKING	1.25 x 1.25	BTN PLY 16 & STIFFENER PLY 1	8 PLY STIFFENER	BTN PLY 16 & 17 (24 PLY)
13	PREPREG BACKING	2.00 X 2.00	BTN PLY 8 & 9 (50%)	16 PLY LAMINATE	
14	GREASE	Ø1.50	BTN PLY 8 & 9 (50%)	16 PLY LAMINATE	SOME OVER DAMPER
ITEM #	DESCRIPTION	QUANTITY	DESIGNATION	LOCATION	NOTES
15	FLAT HEAD BOLT	12	100° FL HD. 1/4-20UNC-2A X 0.500		SANDIA WILL PROVIDE
16	HEX NUT	12	1/4-20UNC-2B		SANDIA WILL PROVIDE
17	SHEAR TIE FLANGE	2	SEE SHEAR TIE FLANGE DRAWING		MADE AT NORDAM
18	SOUND DAMPER	4	4.5" X 5.0" SMACSONIC PADS	16 PLY SUBSTRUCTURE	SANDIA WILL PROVIDE
19	SEALANT	AS NEEDED			SANDIA WILL PROVIDE

Figure B-13. Panel 2A—Schematic and defect descriptions

PANEL 2A - 24 PLY
CO-CURED SUBSTRUCTURE

ITEM #	FLAW TYPE	SIZE	PLY LAYER	LOCATION	NOTES
1	MISSING SEALANT	AS SHOWN	BTN PLY 8 & SHEAR TIE FLANGE	8 PLY SHEAR TIE PAD	
2	PILLOW INSERT	∅2.00	BTN PLY 16 & SOUND DAMPER	16 PLY LAMINATE	
3	PILLOW INSERT	1.00 X 1.00	BTN LAM PLY 16 & ST PAD PLY 1	8 PLY SHEAR TIE PAD	
4	PILLOW INSERT	1.00 X 1.00	BTN PLY 2 & 3 OF STIFFENER	8 PLY STIFFENER	BTN PLY 18 & 19 (24 PLY)
5	PILLOW INSERT	∅1.50	BTN PLY 4 & 5 (25%)	16 PLY LAMINATE	HALF OVER DAMPER
6	PILLOW INSERT	1.75 X 0.50	BTN PLY 4 & 5 OF ST PAD	8 PLY SHEAR TIE PAD	BTN PLY 20 & 21 (24 PLY)
7	PILLOW INSERT	∅1.25	BTN PLY 8 & 9 (50%)	16 PLY LAMINATE	HALF OVER STIFFENER
8	PILLOW INSERT	∅0.50	BTN PLY 6 & 7 OF ST PAD	8 PLY SHEAR TIE PAD	BTN PLY 22 & 23 (24 PLY)
9	DREMEL CUT	~0.05 X 1.00	SHEAR TIE FLANGE AS SHOWN	SHEAR TIE FLANGE	
10	FLAT BOTTOMED HOLE	∅0.25	0.015" ↧ (BTN PLIES 6 & 7)	8 PLY STIFFENER	MACHINED FLAW
11	FLAT BOTTOMED HOLE	∅0.75	0.030" ↧ (BTN PLIES 12 & 13)	16 PLY LAMINATE	MACHINED FLAW
12	PREPREG BACKING	1.25 x 1.25	BTN PLY 16 & STIFFENER PLY 1	8 PLY STIFFENER	BTN PLY 16 & 17 (24 PLY)
13	PREPREG BACKING	2.00 X 2.00	BTN PLY 8 & 9 (50%)	16 PLY LAMINATE	
14	GREASE	∅1.50	BTN PLY 8 & 9 (50%)	16 PLY LAMINATE	SOME OVER DAMPER

ITEM #	DESCRIPTION	QUANTITY	DESIGNATION	LOCATION	NOTES
15	FLAT HEAD BOLT	12	100° FL HD, 1/4-20UNC-2A X 0.500		SANDIA WILL PROVIDE
16	HEX NUT	12	1/4-20UNC-2B		SANDIA WILL PROVIDE
17	SHEAR TIE FLANGE	2	SEE SHEAR TIE FLANGE DRAWING		MADE AT NORDAM
18	SOUND DAMPER	4	4.5" X 5.0" SMACSONIC PADS	16 PLY SUBSTRUCTURE	SANDIA WILL PROVIDE
19	SEALANT	AS NEEDED			SANDIA WILL PROVIDE

Figure B-14. Panel 2A—Defect type, size, layer, and location

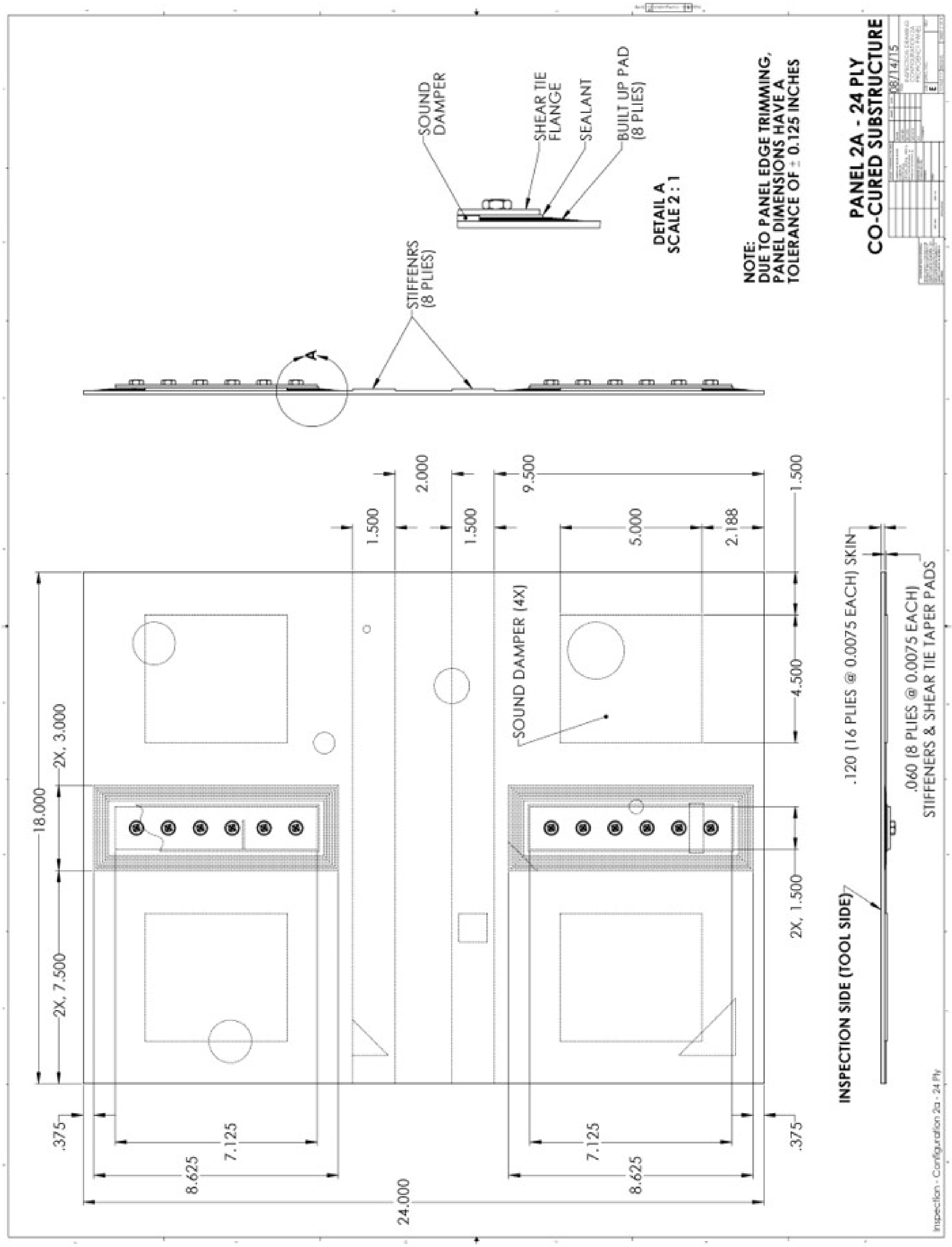


Figure B-15. Panel 2A—Substructure locations

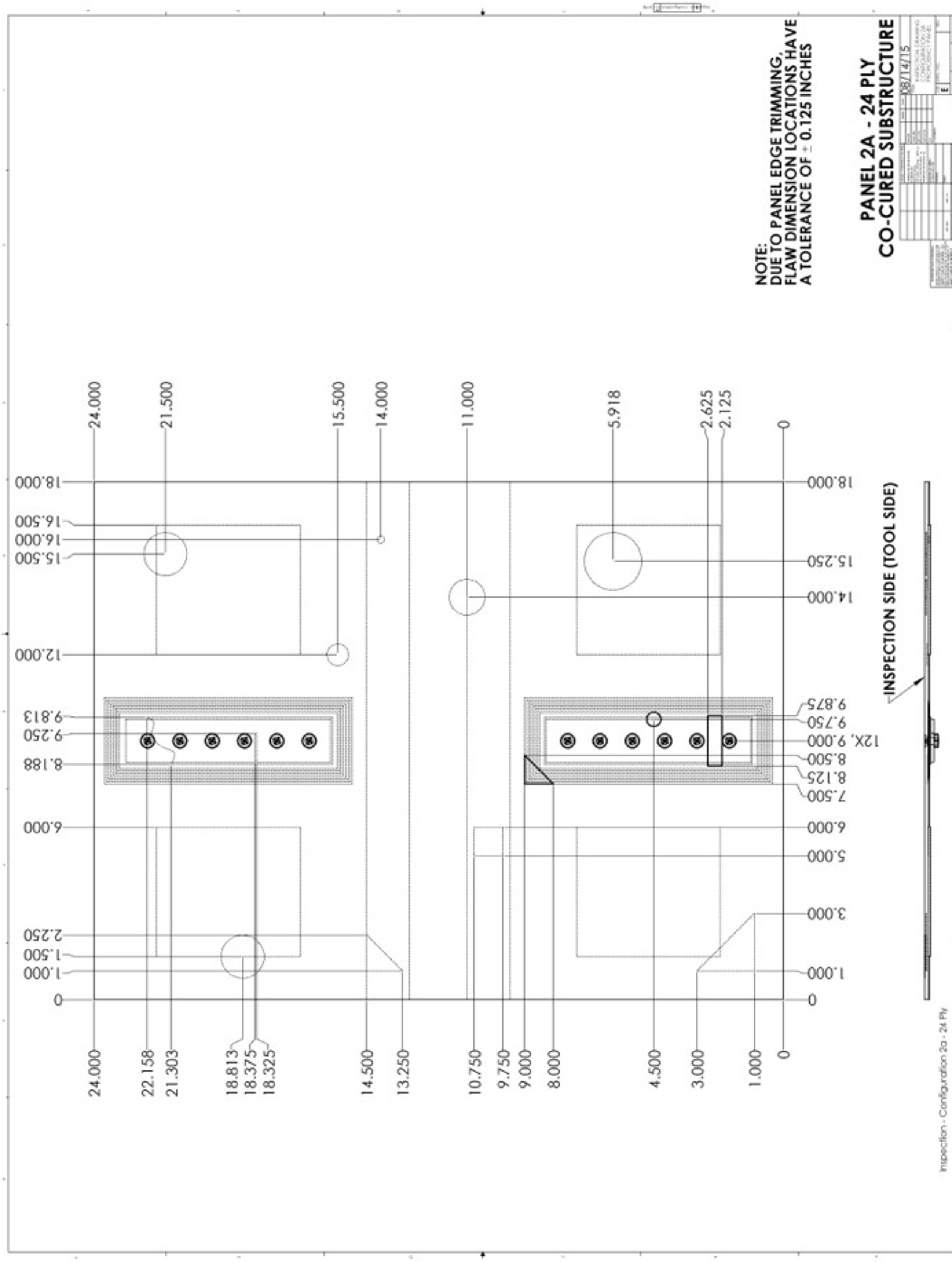


Figure B-16. Panel 2A—Defect locations

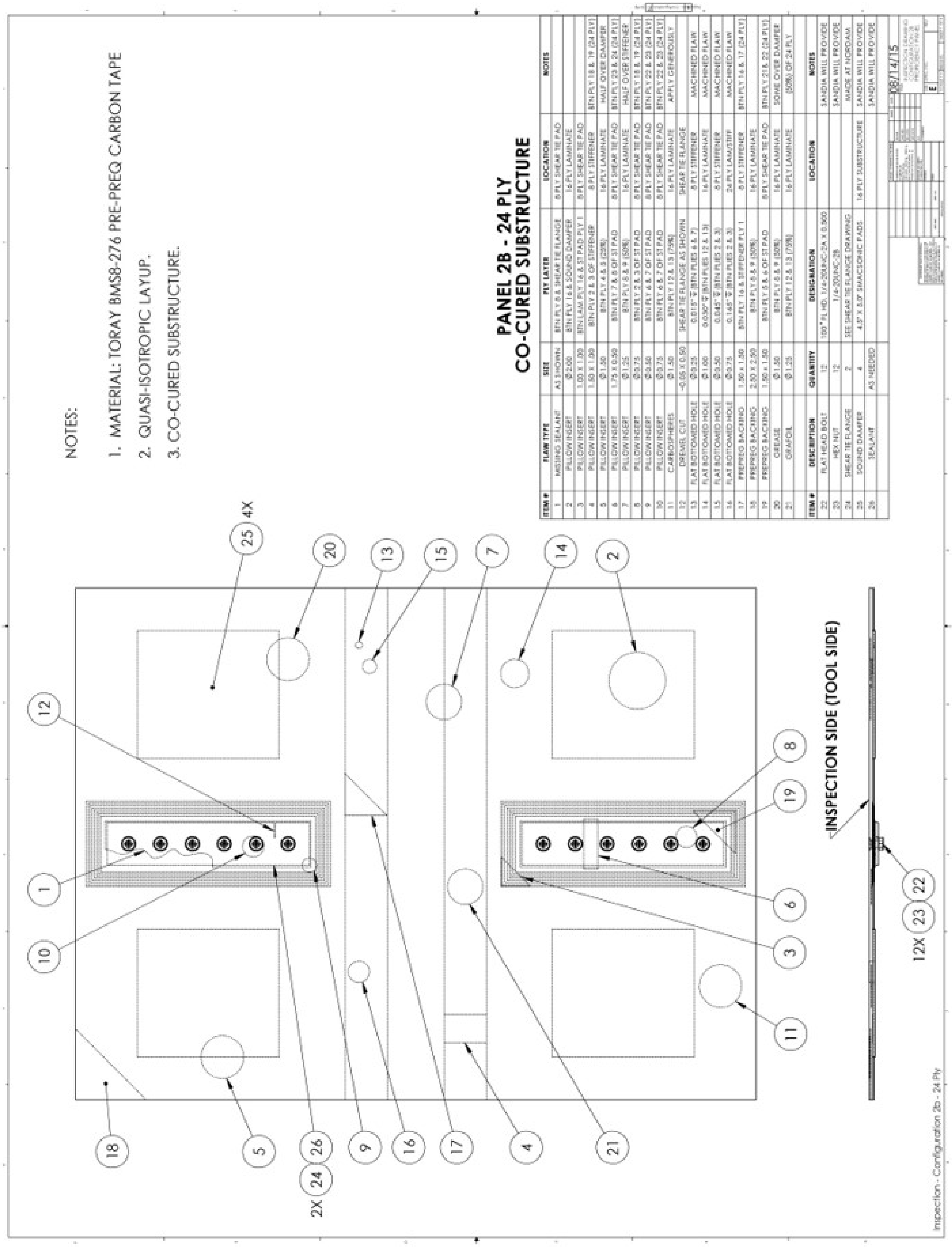


Figure B-17. Panel 2B—Schematic and defect descriptions

PANEL 2B - 24 PLY
CO-CURED SUBSTRUCTURE

ITEM #	FLAW TYPE	SIZE	PLY LAYER	LOCATION	NOTES
1	MISSING SEALANT	AS SHOWN	BTN PLY 8 & SHEAR TIE FLANGE	8 PLY SHEAR TIE PAD	
2	PILLOW INSERT	∅2.00	BTN PLY 16 & SOUND DAMPER	16 PLY LAMINATE	
3	PILLOW INSERT	1.00 X 1.00	BTN LAM PLY 16 & ST PAD PLY 1	8 PLY SHEAR TIE PAD	
4	PILLOW INSERT	1.50 X 1.00	BTN PLY 2 & 3 OF STIFFENER	8 PLY STIFFENER	BTN PLY 18 & 19 (24 PLY)
5	PILLOW INSERT	∅1.50	BTN PLY 4 & 5 (25%)	16 PLY LAMINATE	HALF OVER DAMPER
6	PILLOW INSERT	1.75 X 0.50	BTN PLY 7 & 8 OF ST PAD	8 PLY SHEAR TIE PAD	BTN PLY 23 & 24 (24 PLY)
7	PILLOW INSERT	∅1.25	BTN PLY 8 & 9 (50%)	16 PLY LAMINATE	HALF OVER STIFFENER
8	PILLOW INSERT	∅0.75	BTN PLY 2 & 3 OF ST PAD	8 PLY SHEAR TIE PAD	BTN PLY 18 & 19 (24 PLY)
9	PILLOW INSERT	∅0.50	BTN PLY 6 & 7 OF ST PAD	8 PLY SHEAR TIE PAD	BTN PLY 22 & 23 (24 PLY)
10	PILLOW INSERT	∅0.75	BTN PLY 6 & 7 OF ST PAD	8 PLY SHEAR TIE PAD	BTN PLY 22 & 23 (24 PLY)
11	CARBOSPHERES	∅1.50	BTN PLY 12 & 13 (75%)	16 PLY LAMINATE	APPLY GENEROUSLY
12	DREMEL CUT	~0.05 X 0.50	SHEAR TIE FLANGE AS SHOWN	SHEAR TIE FLANGE	
13	FLAT BOTTOMED HOLE	∅0.25	0.015" ↧ (BTN PLIES 6 & 7)	8 PLY STIFFENER	MACHINED FLAW
14	FLAT BOTTOMED HOLE	∅1.00	0.030" ↧ (BTN PLIES 12 & 13)	16 PLY LAMINATE	MACHINED FLAW
15	FLAT BOTTOMED HOLE	∅0.50	0.045" ↧ (BTN PLIES 2 & 3)	8 PLY STIFFENER	MACHINED FLAW
16	FLAT BOTTOMED HOLE	∅0.75	0.165" ↧ (BTN PLIES 2 & 3)	24 PLY LAM/STIFF	MACHINED FLAW
17	PREPREG BACKING	1.50 x 1.50	BTN PLY 16 & STIFFENER PLY 1	8 PLY STIFFENER	BTN PLY 16 & 17 (24 PLY)
18	PREPREG BACKING	2.50 X 2.50	BTN PLY 8 & 9 (50%)	16 PLY LAMINATE	
19	PREPREG BACKING	1.50 x 1.50	BTN PLY 5 & 6 OF ST PAD	8 PLY SHEAR TIE PAD	BTN PLY 21& 22 (24 PLY)
20	GREASE	∅1.50	BTN PLY 8 & 9 (50%)	16 PLY LAMINATE	SOME OVER DAMPER
21	GRAFOIL	∅1.25	BTN PLY 12 & 13 (75%)	16 PLY LAMINATE	(50%) OF 24 PLY
ITEM #	DESCRIPTION	QUANTITY	DESIGNATION	LOCATION	NOTES
22	FLAT HEAD BOLT	12	100° FL HD, 1/4-20UNC-2A X 0.500		SANDIA WILL PROVIDE
23	HEX NUT	12	1/4-20UNC-2B		SANDIA WILL PROVIDE
24	SHEAR TIE FLANGE	2	SEE SHEAR TIE FLANGE DRAWING		MADE AT NORDAM
25	SOUND DAMPER	4	4.5" X 5.0" SMACSONIC PADS	16 PLY SUBSTRUCTURE	SANDIA WILL PROVIDE
26	SEALANT	AS NEEDED			SANDIA WILL PROVIDE

Figure B-18. Panel 2B—Defect type, size, layer, and locations

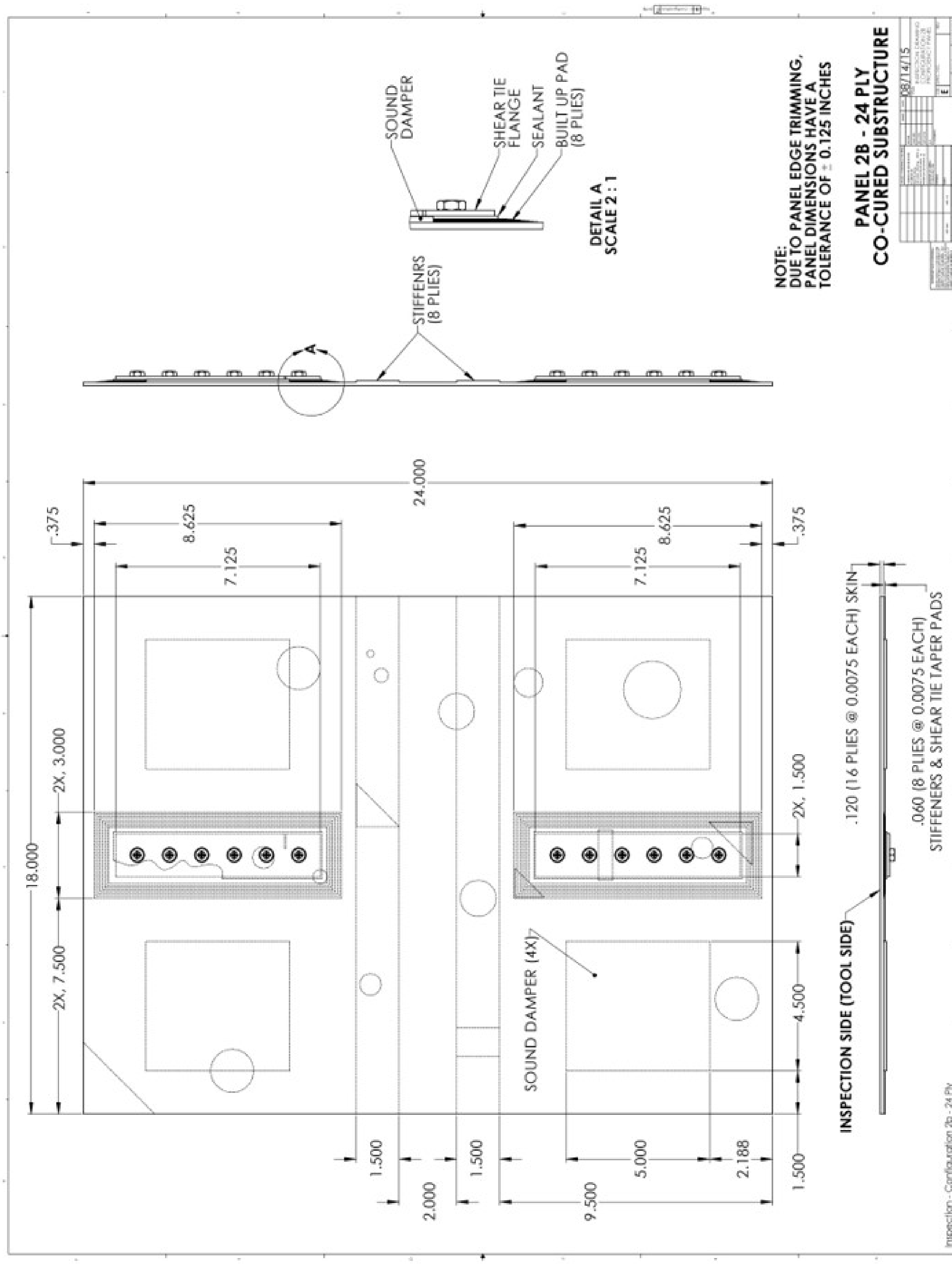


Figure B-19. Panel 2B—Substructure locations

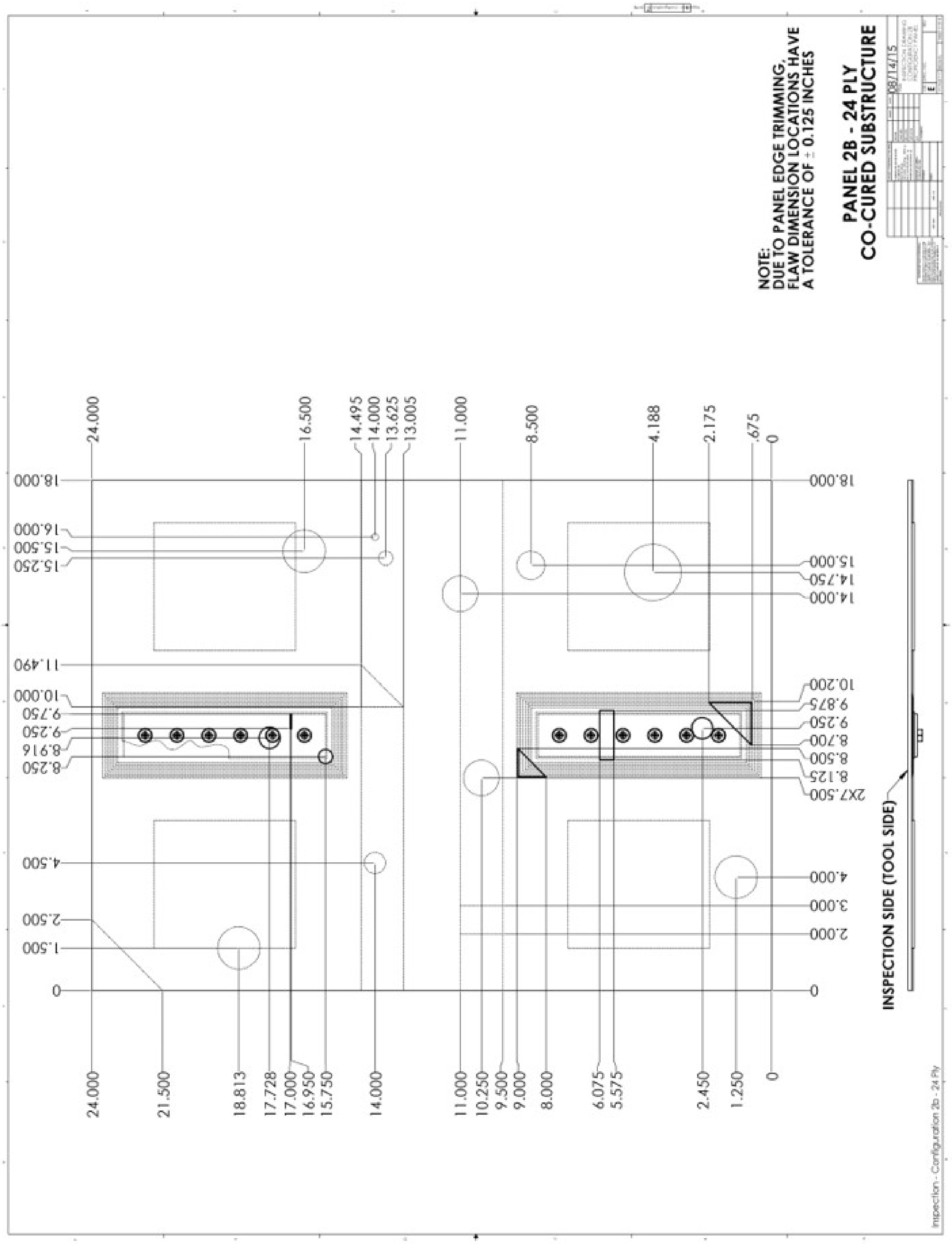


Figure B-20. Panel 2B—Defect locations

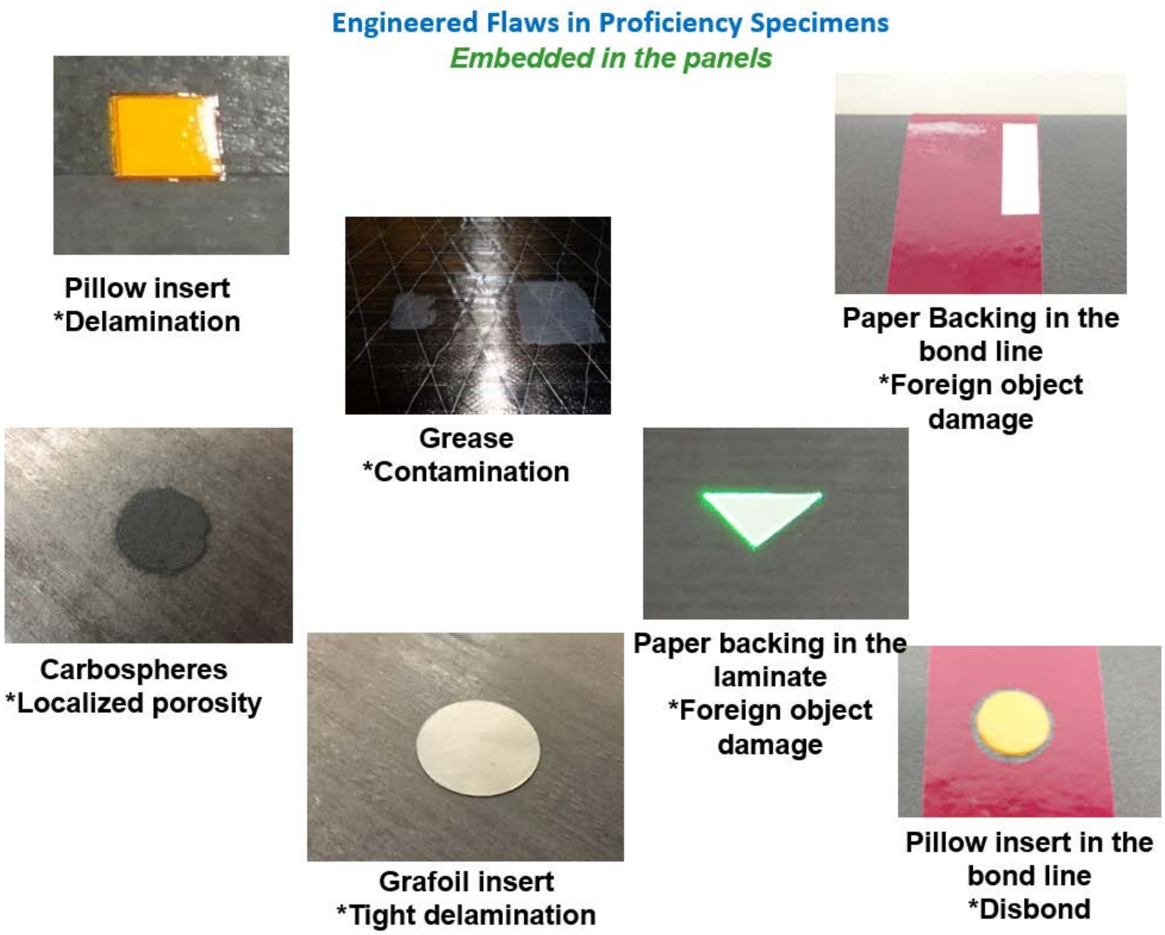


Figure B-21. Pictures of engineered flaws embedded in the proficiency panels

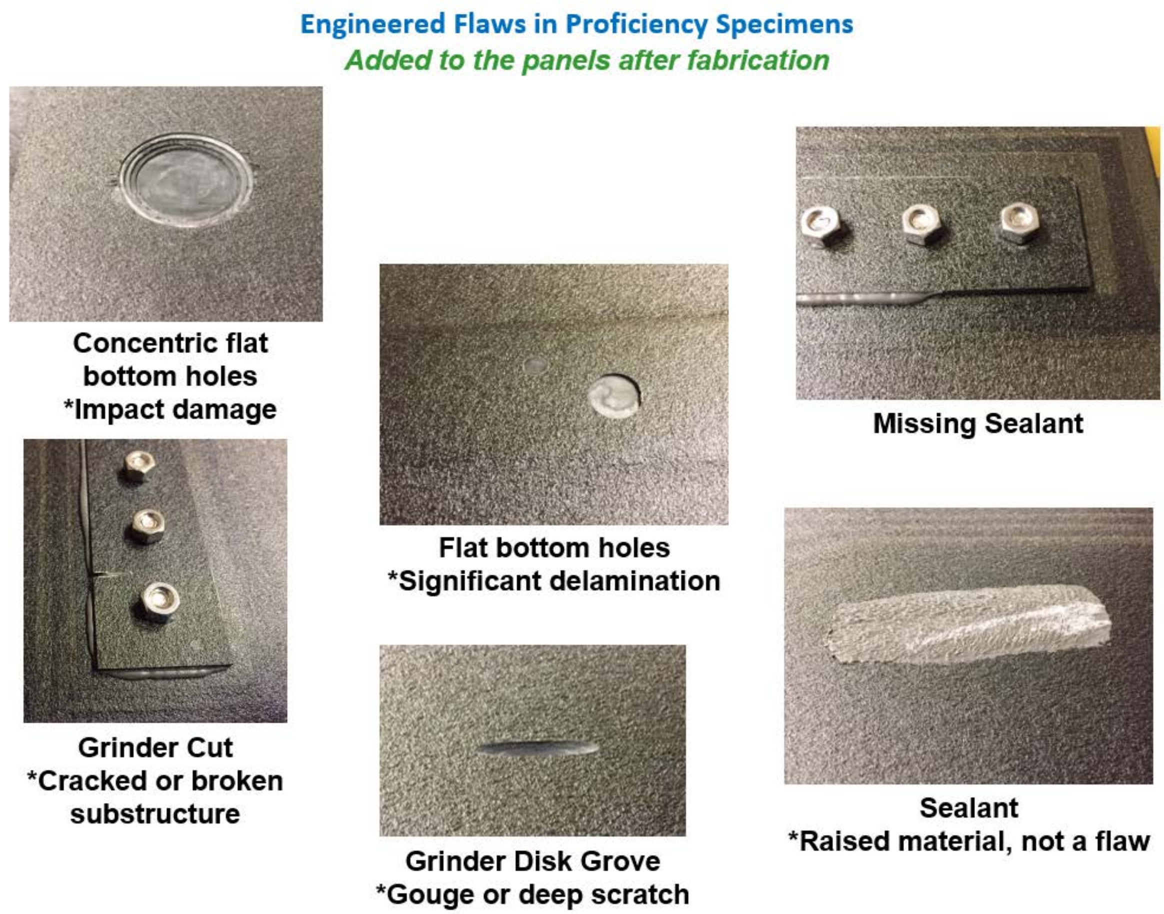


Figure B-22. Pictures of engineered flaws added to the proficiency panels after fabrication and assembly

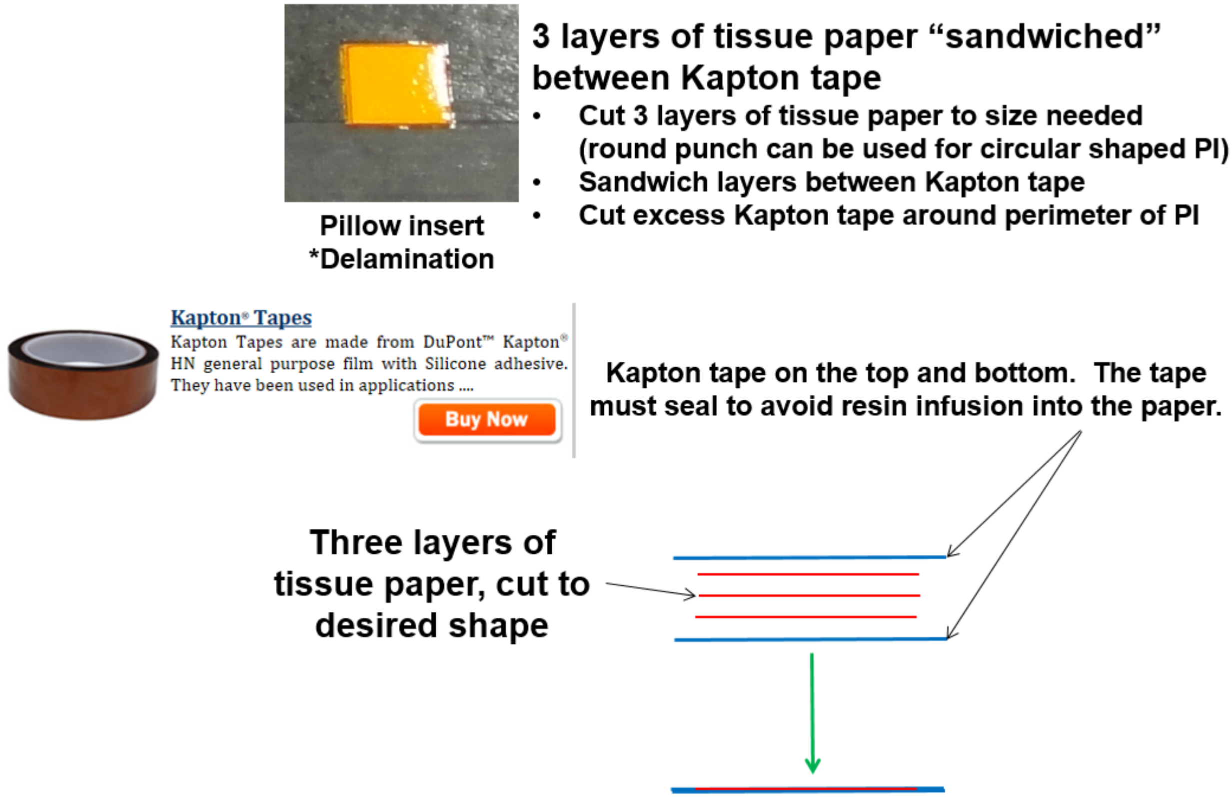
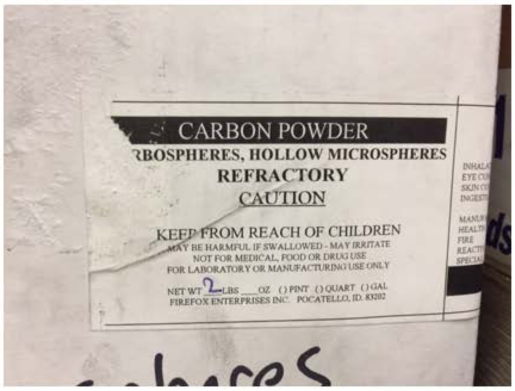
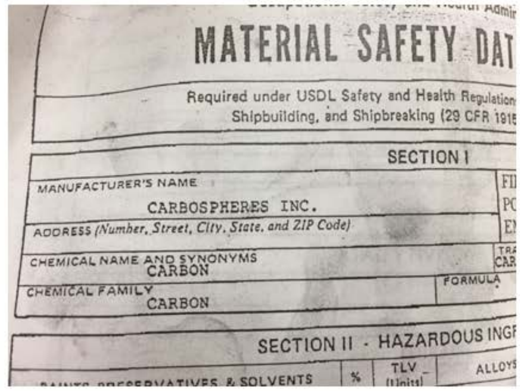


Figure B-23. Schematic showing how pillow inserts are made

- **Cut Mylar template to the desired diameter or shape of defect**
- **Lightly dust Carbospheres onto desired area using the template.**

Carbospheres
***Localized porosity**





http://www.dynamationresearch.com/product/carbospheres.html

CARBOSPHERES

To receive a quote for CARBOSPHERES please call us at **(310) 477-1224** or click the "Prepare a Quote" button next to CARBOSPHERES or the corresponding material.

Figure B-24. Engineered defect representing localized porosity—carbosphere details

Thickness – 0.005"
Use Punch to cut out from sheet

Grafoil insert
*Tight delamination

http://www.sealingdevices.com/flexible-graphite-gaskets

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716.684.7600

STANDARD SPECIFICATIONS FOR GRAFOIL® **Flexible Graphite roll materials**
-Density: 70 lb./ft.^{3} (1.1 g/cc)
-Thickness: .005", .010", .015 ", .020", .025 ", .030 ", .040", .060"
-Length: 25' through 1000'+
-Width: 24" and 39.4" (1 meter)

Figure B-25. Grafoil insert defects

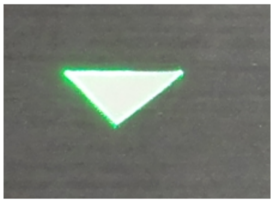
Grease
*Contamination

- Cut Mylar template to the desired diameter or shape of defect
- Lightly brush grease onto desired area using the template.

http://www.ellsworth.com/products/by-manufacturer/dow-corning/lubricant-products/grease/dow-corning-molykote-br2-plus-multi-purpose-e.p.-grease-black-400-g-cartridge/?gclid=CPOz6rvAwskCFc5gfgodANEPGw

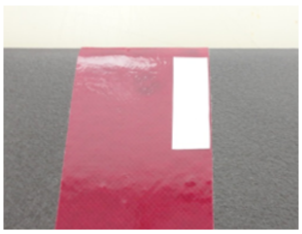
1 (877) 454-9224

Figure B-26. Grease contamination defect details



PrePreg paper backing cut to shape

Paper backing in the laminate
*Foreign object damage



Adhesive film paper backing cut to shape

Paper Backing in the bond line
*Foreign object damage

Figure B-27. Prepreg paper backing foreign object damage details

APPENDIX C—HANDS-ON EXERCISES

Conventional A-Scan Laboratory Exercise 1

Exercise 1 – Calibration: Calibrate, set Material Velocity and TCG Curve

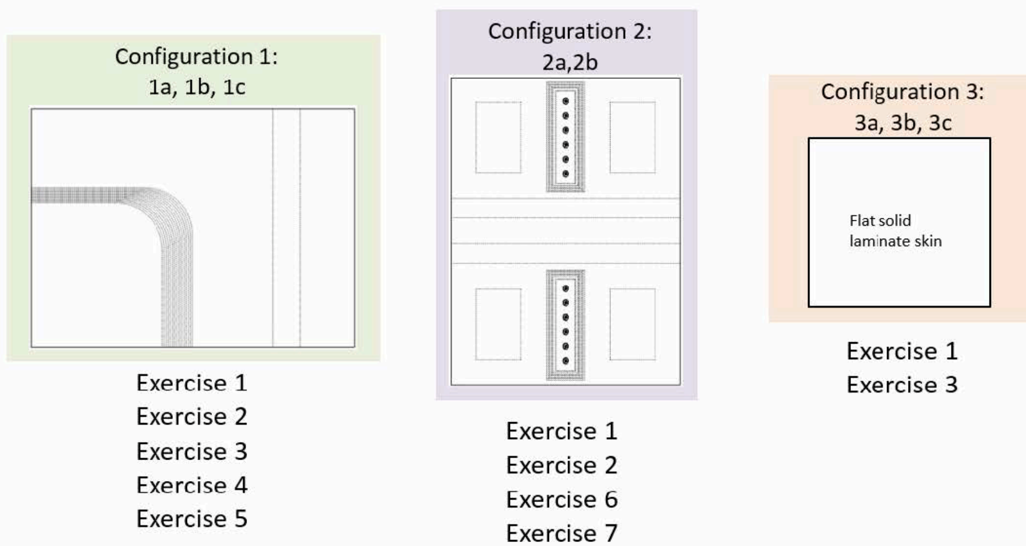
Objective: Set the material velocity, ensure the ultrasonic instrument is properly calibrated, and set the Time Corrected Gain (TCG) curve.

Note: This exercise is the basis for subsequent exercises and it is important that the calibration and TCG curve are set properly. If you are not confident these are set properly, ask for assistance.

The hands-on portion of this training uses the Proficiency Specimens and consists of seven exercises. Figure 1 shows which exercises apply to which panel. Complete the exercises shown in the figure depending on whichever panel has been selected.

Hands-On Exercise Flow Chart

Applicable exercises for each panel configuration



Note: Exercise 1 (Calibration) only needs to be completed once and applies to all subsequent exercises

Figure 1: Hand-On Exercise Flow Chart

Equipment/Materials:

- Copy of the General Ultrasonic Carbon Laminate Inspection Guide (Conventional A-Scan)
  - This is the inspection procedure that all exercises will reference
- Omniscan Unit MX
  - If other UT instrument - must operates at frequency range between 3.5 to 5 MHz
    - Has Time Corrected Gain (TCG)
    - Ability to adjust overall gain with active TCG
- Ultrasonic Transducer
  - 0.5" diameter (12.5 mm) is recommended
  - Recommend frequency between 3.5 and 5 MHz
- Delay line – 3/4" Lucite, Rexolite, ZIP or Polystyrene
- Couplant - Water or Gel
- Ultrasonic transducer cable
- Step Wedge – ST8871

Instructions:

- Obtain reference standard ST8871 used for calibration
- Refer to the General Ultrasonic Carbon Laminate Inspection Guide (A-Scan)
  - Read sections (1) Purpose and (2) Applicability of the Training Guide
  - Begin in section 4.1, Set up Cable, Transducer, and Delay Line
  - Follow the steps in section 4.2, Calibration of the Ultrasonic Unit
    - The instrument should now be set up, calibrated and ready to have the TCG curve set
  - Follow the steps in section 4.3, Setting the Time Corrected Gain

Result: Once this exercise is completed a screen presentation similar to the one shown in Figure 2 should be achieved. Check the settings by hand-scanning over a Configuration 1 Proficiency Specimen (1a, 1b, or 1c). These specimens contain a taper region where the material thickens. The amplitude over various thicknesses of the panel should remain at approximately 80% full-screen height (FSH). *Note: The amplitude will decrease over the secondary bonded region and not remain at 80% FSH. This is due to the adhesive layer. This is addressed in a different exercise.*

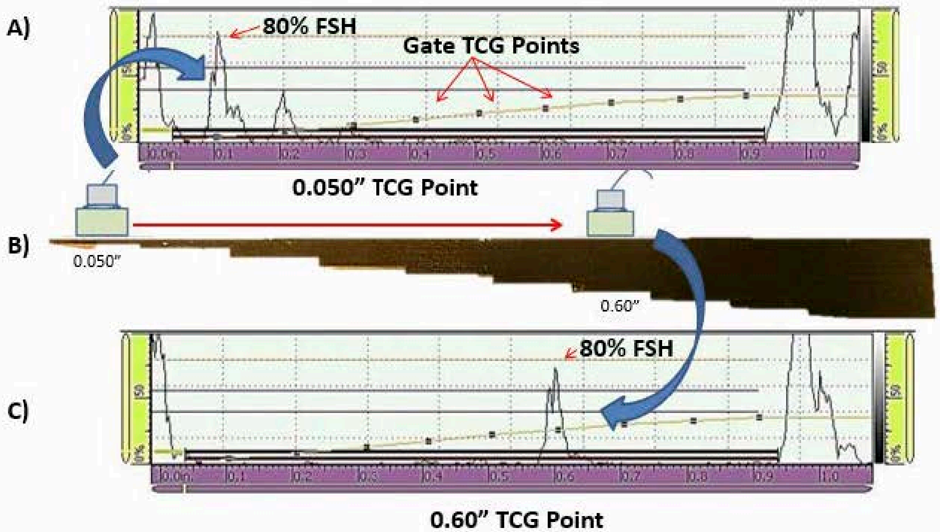


Figure 2: Omniscan A-Scan Presentation Showing Time Corrected Gain Points (TCG) using a 5 Mhz, 0.5" Diameter Transducer with a 0.5" Delay Line. (Gate – green bar at 10% FSH, 8 TCG Points – Red squares)

Conventional A-Scan Laboratory Exercise 2

Exercise 2 – Mark Substructure on Surface of Panel

Objective: Identify the structural configuration drawing associated with the specimen and mark the substructure on the surface using a water soluble crayon.

Note: It is important that the substructure is properly mapped onto the surface of the panel or substructure may be mischaracterized as defects. If you are inspecting specimens 3a, 3b, or 3c skip this exercise and continue to Exercise 3.

Equipment in addition to previous exercise:

- Measuring tool – Micrometer recommended to measure thickness of panels and ruler for flaw sizing
- Pencil, aircraft marking (China marker, crayon)
- Select one Proficiency Specimen (1a, 1b, 1c, 2a, or 2b)

Instructions:

Note: The calibration and TCG curve should have been set in the previous exercise.

- Select one of the Proficiency Specimens from this list: 1a, 1b, 1c, 2a, or 2b. (*Note: configuration 3 specimens do not contain substructure and are not applicable to this exercise.)*
- Refer to the General Ultrasonic Carbon Laminate Inspection Guide (A-Scan)
    - Read section 4.4
    - Follow the steps in section 4.4, Conducting the Inspection. Only follow the instructions through step 6: Marking the Substructure.
- Use the drawings provided to help identify and mark the substructure elements. (*Note: Ensure that the panel is oriented so that the triangle marking is at the upper right-hand corner of the panel during the inspection and when comparing to any drawings or pictures.)*
- Make sure to mark edges of the following:
    - Stiffener elements (co-cured and secondarily bonded)
    - Beginning and end of taper regions
    - Built-up pad sections (steep taper region leading up to shear tie fastener region)
    - Shear tie fastener region
    - Sound dampers
- Once the substructure has been marked on the panel in a similar fashion as shown in Figure 1, move onto the next exercise.

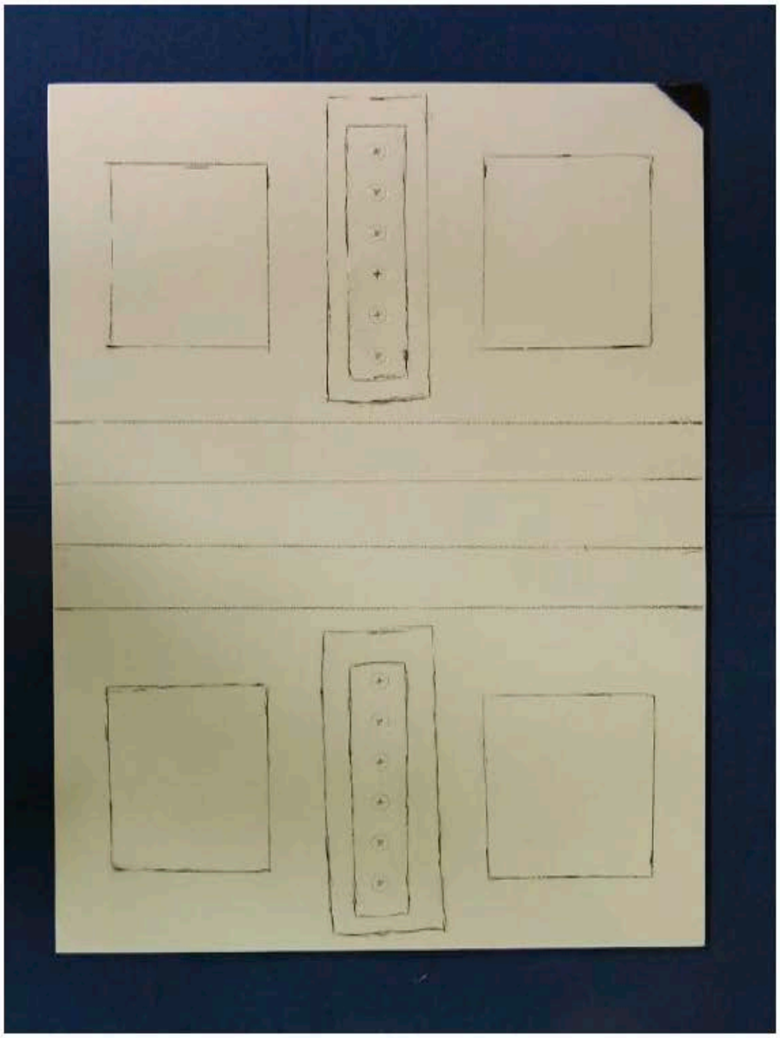


Figure 1: Picture of Configuration 2 Proficiency Specimen with the Substructure Elements and Sound Damper Tiles Marked on the Surface

Drawings:

- The drawing for Proficiency Specimen 1a is shown in Figure 2, and the drawing for Proficiency Specimen 1b is shown in Figure 3. The substructure for these two panels is similar with the addition of a second bonded stiffener on panel 1b. The flaw profile is different for each panel.
- The drawing for Proficiency Specimen 1c is shown in Figure 4.
- The drawing for Proficiency Specimen 2a and 2b is shown in Figure 5. The substructure for these two panels is the same, but the flaw profile is different for each panel.

Proficiency Specimens: 1b

Structure: Taper (10:1 and 20:1) and secondary bond

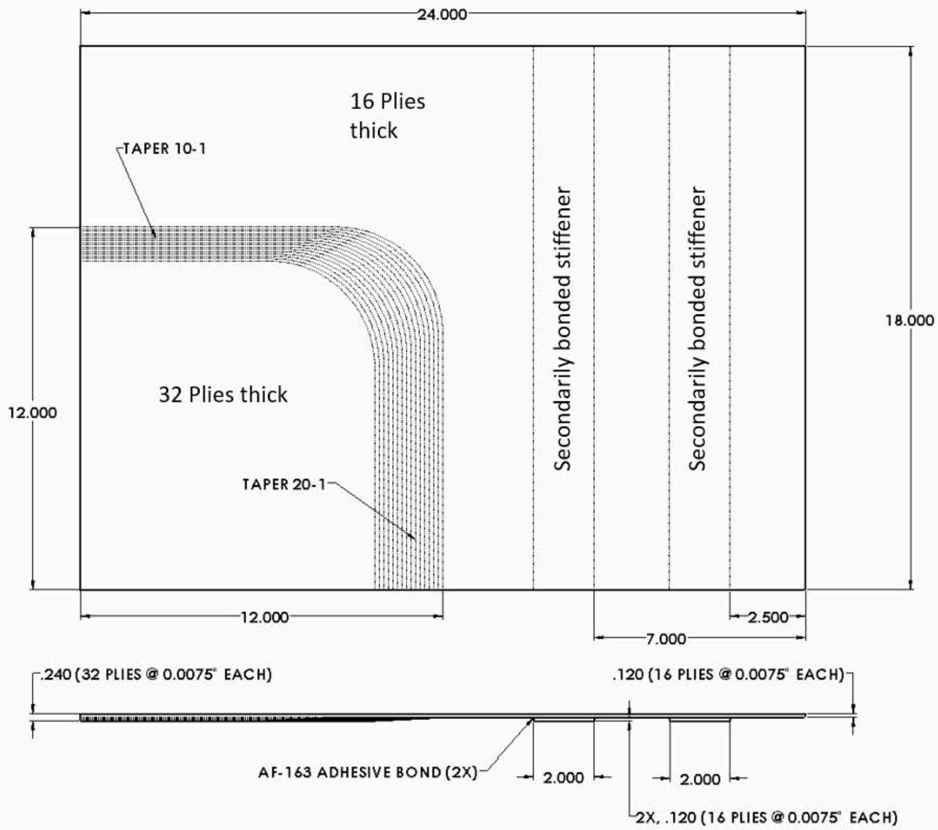


Figure 3: Drawing for Proficiency Specimen 1b

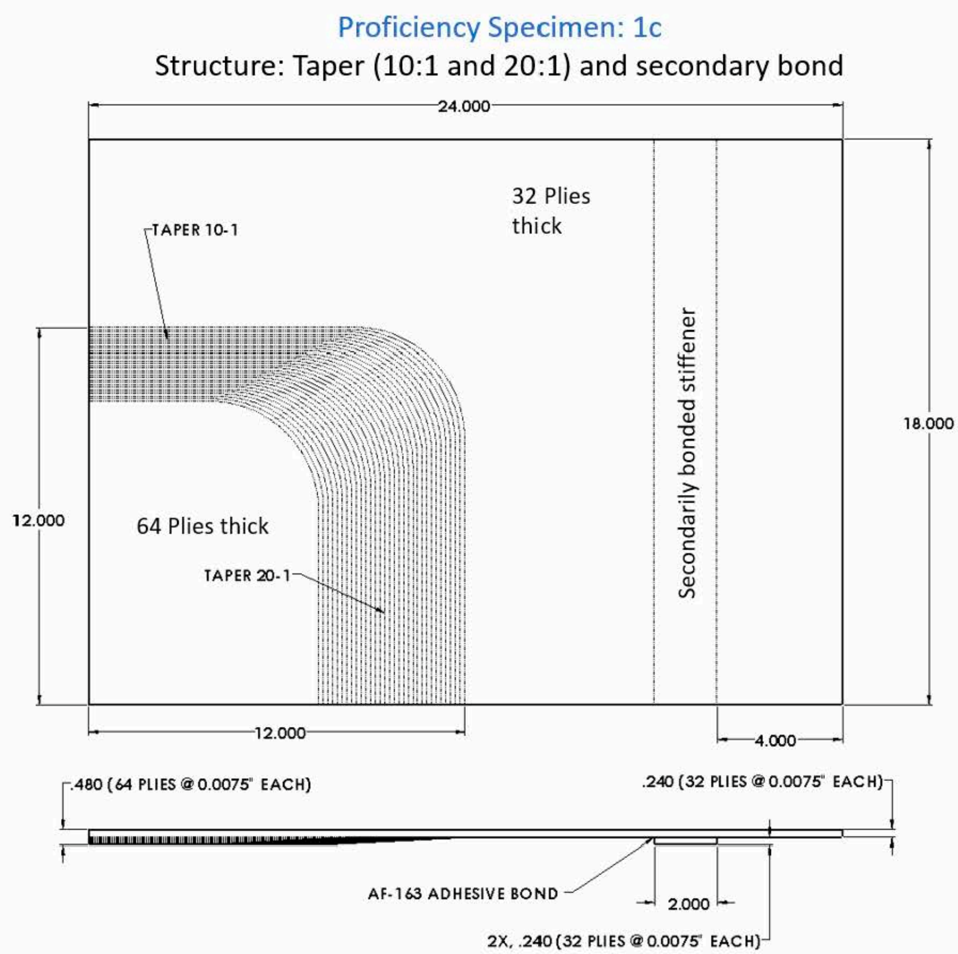


Figure 4: Drawing for Proficiency Specimen 1c

Proficiency Specimens: 2a and 2b

Structure: Uniform thickness skin, pads, fastened shear tie flanges, co-cured stiffeners, sealant

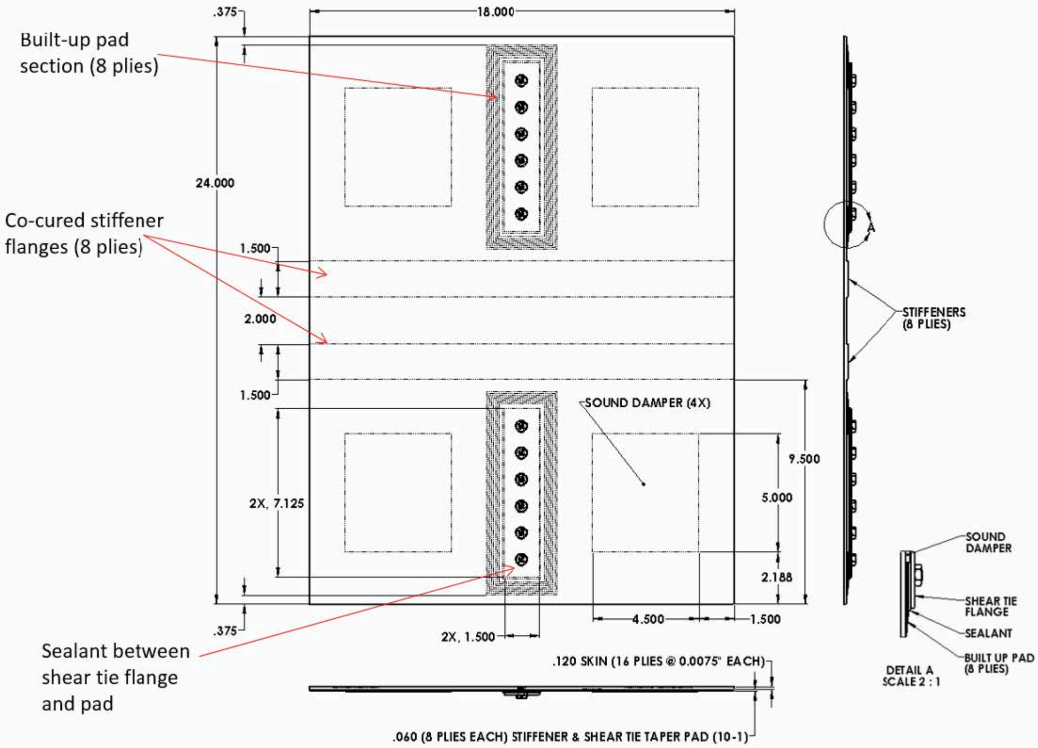


Figure 5: Drawing for Proficiency Specimens 2a and 2b

Results:

- Once this exercise is completed, the substructure has been marked onto the surface of the Proficiency Specimen that was selected. Properly marking the substructure allows the inspector to focus the inspection on isolated regions of interest. This helps ensure that the A-scan results obtained are predictable and that defects are more easily identified.

Conventional A-Scan Laboratory Exercise 3

Exercise 3 – Defect Detection in Uniform Thickness Skin

Objective: Now that the panel has the substructure marked on it, conduct an A-scan inspection over the uniform thickness regions for various types and sizes of engineered defects.

Note: This exercise is only applicable to panels 1a, 1b, 1c, 3a, 3b, and 3c. If you are working on 2a or 2b move on to Exercise 6.

Instructions:

- Now that the substructure has been marked, proceed to inspect the uniform thickness areas. Do not inspect the tapered areas or bonded substructure for this exercise.
- Continue to Step 7 in section 4.4 of the General Ultrasonic Carbon Laminate Inspection Guide (A-Scan).
- Use indexing marks and the straight edge scanning technique to inspect the areas of uniform thickness.
- Mark the edge of defects where the back wall signal amplitude drops by 6dB.
- After completing the inspection of uniform thickness areas, allow the instructor to provide feedback.
- There is no substructure on panels 3a, 3b or 3c and the exercises for that panel are completed. Select another panel and begin with Exercise 2.

Drawings:

- The drawings for Proficiency Specimens 1a, 1b and 1c are shown in Figures 1 and 2. The drawings highlight the areas of uniform thickness in red. Inspect these regions before moving onto the next exercise.

Note: Proficiency Specimen 1b has an additional bonded stiffener not shown in Figure 1. On this panel inspect the area between the two bonded stiffener elements.

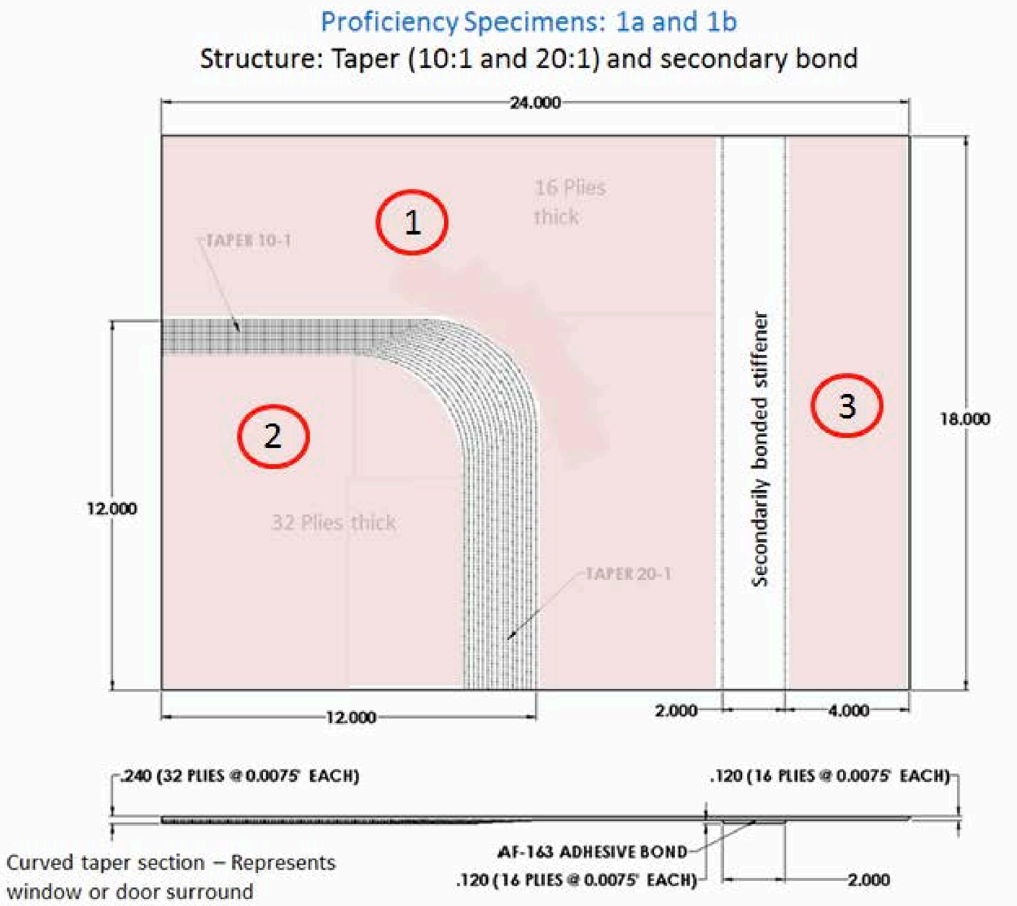


Figure 1: Drawing for Proficiency Specimen 1a and 1b Highlighting Areas to be Inspected in Red.
Note: 1b has an additional bonded stiffener not shown.
Inspect the area between the two bonded stiffener elements.

2

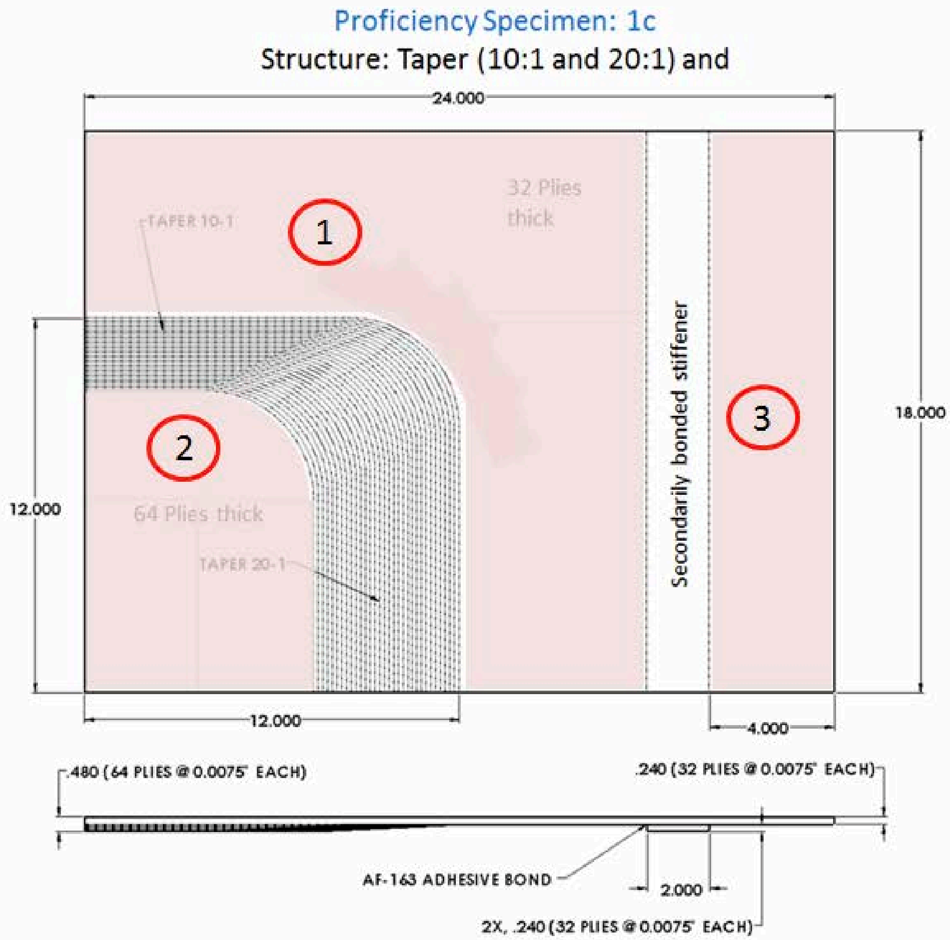


Figure 2: Drawing for Proficiency Specimen 1c Highlighting Areas to be Inspected in Red.

Results:

- For this exercise, the areas of uniform thickness were inspected looking for engineered defects within the 16, 32 and 64 ply thickness regions.
- Beginning the inspection on the areas of uniform thickness simplifies the inspection and limits the chance of substructure signal confusion and can minimize false calls.
- Using the straight edge scanning technique ensures full coverage of the inspection area. It was shown in the Solid Laminate POD study that inspectors who used a straight edge with tick marks performed much better than inspectors who used freehand techniques. The results provided below highlight the performance increase and ability to detect smaller flaws using straight edge techniques.
  - 12-20 Ply Freehand $POD_{[90/95]}$ – 2.39" dia.
  - 12-20 Ply Straight Edge with Tick Marks $POD_{[90/95]}$ – 1.06" dia.
  - 20-32 Ply Freehand $POD_{[90/95]}$ – 1.35" dia.
  - 20-32 Ply Straight Edge with Tick Marks $POD_{[90/95]}$ – 0.64" dia.
  - Overall (combined 12-20 & 20-32 Ply) Freehand $POD_{[90/95]}$ – 1.75" dia.
  - Overall (combined 12-20 & 20-32 Ply) Straight Edge with Tick Marks $POD_{[90/95]}$ – 0.91" dia.

Conventional A-Scan Laboratory Exercise **4**

Exercise 4 – Defect Detection in Tapered Skin

Objective: Now that the substructure has been marked and the areas of uniform thickness have been inspected, the objective is to conduct an A-Scan inspection of the tapered regions on the proficiency panel.

Note: This exercise is only applicable to panels 1a, 1b, and 1c. If you are working on 2a or 2b move on to Exercise 6.

Instructions:

- The tapered region has already been marked onto the panel.
- While inspecting the tapered area, scan along lines of equal thickness. Depending on the taper ratio, scanning across the taper can complicate A-Scan interpretation. Reference Figure 9 in the A-Scan General Inspection Guide and the graphics shown in the Drawings section of this exercise.
- While scanning along lines of equal thickness the back wall signal should not shift unless there is a defect present.
  - Look for any abrupt shifts on the A-Scan signals.
  - The A-Scan signal will shift slowly to the left or right on the display when inspecting across (perpendicular to the taper) a tapered region. A defect will cause an abrupt shift in signal position or a sudden decrease in amplitude.
- Read the A-Scan procedure paying particular attention to section 4.4 Step 8, tapered regions.
  - Mark the edge of flaws where the back wall signal amplitude drops by 6 dB.
- After inspecting the tapered area allow the instructor to provide feedback.

Drawings:

- The drawings for Proficiency Specimens 1a, 1b and 1c are shown in Figures 1 and 2. The drawings highlight the taper area in red. The blue dashed arrow indicates scanning along lines of equal thickness. Inspect these regions before moving onto the next exercise.

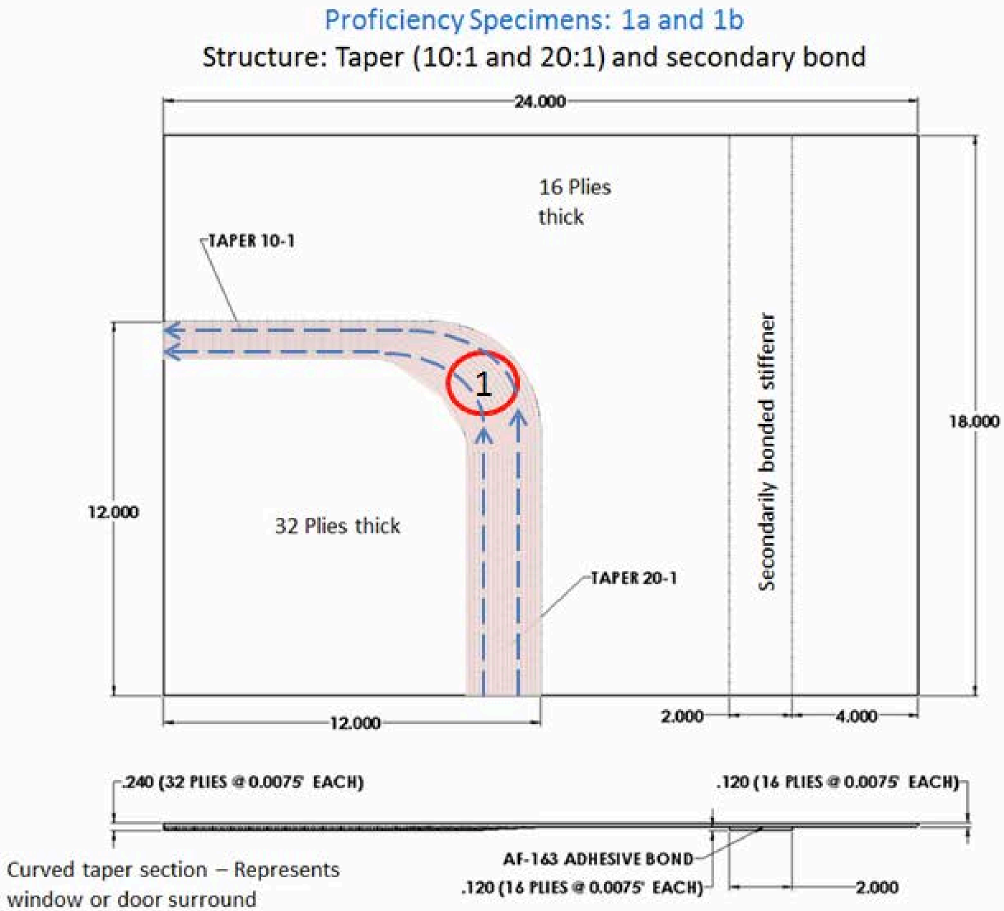


Figure 1: Drawing for Proficiency Specimen 1a and 1b Highlighting Areas to be Inspected in Red

2

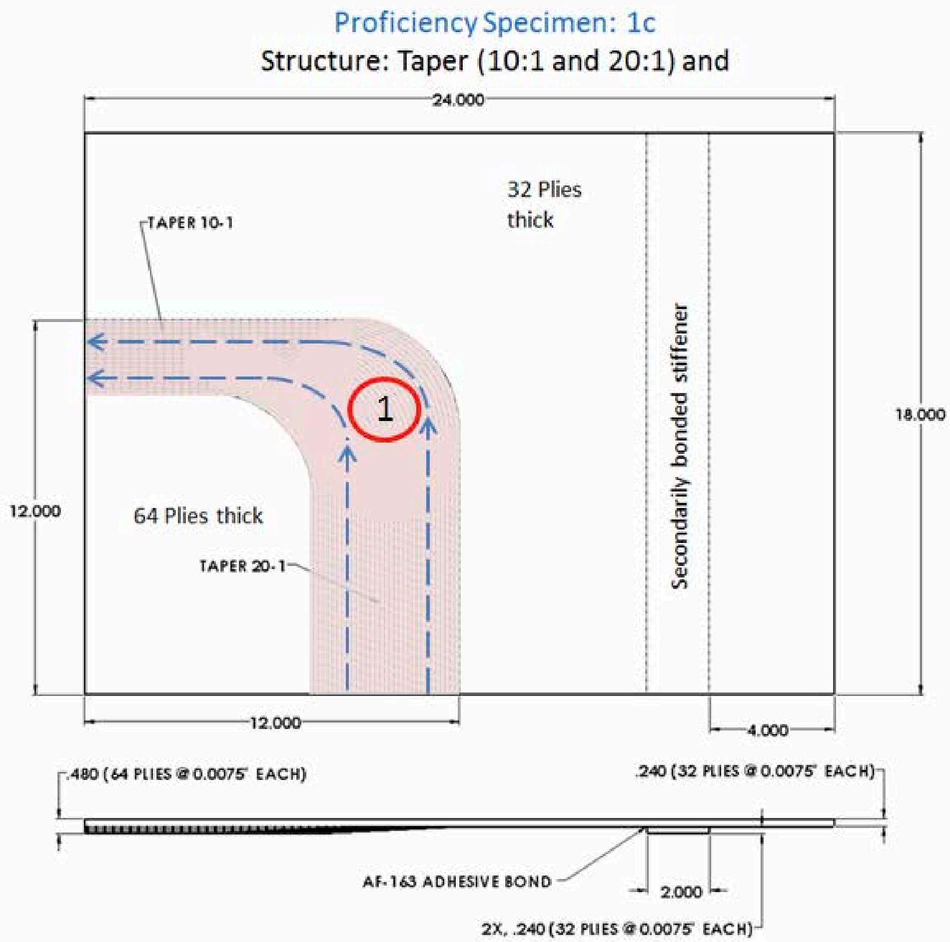


Figure 2: Drawing for Proficiency Specimen 1c Highlighting Areas to be Inspected in Red

Results:

- This exercise should result in an increased familiarity with inspecting tapered solid laminates.
- Inspection over tapered regions can be more complex than inspecting areas of uniform thickness. Identifying these areas and understanding their geometry helps interpret signal shifts and minimizes unexpected signal changes due to geometry.
- The inspector should now be able to distinguish between a gradual tapered shift in the A-Scan display and an abrupt shift over a defect.

Conventional A-Scan Laboratory Exercise 5

Exercise 5 – Inspection of Bonded Substructure

Objective: Inspect over the secondarily bonded stiffeners and analyze A-scan signals for defects. Defects can be present in the parent laminate, bond line and the secondarily bonded stiffener element.

Note: This exercise is only applicable to panels 1a, 1b, and 1c. If you are working on 2a or 2b move on to Exercise 6.

Instructions:

- The secondary bond line has already been marked onto the panel.
- Secondary bond lines are typically more ultrasonically noisy than co-cured bond lines. The gain setting on the instrument being used may need to be increased to penetrate the material.
- Determine the thickness of the skin and the bonded stiffener. Check that the location of the bond line signal and back wall signal of the stiffener element appear at approximately the correct depth on the A-Scan unit.
- Couple to the part and move the transducer over the bonded area to get a general estimate of the baseline amplitude of the bond line signal.
    - Determine the baseline signal amplitude of the bond line and the back wall of the stiffener. Indications of defects in the bond line will be based on the estimated signal amplitude.
- Scanning parallel to the direction of the stiffener, inspect the bond line as described in the General A-Scan Inspection Procedure. Refer to section 4.4 Step 8. Scanning along a straight edge is recommended to ensure full coverage.
- Using the baseline signal amplitude estimated earlier, mark indications where the back wall signal decreases by 6 dB from an adjacent bonded stiffener region, or if the bond line signal increase by 6 dB.
- After completing the inspection allow the instructor to grade the indications marked on the panel.
- All exercises have been completed for this panel (1a, 1b or 1c). Select another panel and begin with Exercise 2.

Drawings:

- The drawing for Proficiency Specimens 1a, 1b and 1c are shown in Figures 1, 2 and 3 with the bonded stiffener to be inspected highlighted in red.

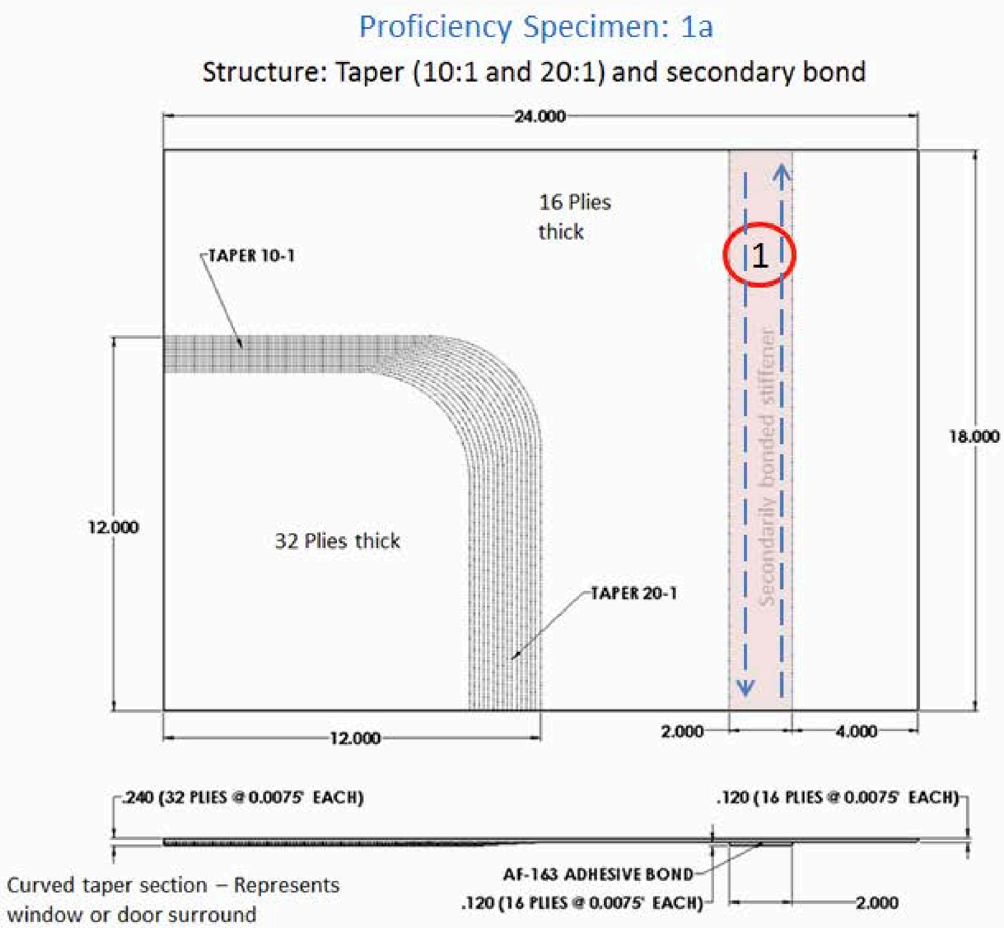


Figure 1: Drawing for Proficiency Specimen 1a Highlighting Areas to be Inspected in Red

2

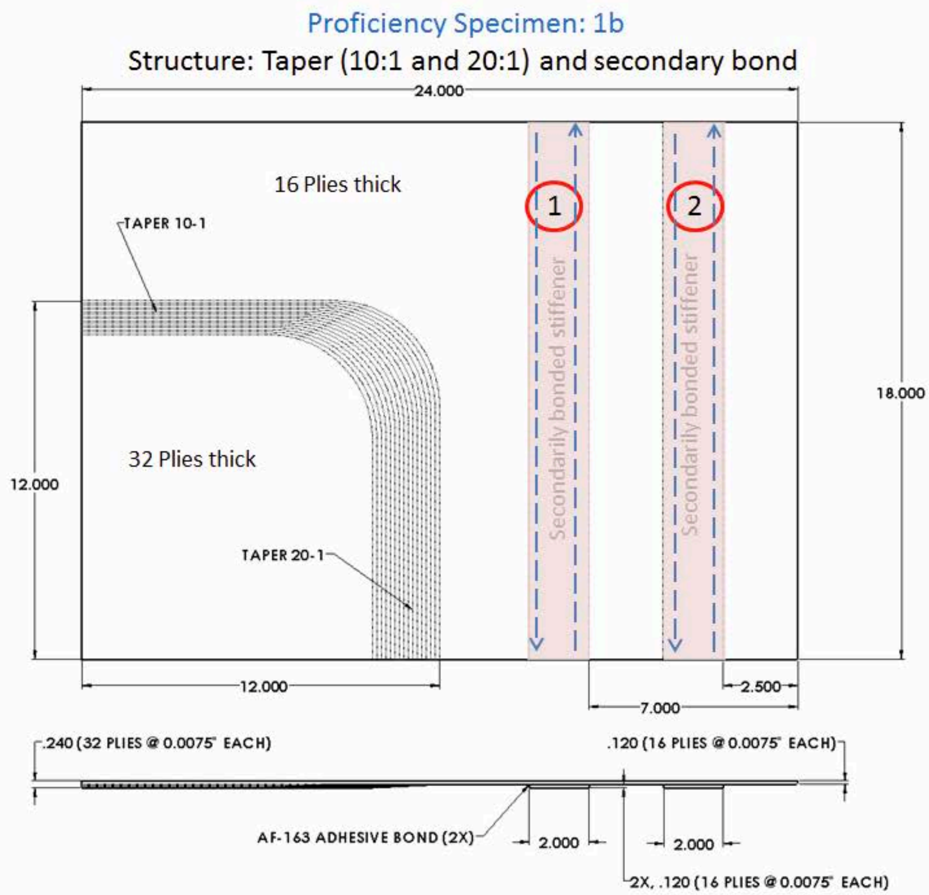


Figure 2: Drawing for Proficiency Specimen 1b Highlighting Areas to be Inspected in Red

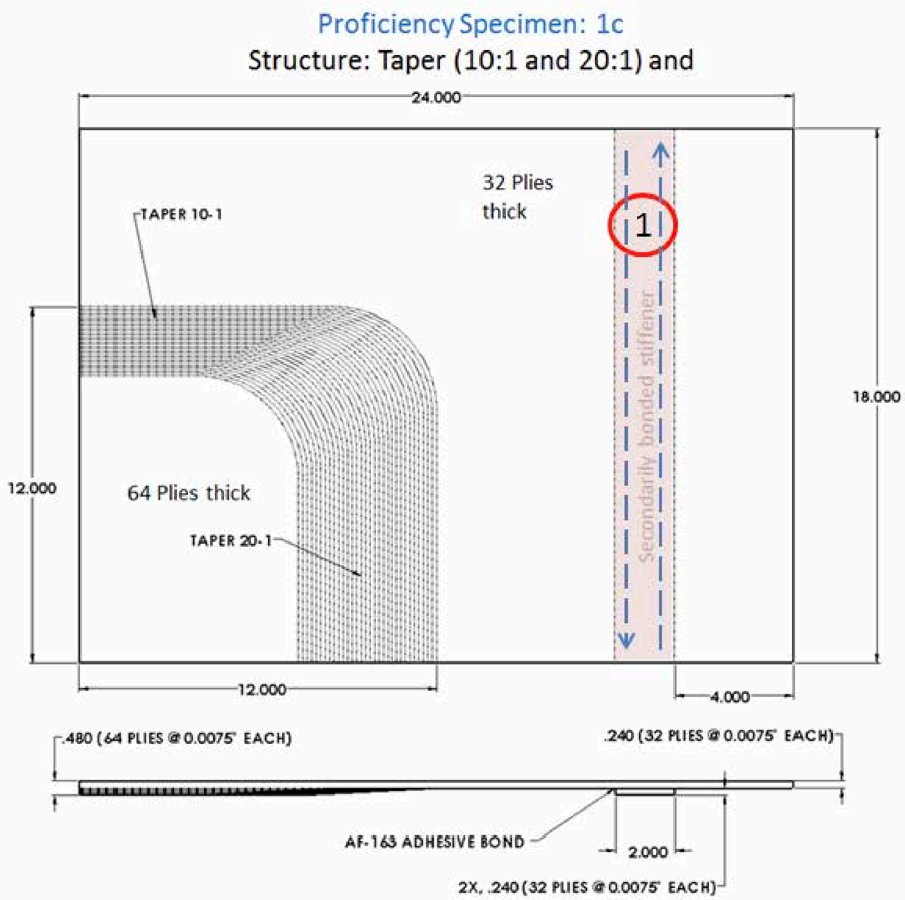


Figure 3: Drawing for Proficiency Specimen 1c Highlighting Areas to be Inspected in Red

Results:

- The areas containing secondary bond lines were inspected for defects. Defects present within the parent laminate, bond line and stiffener element were observed.
- A general understanding of the ultrasonic response of bonded substructures was gained.

Conventional A-Scan Laboratory Exercise 6

Exercise 6 – Inspection of Co-Cured Substructure

Objective: Inspect the co-cured stiffeners and analyze A-Scan signals for defects.

Note: This exercise is only applicable to panels 2a and 2b.

Instructions:

- The co-cured bond lines have already been marked onto the panel.
- Determine the baseline signal amplitude over the stiffener and mark indications where the back wall signal decreases by 6 dB.
- Scan the co-cured stiffeners on specimens 2a and 2b as shown in red in Figure 1.
- When scanning, scan in the direction parallel to the stiffener.
    - Note: Properly bonded co-cured stiffeners typically do not have an interface signal between the back wall of the parent laminate and the stiffener. The stiffener region typically appears as though it is a thicker laminate.
- After completing the inspection have the instructor provide feedback.
- Continue on to Exercise 7.

Drawings:

- The drawings for Proficiency Specimen 2a and 2b are shown in Figures 1 and 2. Note: the substructure for these two panels is the same, but the flaw profile is different for each panel.

Proficiency Specimens: 2a and 2b

Structure: Uniform thickness skin, pads, fastened shear tie flanges, co-cured stiffeners, sealant

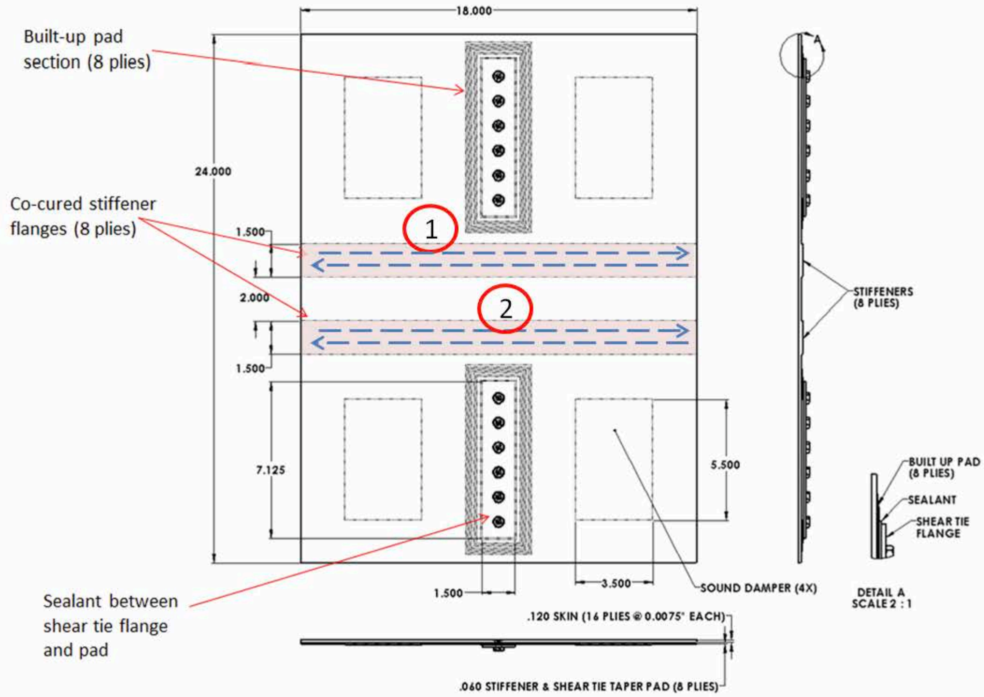


Figure 1: Drawing for Proficiency Specimens 2a and 2b

2

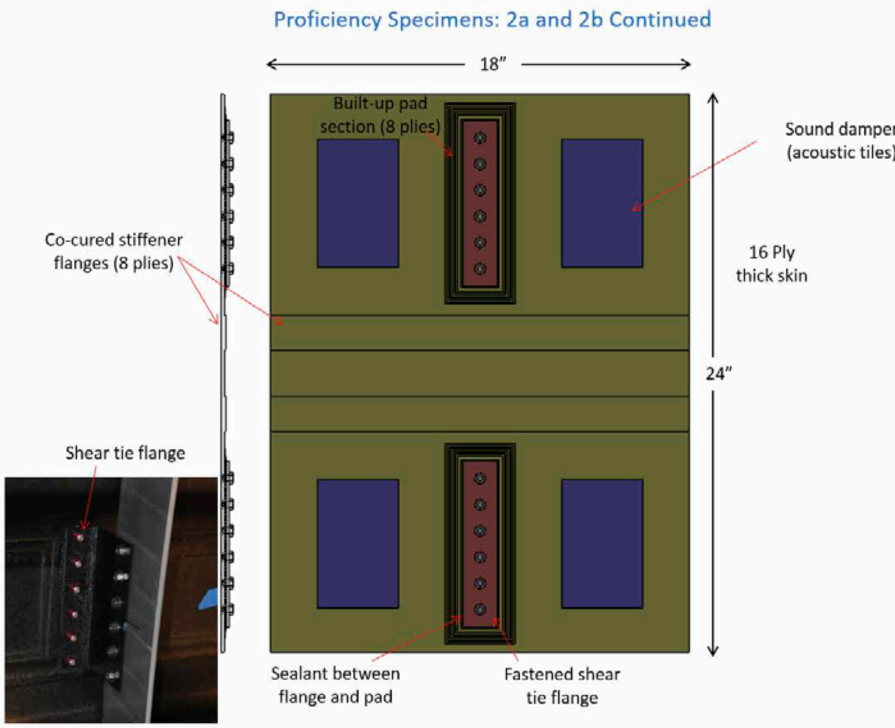


Figure 2: Additional Drawing Information for Proficiency Specimens 2a and 2b

Results:

- The areas containing co-cured bond lines were inspected looking for engineered defects within the parent laminate, bond line and stiffener elements.
- A general understanding of the ultrasonic response of co-cured substructure was obtained.
- Identification of ultrasonic signal changes were observed within the co-cured laminate interface and the bonded stiffener back wall.

Conventional A-Scan Laboratory Exercise 7

Exercise 7 – Defect Detection around Other Aircraft Elements

Objective: Practice inspecting built-up pads, shear ties, fastener regions, sealant spots, and acoustic tiles.

Note: This exercise is only applicable to panels 2a and 2b.

Instructions:

- The acoustic tiles, built-up pad sections and shear ties should have been marked.
- Scan the built-up pads and shear tie sections first, then move on to the areas that have uniform thickness with acoustic tiles next as shown in Figure 1.
  - **Built-up Pad** - An area that has a buildup of carbon plies and is typically located in areas where components are fastened together. The built-up area typically has a short, steep, taper close to the buildup. The A-Scan signal response will shift to the right on the display as the transducer is moved over the built-up pad and will give an appearance of a steep taper.
  - **Shear Ties** - Usually an angle type bracket that is used to connect the skin to a frame. Typical shear ties can have complex shapes and specific load carrying requirements. On the Proficiency Specimens, the shear ties are located at the build-up pad and connected to the skin using fasteners and sealant. As the part is scanned, look for abrupt changes in the A-Scan signals while the transducer is moved over the shear tie area. Take into consideration the sealant and the fasteners.
  - **Fasteners** - Scan the laminate area around each fastener. Ensure the transducer fits properly between the fasteners.
  - **Sealant Spots** - Sealant spots can be mistaken for defects if only viewing the amplitude of the back wall signal. There are a few sealant spots on the backside of the panels. Try to identify these spots and differentiate them from the acoustic tiles and engineered defects.
  - **Acoustic Tiles** – Acoustic tiles are placed in various fuselage locations to help mitigate excessive acoustic noise. Acoustic tiles cause the amplitude of the A-Scan signal to decrease slightly. The chance of false calls will decrease by identifying the location of acoustic tiles prior to performing the inspection.
- After completing the inspection, have the instructor grade the panel and provide feedback.

Drawings:

- The drawings for Proficiency Specimen 2a and 2b are shown in Figures 1 and 2. Inspect the areas shown in blue and red. Note: the substructure for these two panels is the same, but the flaw profile is different for each panel.

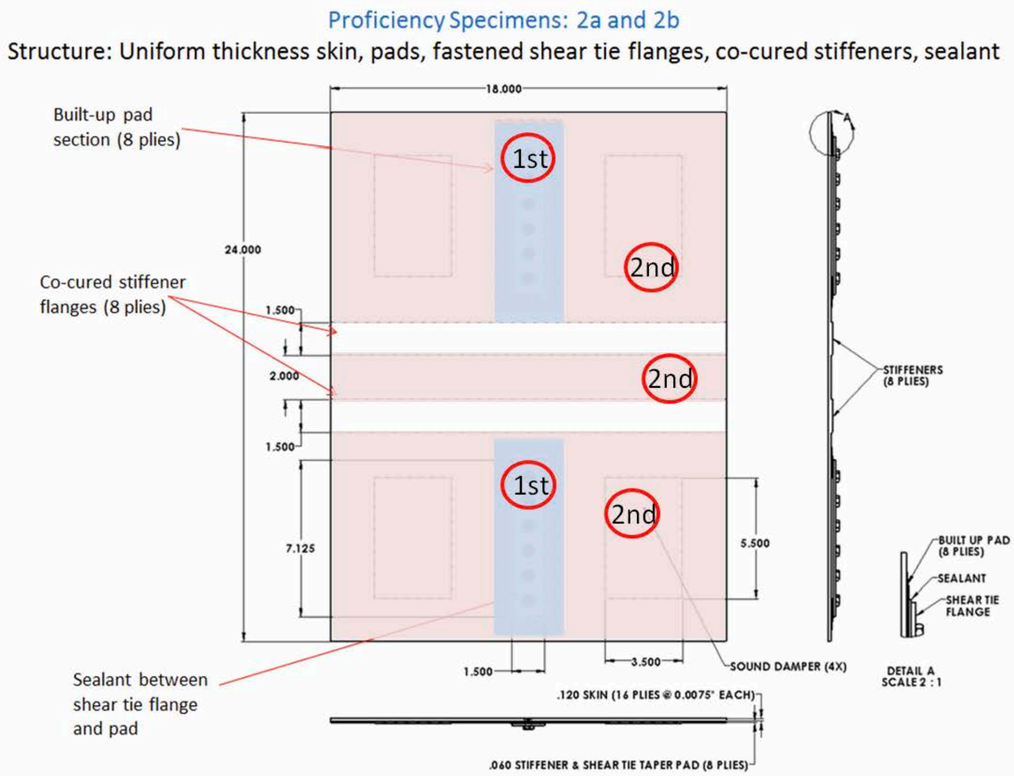


Figure 1: Drawing for Proficiency Specimens 2a and 2b

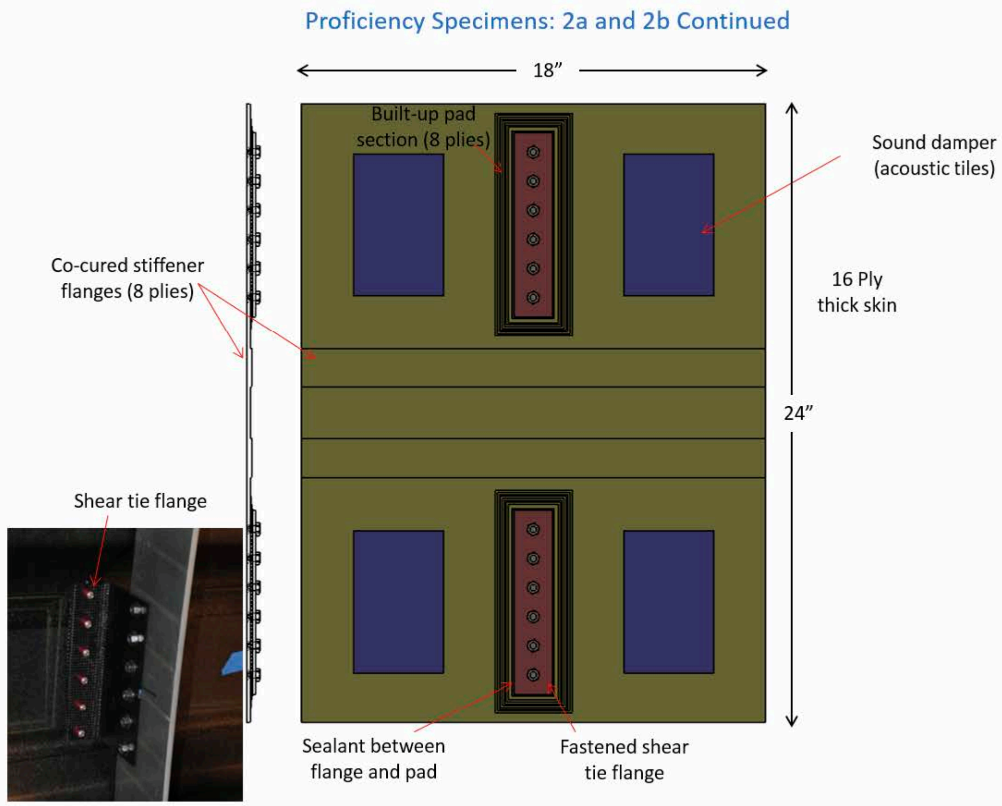


Figure 2: Additional Drawing Information for Proficiency Specimens 2a and 2b

Results: This exercise demonstrated the importance of marking the substructure onto the surface of the part. Separating the part into inspection zones depending on the type of substructure increases efficiency and accuracy. The inspector gained experience inspecting built-up pad areas, shear ties, the laminate around fasteners, sealant spots, and acoustic tiles. The effect of backside artifacts on the amplitude of the A-Scan signal was observed.

APPENDIX D—STUDENT PACKET: A-SCAN AND C-SCAN INSPECTION PROCEDURE

Composite Inspector Training Course

General Ultrasonic Carbon Laminate Inspection Guide (Conventional A-Scan)

FAA Airworthiness Assurance NDI Validation Center (AANC)
Sandia National Laboratories

July 2016

Federal Aviation Administration
FAA William J. Hughes Technical Center, Atlantic City International Airport, NJ 08405
Office of Aviation Research, Washington, D.C. 20591

Sandia National Laboratories
PO Box 5800
Albuquerque, NM 87185

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1 Purpose

The purpose of this training guide is to provide a general A-Scan inspection procedure for solid laminate composite aircraft parts. It is to be used in conjunction with the Composite Inspector Training class and the Proficiency Specimen set developed by Sandia National Labs and the FAA.

Note: This training guide is applicable to the nondestructive inspection of carbon fiber laminated structures and to be used as a general guide during training, not to override any procedures developed by any specific aircraft manufacture or airline.

2 Applicability of Training Guideline

This training guide is for ultrasonic inspection of composite materials and bonded laminates using a single element transducer with a zero-degree delay line. It generally applies to laminates built with unidirectional or woven ply configurations and not sandwich structures. Although there are many ultrasonic units commercially available and capable of inspecting solid laminate composites, the Olympus OmniScan unit was selected for demonstration purposes in this training guide due to its wide range of capabilities and common usage across the aerospace sector.

Since T800 pre-preg tape was used to fabricate the Proficiency Specimens, reference standards made from this material and other reference standards used to set up equipment to inspect this material were used. When inspecting actual aircraft components follow manufacture procedures and set up on reference standards designed to be used with the material the component is made from.

3 List of Materials

- Sandia/FAA Proficiency Specimens
- OmniScan MX or equivalent A-scan instrument
    - If other UT instrument is used - must operate at frequency range between 3.5 to 5 MHz
        - Has Time Corrected Gain (TCG)
        - Ability to adjust overall gain with active TCG
- Ultrasonic transducer
    - 0.5" diameter (12.5 mm) is recommended
    - Recommend frequency between 3.5 and 5 MHz
- Delay line – ¾" lucite, rexolite, ZIP or polystyrene
- Couplant - water or gel
- Straight edge – preferably 24" or longer
- Measuring tool – micrometer recommended to measure thickness of panels and ruler for flaw size
- Ultrasonic transducer cable
- Pencil, aircraft marking (China marker, crayon)
- Step wedge – ST8871

4 General Inspection Procedure

4.1 Set up Cable, Transducer and Delay Line

Apply couplant to the delay line and secure it to the transducer as shown in Figure 1. Connect the transducer to the transducer cable and connect it to the ultrasonic unit. For the OmniScan MX, this is shown in Figure 2.

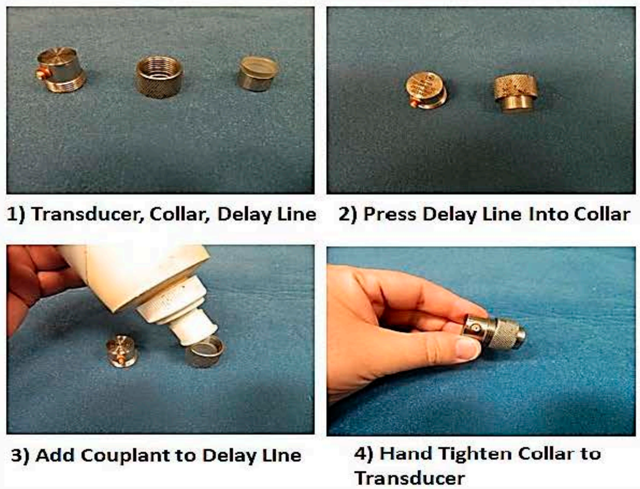

Figure 1: Secure the Delay Line to the Transducer

Figure 2: OmniScan MX Conventional Transducer Connector

4.2 Calibration of the Ultrasonic Unit

Calibration can be accomplished with either a step wedge or a tapered wedge. Typically, the step wedge is preferred for calibration, due to the scattering effect a tapered wedge has on the sound beam path. This scattering can potentially cause a reduction in signal strength. For this general training guide, the step wedge is used.

Step 1: Manually set the ultrasonic velocity of the material to 0.114in/µs or 2900m/s, the ultrasonic velocity of the carbon material.

Step 2: Ensure the instrument is in conventional inspection mode. Set the X-axis to display thickness/depth in inches.

Step 3: Apply couplant to the surface of the reference standard step wedge (ST8871 shown in Figure 3).

Note: It is critical to ensure that the calibration block is made from the same material and build-up as the aircraft part being inspected.

Step 4: Ensure that the delay line is thick enough to inspect the thickest region of interest. If the delay line is too thin the interface signal will interfere with back wall signal. The maximum thickness of the proficiency specimen set is 0.48".

- Standard Lucite delay line of 0.75" are useful for structure up to .94" or 24 mm.
- Standard Rexolite 0.75" delay – 0.975" of carbon
- Standard ZIP – Impedance matched material – 1.01" of carbon

Step 5: Use the delay controls to put the front surface signal at the left edge of the screen. Move the delay line signal so that it peaks at 0.0 inches. See reference graphic in Figure 4.

Step 6: Use the range controls to display up to 0.8" depth

Note: If the approximate thickness of material being inspected is known, a range adjustment to 10 % greater than the thickness can be used.

Step 7: Place the transducer on the 0.600" (15.24 mm) step of the step wedge

- Ensure that the back wall signal reflector peak is at 0.6" depth. If not, adjust the delay control.

Note:The range is now set to inspect laminate up to 0.6" thick

Step 8: Put the transducer on the 0.300" (7.62 mm) step and ensure that the back wall reflector peak is at 0.3" on the screen. Check the remainging steps on the step wedge that are less than 0.6".

Note: Since the Time Corrected Gain has not been set yet, you may need to manually adjust the gain setting to increase or decrease the amplitude of the back wall signal.

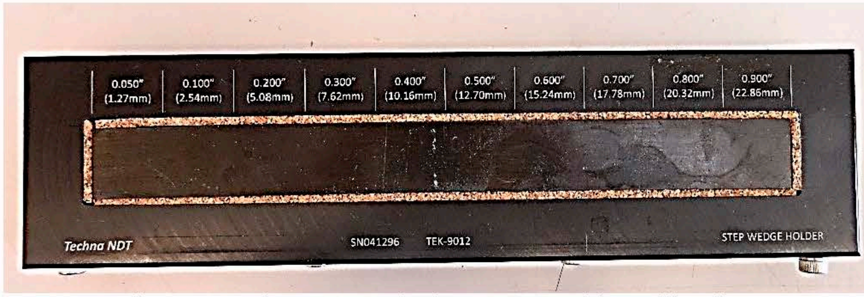

Figure 3: Reference Standard ST8871 Used for Calibration

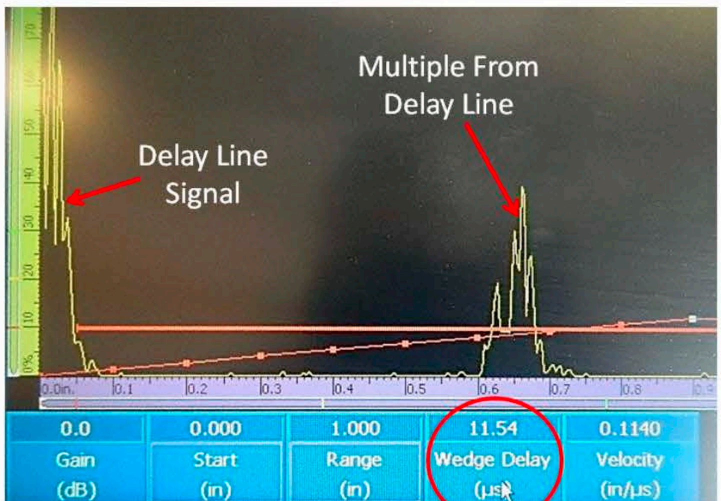

Figure 4: A-Scan Showing Delay Line Signal Moved to the Left of the Screen and the Resulting Delay Line Multiple using a 3.5 MHz Transducer with a 1/2" Rexolite Delay Line (Note: ¾" Delay Line Multiple will Occur Further to the Right)

4.3 Setting the Time Corrected Gain

Use the steps of the ST8871 reference standard to manually set the Time Corrected Gain (TCG).

Note: It is important to ensure that your calibration points extend past the thickest region of the part to be inspected. Always verify thickness in the component drawings, aircraft manuals, or other manufacturing inspection requirements.

If using an OmniScan Instrument start the Wizard > Calibration > Sizing > TCG, to set the following parameters. If using a different, but equivalent instrument, proceed through the remainder of section 4.3 setting the TCG of your instrument per the manufacturer's instructions.

Step 1: Place couplant on reference standard ST8871.

<u>Step 2:</u> Start the Gate at 0.04" or just after the front surface signal, making sure the 0.05" step can be captured in the gate. See Reference graphic in Figure 5 A.

<u>Step 3:</u> Set the Ref Amplitude to 80%.

<u>Step 4:</u> Set the start of the A-Scan window to 0.0" and the Range to 0.7"

<u>Step 5:</u> The Gate start should already be set to 0.04". Set the Width to 0.7", and the threshold to 30%.

Caution: Do not extend the gate into the delay line multiple signal. The amplitude of the delay line multiple will swamp any other signals of interest.

<u>Step 6:</u> Position the transducer on the 0.05" step, making sure the signal is in the red gate, and its peak is below 75% full screen height (FSH). Add the TCG point.

Note: If the instrument is reading 0.0 dB and the signal is still over 100% FSH activate the attenuation feature, UT Settings > Advanced > Attenuation = On

<u>Step 7:</u> Repeat steps 4 through 6 to add TCG points at 0.05", 0.1", 0.2", 0.3", 0.4", 0.5", 0.6", and 0.7". Refer to Figure 5 A, B, C.

Note: When the TCG points are properly calibrated, each having their own gain setting, every step across the carbon step wedge will provide a back wall signal reflection of 80% FSH (see Figure 5).

<u>Step 8: Check that the amplitude is approximately 80% FSH on all steps from thickness 0.05" to 0.7". If an area is not displaying correctly, reset the TCG curve and repeat Section 4.3</u>

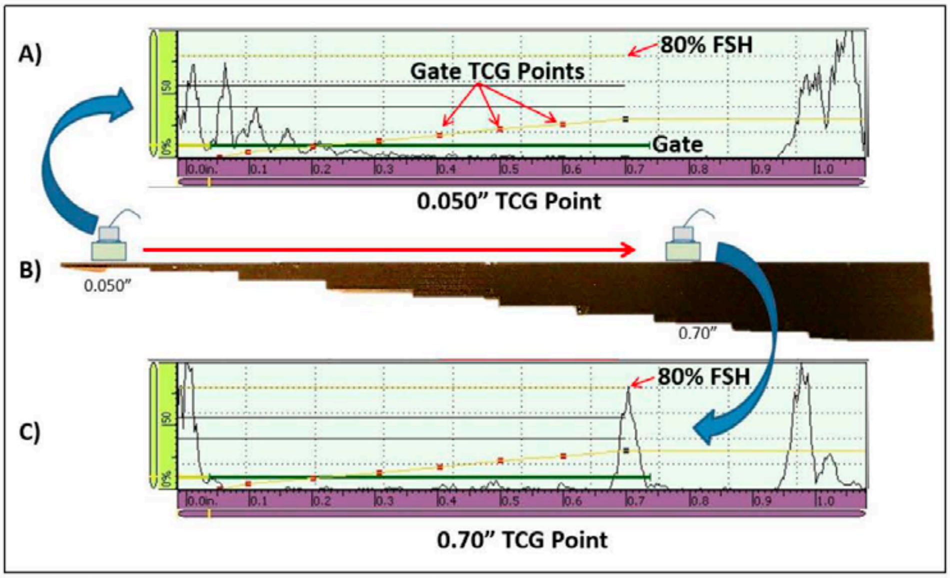

Figure 5: OmniScan A-Scan Presentation Showing Time Corrected Gain Points (TCG) using a 3.5 MHz, 0.5" Diameter Transducer with a 0.75" Delay Line. (Gate – green bar at 10% FSH, 8 TCG Points – Red squares)

***CAUTION:** Subsequent modification of velocity may affect any time corrected gain (TCG) settings that have been programmed.*

4.4 Conducting the Inspection

On an actual aircraft the first step is to visually inspect the area of potential damage. The Proficiency Specimens do not contain visual indications of defects, but this is good practice. Many times composite aircraft damage will be visible to the eye, but is typically larger than what is visulaly indicated. Ultrasonic inspecion is the primary inspection technique when sizing and characterizing this sub-surface damage.

When scanning over the area of possible damage, use a straight edge indexing ½ the transducer diameter and extend the scan area (as required per manufacturers instructions) past any suspected damage. Where the back wall signal decreases by 50% or shifts to the left of the backwall signal, stop the movement of the transducer and mark a point at the center of the transducer with water resistant crayon. After indexing over the entire suspect area, connect the points to show the outline pattern of the damaged area.

Note: Equipment calibration and the TCG curve should have already been set following 4.2 and 4.3.

Step 1: Visually inspect the surface of the part for any indications of damage.

Step 2: Ensure TCG curve is set to "on."

Step 3: Determine the thickness of the laminate by measuring or use drawings.

- If the part has multiple thicknesses due to substructure or tapered regions, measure the maximum thickness of panel (T_{max}) and the minimum thickness (T_{min}).

Step 4: Ensure that the delay line is thick enough to inspect the thickest region of interest.

Step 5: Place the transducer on a good area of the part. The back wall signal should be around 80% amplitude using the preset TCG curve when the transducer is placed in contact with the part. If not, adjust the gain to put the back wall signal at 80%. Check that the measured ultrasonic thickness and the physically measured thickness are similar.

Step 6: Mark the location of substructure elements and thickness changes using a water resistant crayon as seen in Figure 6. Use a combination of the drawings provided and hand scanning across substrucure elements. A-Scan indications of different structural elements are as follows:

- **Edge of Stiffener** - There is no back wall signal and the location where signal loss occurs is linear and aligned with the stiffener. See reference graphic in Figure 7.
- **Tapered Skin Region** - The back wall signal gradually shifts to the right. If TCG is on then amplitude should not decrease. See reference graphic in Figure 8.
- **Bonded Stiffener Flange or Bonded Part** – The back wall signal (now the bond line signal) decreases and a new back wall signal appears to the right of the back wall. See reference graphic in Figure 16b.
- **Presence of backside sealant/acoustic tile** – The back wall signal decreases in amplitude. No new intermediate signal appears. There is no increase in baseline noise. See reference graphic in Figure 10.

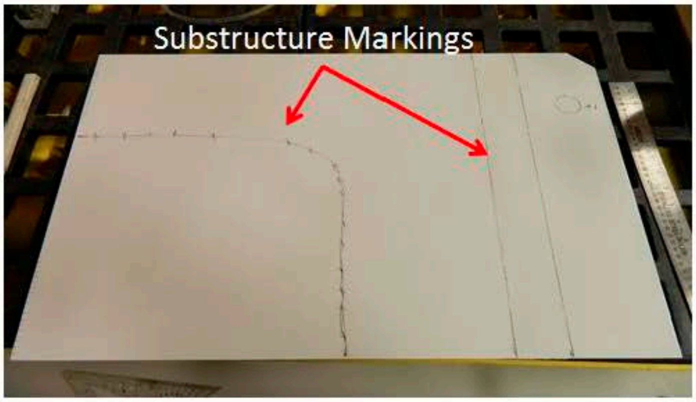

Figure 6: Example Proficiency Specimen with the Substructure Marked on the Surface of the Panel

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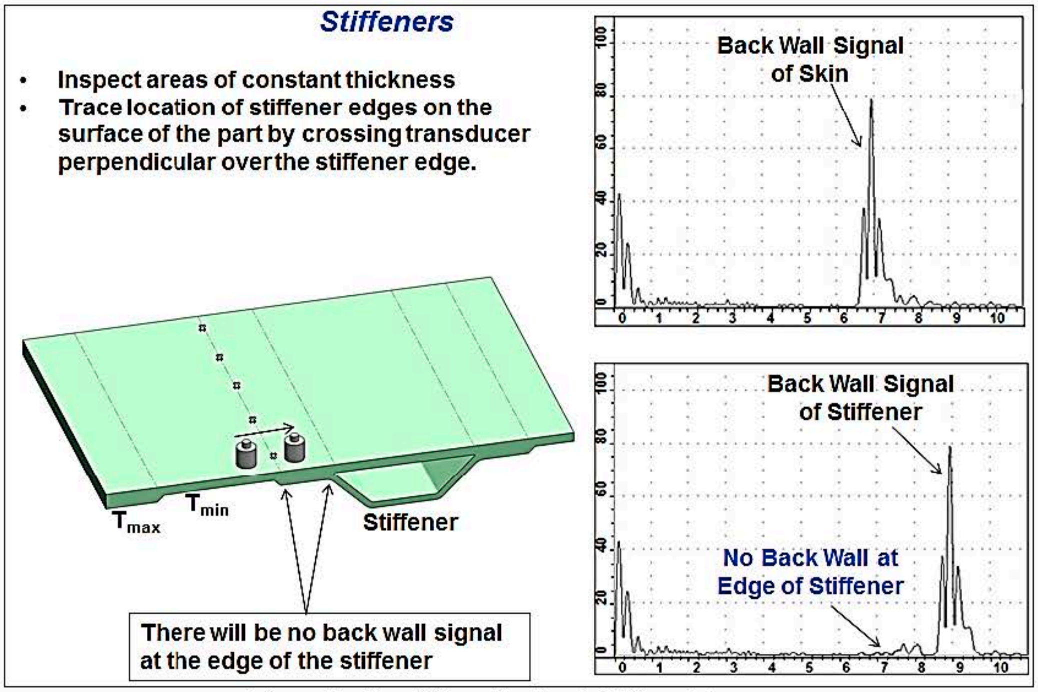


Figure 7: Identifying Co-Cured Stiffener Flanges

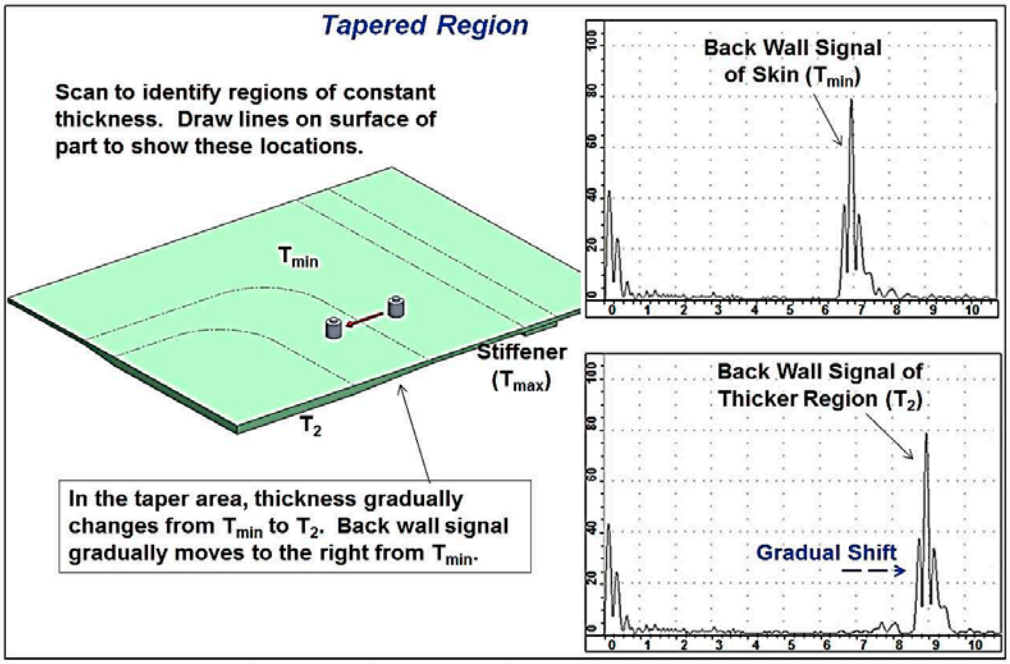


Figure 8: Identifying and Marking Tapered Region

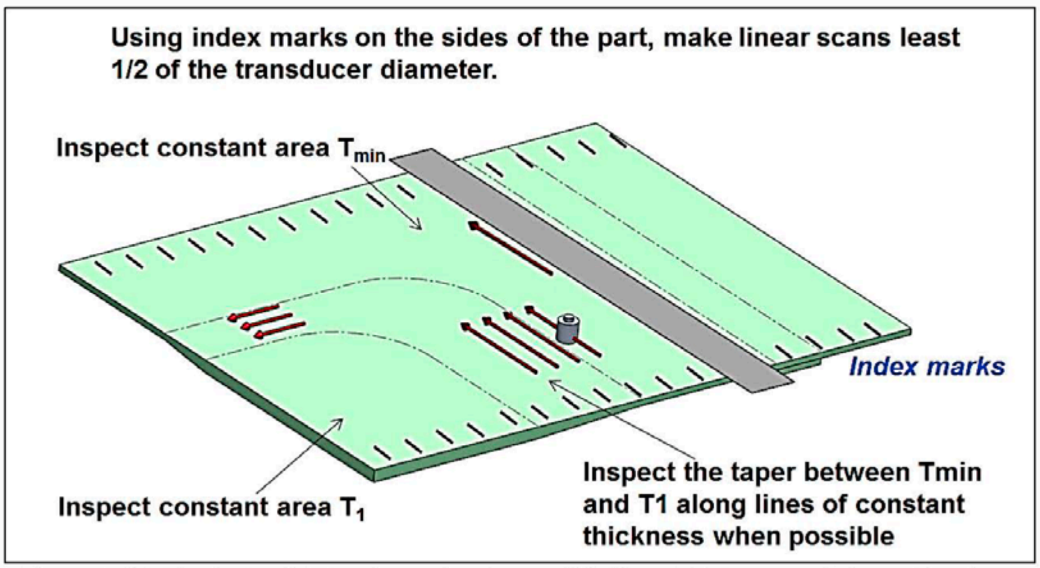


Figure 9: Making Index Marks, Inspecting Areas of Uniform Thickness and Tapered Regions

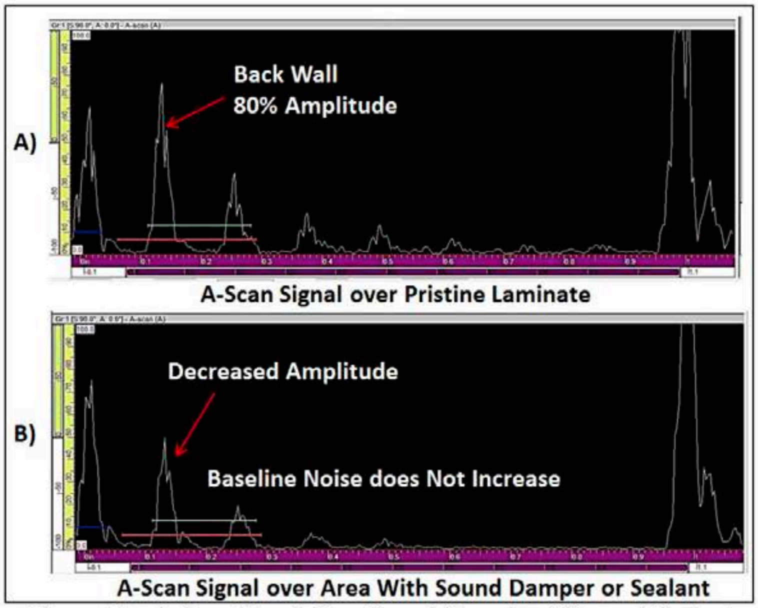
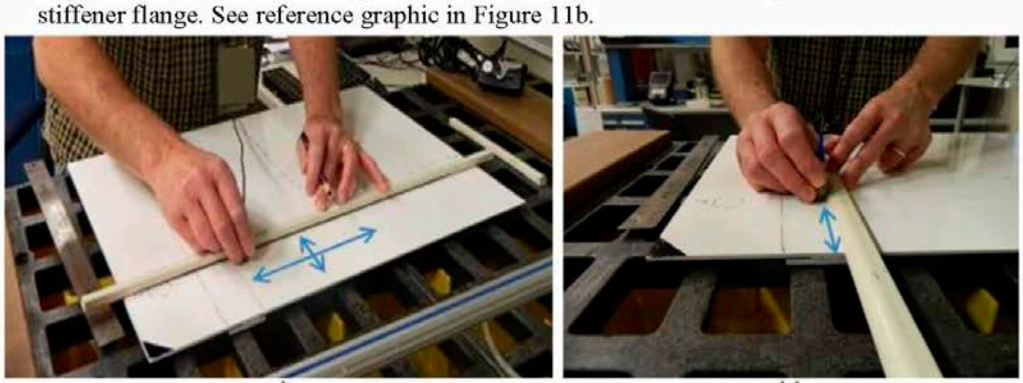


Figure 10: A-Scan Signal Over Sound Damping Tiles and Sealant

Step 7: Make marks (index marks) along two parallel edges of the part that are no larger than 1/2 of the probe diameter. See index marks in Figure 9.

Step 8: Use the edge marks to make parallel scans along a straight edge over uniform thickness regions of the part. This is also shown in Figure 9.

- Inspecion in the taper region should be done along lines of constant thickness when possible.
- Inspection in the stiffener flange areas shouldbe done in the direction parallel to the stiffener flange. See reference graphic in Figure 11b.



a) b)

Figure 11: (a) Example of Scanning along a Straight Edge using Indexing Marks in Either the Horizontal or Vertical Direction over Uniform Thickness Regions and (b) Scanning Parallel over Substructure Element

Step 9: While scanning along the straight edge, ***mark the edge of the flaw where the back wall signal amplitude drops by 6 dB***. See reference graphic in Figure 12, Figure **13**, Figure 14.

Step 10: Mark all flaw indications. See reference graphics in FiguresFigure **14**Figure 15 Figure 16.

Note: On actual components follow manufacturer procedures for identifying/calling defects.

- Uniform thickness – Mark indications where the ***back wall signal decreases more than 6 dB and the defect is greater than 0.5 inch in diameter***. See reference graphic in Figure 15b.
- Taper Region – Mark indications where the ***back wall signal decreases more than 6 dB and the defect is greater than 0.5 inch in diameter***.
- Bonded Stiffeners – Mark indications where the ***back wall signal decreases by 6 dB from an adjacent bonded stiffener region, or the bond line signal increase by 6 dB, and the defect is greater than 0.5 inch in diameterer***. Defect may also be present in the skin lamiante over the bondline and in the stiffener element. See reference graphic in Figure 16.
- Co-Cured Stiffeners - Mark incications where the ***back wall signal decreases more than 6 dB and the defect is greater than 0.5 inch in diameter***. See reference graphic in Figure 15.

Note:The back wall signal amplitude can decrease for multiple reasons including dampening materials and sealants. Investigate the surounding area to identify external causes.

4.4.1 Identifying Defects

A-Scan screen indications of defects:

- Skin Delamination - The back wall signal decreases out of view and a new intermediate signal occurs to the left of the original back wall signal. See reference graphic in Figure 14.
- Near Surface Delamination - The back wall signal decreases out of view and no new signal occurs. The front surface signal increases.
- Light Concentrated Porosity - The back wall surface decreases and slightly shifts to the right – possibly slight intermediate signal and increase in baseline noise.
- High Concentrated Porosity - The back wall signal decreases out of view and baseline noise increases – possibly intermediate signal where porosity is concentrated.
- Grease Contamination - The back wall surface decreases and an intermediate signal appears.
- Delamination in the stiffener flange (co-cured) - The stiffener flange back wall signal goes out of view and a new signal occurs to the left of the stiffener flange back wall. See reference graphic in Figure 15d.
- Skin-to-Stiffener Disbond - The back wall signal from the stiffener goes out of view and a new signal appears at the location of the back wall of the skin. See reference graphic in Figure 15c.

- Delamination in the stiffener flange (bonded) - The stiffener flange back wall signal goes out of view and a new signal occurs to the left of the stiffener flange back wall. See reference graphic in Figure 16c.

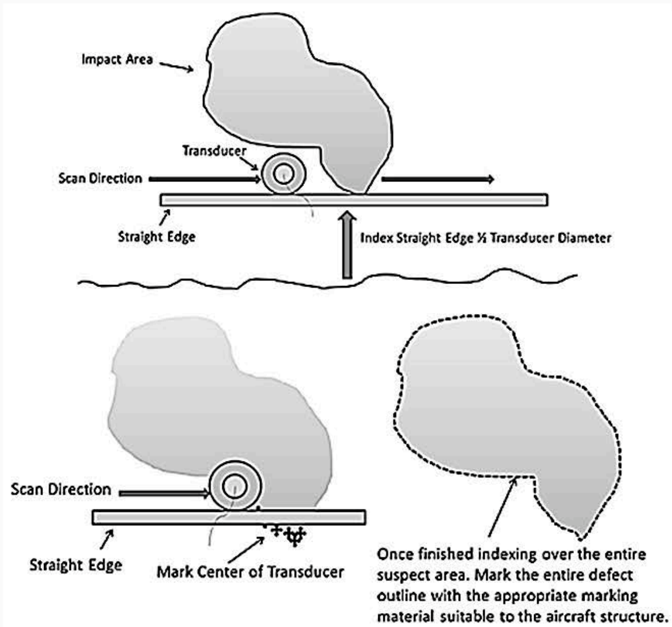


Figure 12: Mapping Damage Using Straight Edge on the Inspection Article

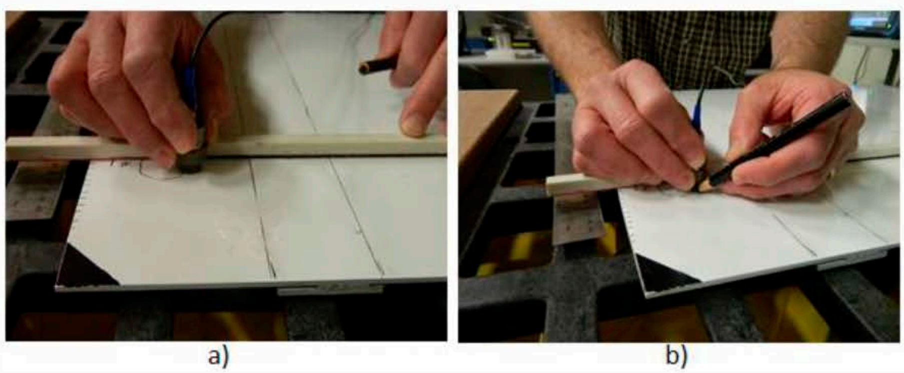


Figure 13: (a) Example of Stopping Transducer at Edge of Defect and (b) Marking Center of Transducer at Edge of Defect

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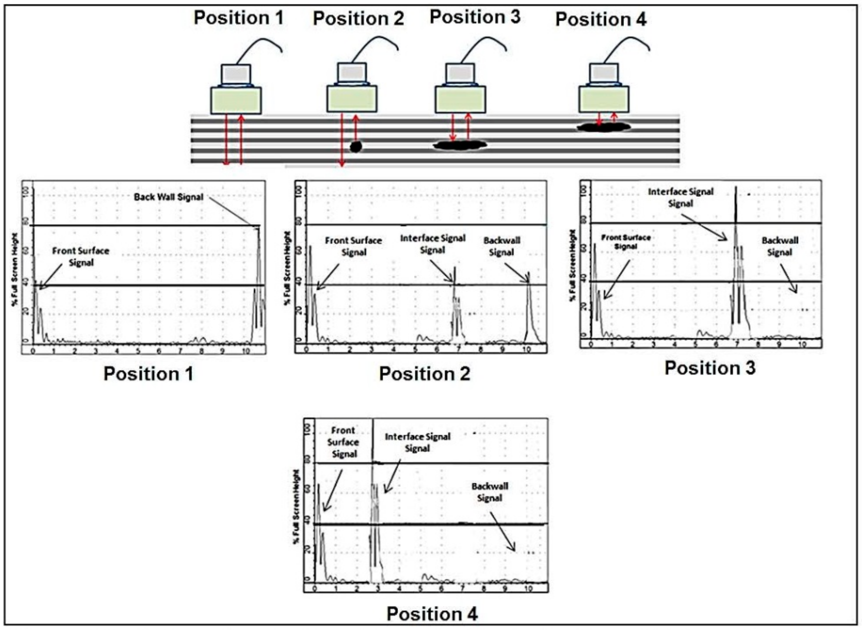

Figure 14: A-Scan Signals from Delamination in Uniform Thickness Laminate

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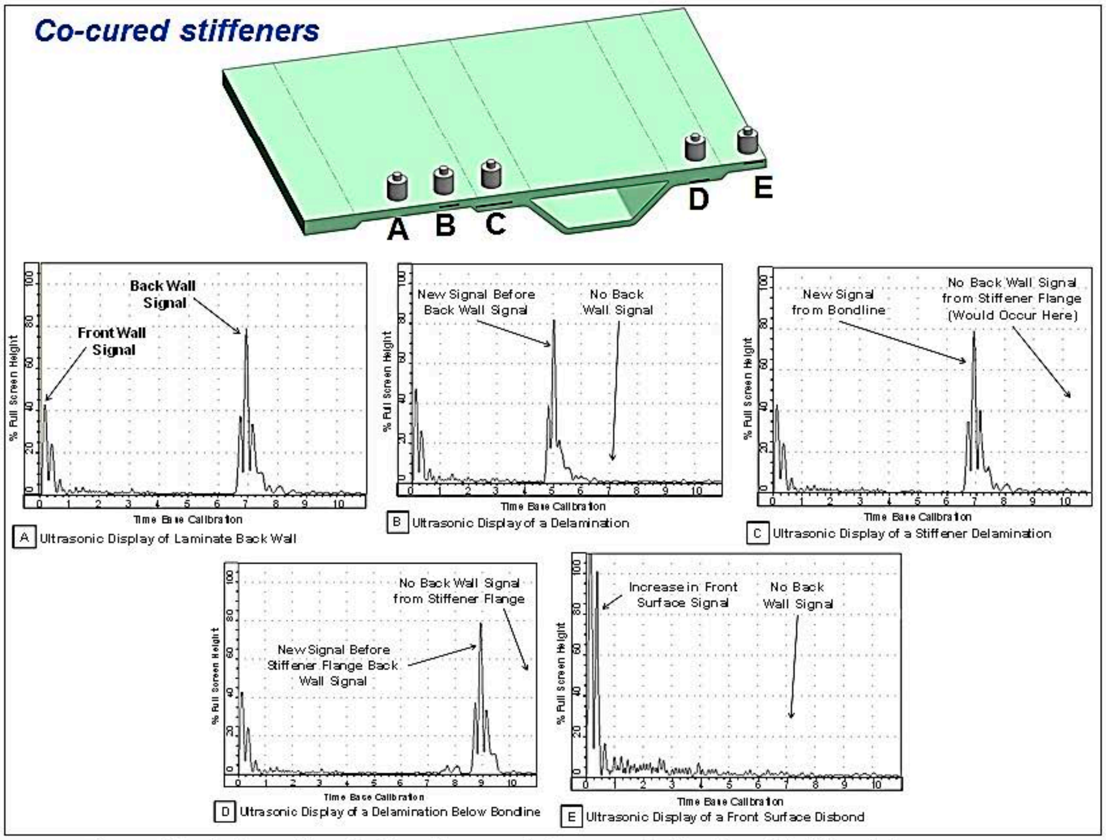

Figure 15: A-Scan Signals from Damaged Structure in Co-Cured Stiffened Areas

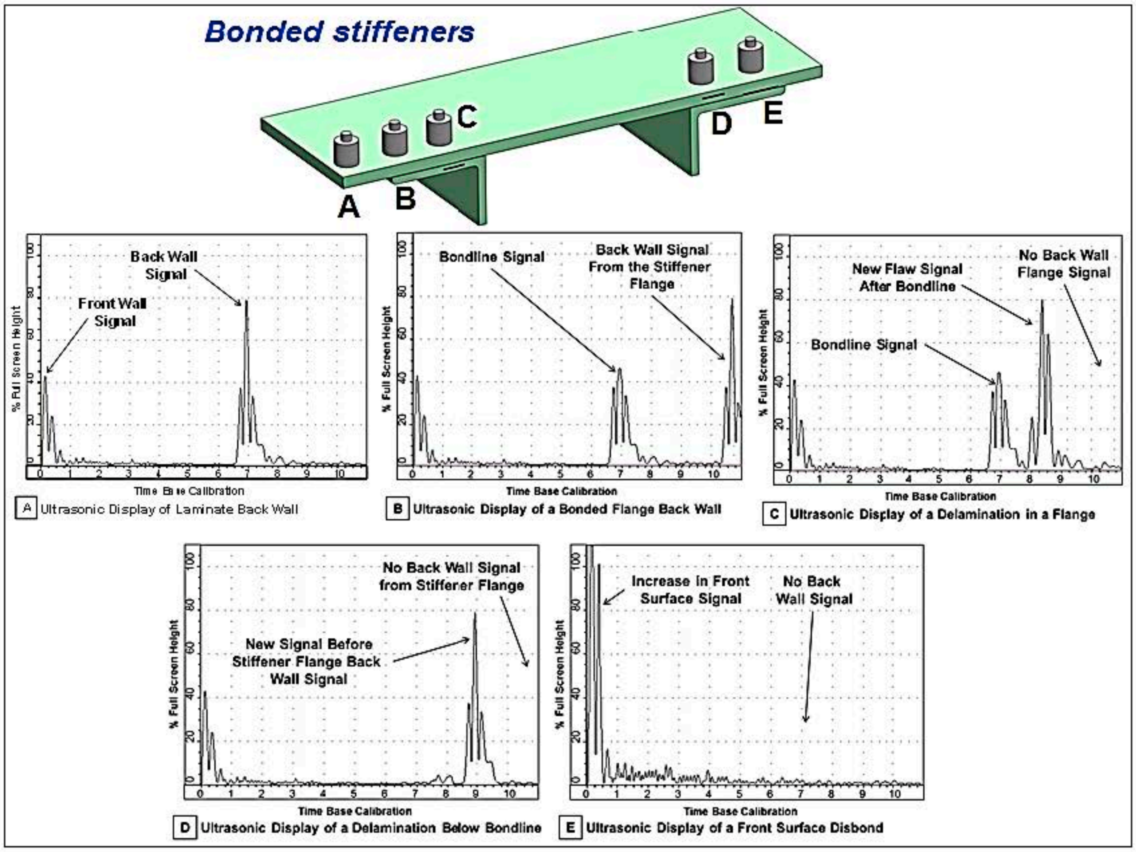

Figure 16: A-Scan Signals from Damaged Structure in Bonded Stiffener Areas

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Composite Inspector Training Course

General Ultrasonic Carbon Laminate Inspection Guide (Conventional C-Scan)

FAA Airworthiness Assurance NDI Validation Center (AANC)
Sandia National Laboratories

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Federal Aviation Administration
FAA William J. Hughes Technical Center, Atlantic City International Airport, NJ 08405
Office of Aviation Research, Washington, D.C. 20591

Sandia National Laboratories
PO Box 5800
Albuquerque, NM 87185

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List of Figures

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Note: This training guide is applicable to the nondestructive inspection of carbon fiber laminated structures and to be used as a general guide during training, not to override any procedures developed by any specific aircraft manufacturer or airline.

2 Applicability of Training Guideline

This training guide is for ultrasonic inspection of composite materials and bonded laminates using an array transducer to generate C-scan inspection results. It generally applies to laminates built with unidirectional or woven ply configurations and not sandwich structures. Although there are many ultrasonic units commercially available and capable of inspecting solid laminate composites, the Olympus OmniScsan unit was selected for demonstration purposes in this training guide due to its wide range of capabilities and common usage across the aerospace sectors.

Since T800 pre-preg tape was used to fabricate the Proficiency Specimens, reference standards made from this material and other reference standards used to set up equipment to inspect this material were used. When inspecting actual aircraft components follow manufacture procedures and set up on reference standards designed to be used with the material the component is made from.

Note: A standard single element transducer attached to a two dimensional encoder can be used to generate a C-scan display, but that is not included in this general procedure.

3 List of Materials

- Sandia/FAA Proficiency Specimens
- OmniScan MX, MX2, or equivalent
- 3.5L64-NW1 Olympus Phased Array Probe or equivalent
- Olympus SNW1-OL-WP5 Delay Line or Rexolite ABWX831 Delay Wedge
- X-Y Glider or equivalent
- Couplant - Water or Gel
- Measuring tool – Micrometer recommended to measure thickness of panels and ruler for flaw size
- Pencil, Aircraft Marking (China marker, crayon)
- Step Wedge – ST8871

4 General Inspection Procedure

4.1 Set up the Equipment

Step 1: Apply couplant to the transducer and secure it to the delay line by sliding it horizontally into the groves of the delay line then tighten it with the four screws. See Figure 1.

CAUTION: Tightening the screws too tight can cause the delay to crack.

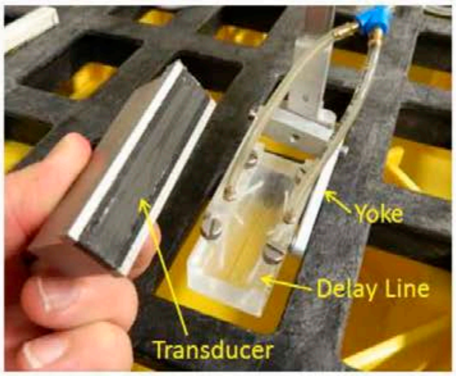

Figure 1: Connecting the Transducer to the Delay Line

Step 2: Connect the cables from the transducer and the X-Y glider to the ultrasonic unit following manufacturer instructions. The OmiScan MX2 connector locations are shown in Figure 2.

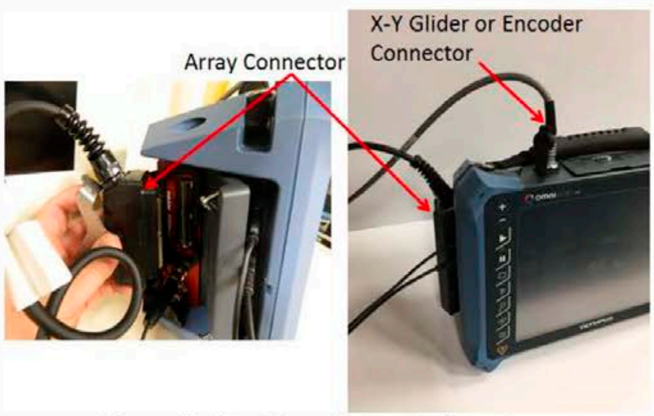

Figure 2: OmniScan Connector Locations

4.2 Calibration of the Ultrasonic Unit

Calibration can be accomplished with either a step wedge or a tapered wedge. Typically, the step wedge is preferred for calibration due to the scattering effect a tapered wedge has on the

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sound beam path. This scattering can potentially cause a reduction in signal strength. For this general training guide, the step wedge is used.

Note: If a pre-developed set up file is provided by the instructor for composite laminate defect detection, load the file and follow the calibration steps to check the set up prior to the inspection.

If using an OmniScan Instrument, start the Wizard > Setup, to set the following parameters. If using a different, but equivalent instrument, proceed through the remainder of section 4.2, instrument parameter setup, per the manufacturer's instructions.

Step 1: Start the Wizard > Setup. Confirm the instrument is in Phased Array Mode and set the application to Composite. Press Next.

Step 2: Enter the probe and wedge information if it was not auto recognized. Press next.

Step 3: Set the Element Quantity in an aperture to between 5 and 8. Five is recommended. Specify the first and last element of the array. First element = 1, and last element = # of elements in the transducer. Press Next, then Generate. This will be the end of Wizard settings.

Step 4: Check that the ultrasonic velocity of the material is set at 0.114in/μs or 2900m/s.

Step 5: Set the frequency to the frequency of the transducer.

Step 6: Set the receiver Filter to none, or broadband.

Step 7: Set the scan mode to linear at 0. This will only be required on older OmniScan software versions and other equivalent inspection systems.

Step 8: Display an active S-scan on the unit. Hold the transducer in air and check that the signal reflection from the back surface of the delay line is consistent. If not, check coupling and individual apertures in the array.

Note: If any of the apertures are out of specification, calibrate the array following manufacturer's instructions.

Step 9: Display the A-Scan for the center aperature of the phased array probe.

Step 10: Apply couplant to the surface of the ST8871 reference standard step wedge shown in Figure 3. Either water or gel can be used.

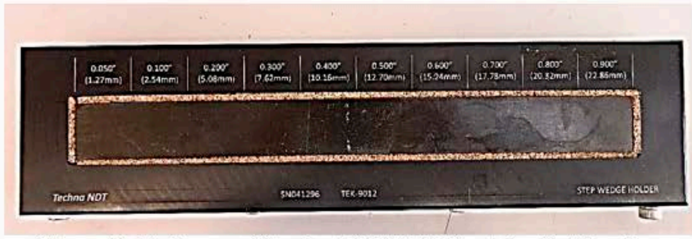

Figure 3: Reference Standard ST8871 Used for Calibration

Step 11: Ensure that the delay line is thick enough to inspect the thickest region of interest
Note: If the delay line is too thin, the multiple from the delay line will interfere with back wall signal.

- The standard Lucite delay line for the array is about 20 mm. thick and is useful for structure up to 0.9" or about 22 mm. The maximum thickness of the profeciency specimen set is 0.48".

Step 12: For setup purposes, set the range controls to display 1.0 " depth so that the delay line multiple is visible on the A-scan as shown in Figure 4.

Step 13: Use the wedge delay controls to put the front surface signal at the left edge of the screen, so it peaks at 0.0 inches. The wedge delay line signal and the multiple from the delay line are shown in Figure 4, at approximately 0.95" apart.

Figure 4: A-Scan Showing Delay Line Signal Moved to the Left of the Screen and the Resulting Delay Line Multiple

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Step 14: Place the transducer on the 0.600" (15.24 mm) step of the step wedge ensuring that the back wall signal reflector peak is at 0.6" depth. If not, adjust the wedge delay so the signal peaks at 0.6".

Step 15: Put the transducer on the 0.300" (7.62 mm) step and ensure that the back wall reflector peak is at 0.3" on the screen. Check the remaining steps on the step wedge.

4.3 Setting the Time Corrected Gain

Use the steps of the ST8871 reference standard to manually set the Time Corrected Gain (TCG) of the unit.

Note: *It is important to ensure that your calibration points extend past the thickest region of the part to be inspected. Always verify thickness by physical measurement, component drawings, or aircraft manuals.*

If using an OmniScan Instrument start the Wizard > Calibration > Sizing > TCG, to set the following parameters. If using a different, but equivalent instrument, proceed through the remainder of section 4.3 Setting the TCG of your instrument per the manufacturer's instructions.

Step 1: Place the transducer on the 0.05" step of the ST8871 Reference standard. Adjust the gain so the signal peak is between 50% and 75% Full Screen Height (FSH).

Step 2: Press the "Reset All" button for the TCG to clear any previous curves. Press start for the Wizard > Calibration > Sizing > TCG.

Step 2: Click next at the first menu.

Step 3: Start the Gate at 0.04" or just after the front surface signal, making sure the 0.05" step can be captured in the gate.

Step 3: Set the Ref Amplitude to 80%. Click Next.

Step 4: Make sure all probe apertures are active for the TCG setup. Click Next.

Step 5: Set the start of the A-Scan window to 0.0" and the Range to 1.0". Click Next.

Step 6: The Gate start should already be set to 0.04". Set the Width to 0.85", and the threshold to 30%. Click Next.

Caution: Do not extend the gate into the delay line multiple signal. The amplitude of the delay line multiple will swamp any other signals of interest.

Step 7: Position the probe on the 0.05" step of the reference standard so that the edge of the probe with element number 1, is at the edge of the standard. See the left image in Figure 5. Slide the probe across the 0.05" step so that all of the elements see the step thickness, Figure 5, middle and right images. As the probe is slid across the reference standard, a green envelop line will be drawn across all apertures. See Figure 6. When all apertures have a good signal from the step wedge a straight green line will remain on the screen. At this point the TCG curve point can be added.

Figure 5: Showing the progression of sliding the probe across the 0.05" step

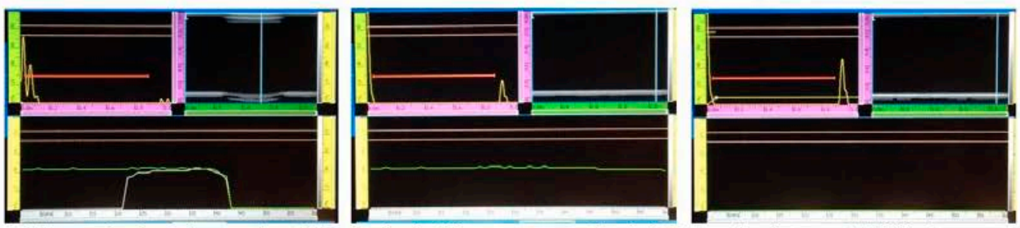


Figure 6: Left image is of the probe half way across the 0.05" standard step. Middle image shows a good signal from the 0.05" step across all apertures, ready to add the TCG point. The last image shows the first TCG point on the A-Scan window.

Note: The green envelope line must be below 75% for the point to be accepted.

Step 8: Repeat Steps 3 through 6 to add TCG points at 0.05", 0.1", 0.2", 0.3", 0.4", 0.5", 0.6", 0.7", and 0.8" as shown in Figure 7.

Figure 7: Image of the final TCG curve points on the screen from steps 0.05" up to 0.8'

Step 9: Check the TCG Calibration.

A) Set the display to show one A-Scan and two C-scans. Set one C-Scan to Time of Flight (TOF) and the second to Amplitude using the same gate setup for both.
B) Select a Rainbow color pallet for the TOF display.
C) Select a black/white color (gray scale) pallet for the Amplitude display.
D) Set the TOF color pallet range to match the thickness of the reference standard.
E) Secure the transducer delay line to the yoke of the X-Y glider. When setting up the phased array transducer, be aware of the orientation of the probe elements. Ensure element 1 will be at the bottom, or zero point, of the scanning area.
F) Setup the X-Y glider on a flat surface such as a table. If an active water supply is used, set the specimen in a pan or similar water containment vessel. Otherwise, water can be sprayed onto the part and cleaned up with a towel.
G) Within the Scan menu, set the Scan and Index encoders, based on the orientation of the reference standard under the glider.
H) Check that the resolution for both encoders is set to the correct resolution listed on the individual encoders.
I) Set the area of the scan. Scan Start = 0", Scan End = 10". Index Start = 0, and Index End = (Index resolution for a one-line scan. Value will be grayed out on bottom right)
J) In the Scan > Start menu set the start mode to Reset All. Place the probe attached to the glider at the zero position on the standard. See Figure 8.
K) Press the "Start" Button.
L) Scan over the full Reference Standard being careful to have good coupling and not moving too fast that data is skipped. See Figure 9.

Note: When the TCG points are properly calibrated each having their own gain setting, every step across the carbon step wedge will provide a back wall signal reflection of 80% FSH (see Figure 9).

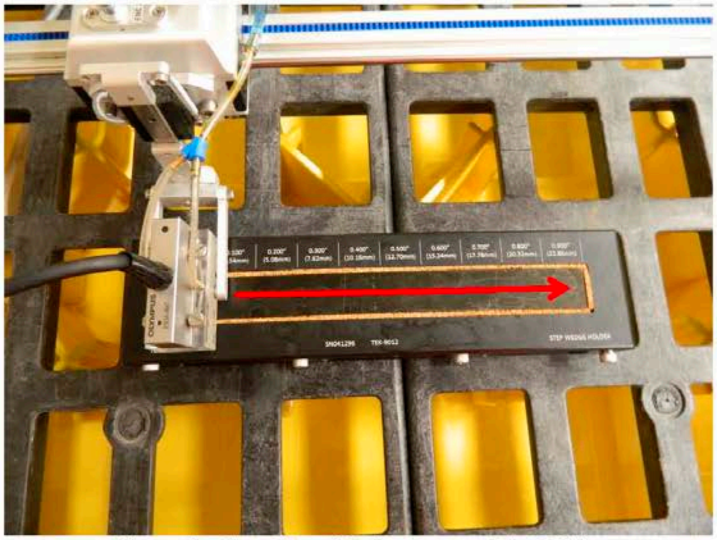


Figure 8: Scan Step Wedge Located in Holder

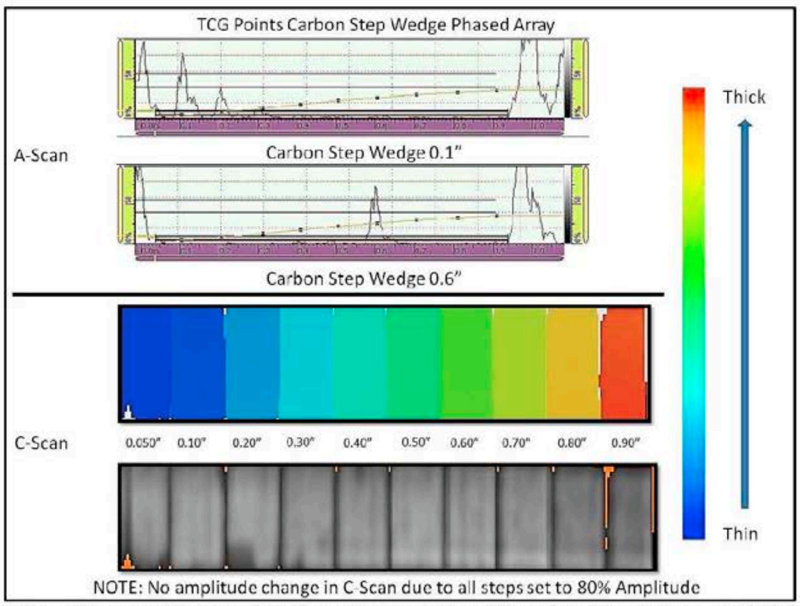


Figure 9: OmniScan A-Scan and C-Scan Presentation Showing Time Corrected Gain (TCG) Points Using a 3.5L64 Phased Array probe Time of Flight and Amplitude

Note: Also, note that the color pallet range needs to match the thickness of the standard. If it doesn't match thickness of the standard, your ability to tell flaws from non-flaws becomes diminished.

***CAUTION**: Depending on the ultrasonic unit, subsequent modification of velocity may affect any time corrected gain (TCG) settings that have been programmed.*

4.4 Setting Gates for the Inspection

Gates are used to track the amplitude and position of peak signals obtained during the inspection. For damage detection in solid laminates, time of flight (TOF) is typically used and recommended. Amplitude (Amp) C-scans in addition to TOF can also be helpful during analysis.

Step 1: For detecting defects in the parent lamainte, set three gates as shown in Figure 10.

Note: The end position of TOF and Amp gate will change depending on the thickness of the area to be inspected.

Gate	Function	Location	Threshold	Synchro	Peak Selection	Peak Measure
Interface Gate (Yellow)	Track Changes in position of Interface Signal	Start: Immediately before entry signal. End: Immediately after entry signal	10%	Pulse	Max Peak	Edge
TOF Gate (Red)	Monitor the Time-of-Flight	Start: Immediately after entry signal End: After T_{max} backwall signal	5%	I Gate	Max Peak	Peak
Amp Gate (Green)	Monitor backwall signal amplitude	Start: Before T_{min} backwall signal. End: After T_{max} backwall signal.	10%	I Gate	Max Peak	Peak

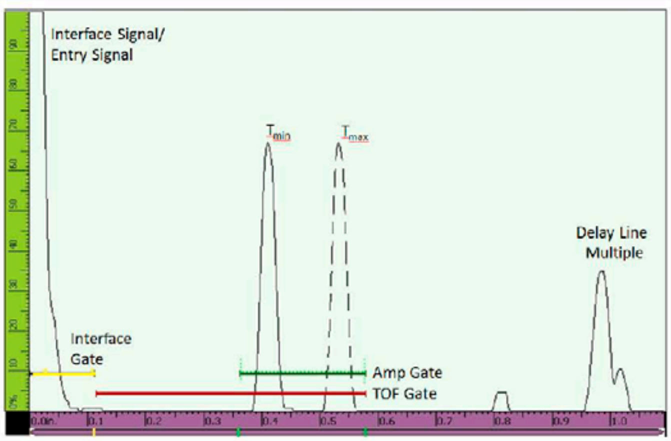


Figure 10: Image Showing Gate Locations Over the Panel Thicknesses

Step 2: For detecting defects in secondary bonded structure, set three gates as shown in Figure 11.

Gate	Function	Location	Threshold	Synchro	Peak Selection	Peak Measure
Interface Gate (Yellow)	Track Changes in position of Interface Signal	Start: Immediately before entry signal. End: Immediately after entry signal.	10%	Pulse	Max Peak	Edge
Gate 1 (Red)	Monitor the Time-of-Flight & Amplitude of the Panel	Start: Immediately after entry signal. End: After panel backwall signal.	5%	I Gate	Max Peak	Peak
Amp Gate (Green)	Monitor the Time-of-Flight & Amplitude of the Bonded Substructure	Start: Immediately after panel backwall signal. End: After Bonded substructure backwall signal	10%	I Gate	Max Peak	Peak

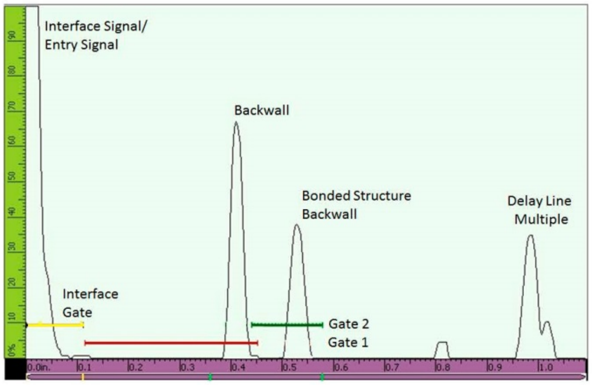


Figure 11: Image Showing Gate Setting for Second Layer Defect Detection

4.5 Scanning the Panel and Marking Substructure

On an actual aircraft the first step is to visually inspect the area of potential damage. The Proficiency Specimens do not contain visual indications of defects, but this is good practice. Many times composite aircraft damage will be visible to the eye, but is typically larger than what is visulaly indicated. Ultrasonic C-scan inspection provides a top down view of the component resulting in a broad view of the damage and component substructure. Ultrasonic C-scan has been shown to be an effective tool to accurately detect and size flaws in composite laminates.

It is more difficult to mark indications when using the larger footprint of the array transducer than single element A-scan. Further detailed sizing after a C-scan is constructed can be accomplished using singel element A-scan.

Note: Equipment calibration, delay line thickness, and the TCG curve should have already been set and checked following sections 4.2, 4.3, and 4.4

Step 1: Visually inspect the surface of the part for any indications of damage.

Step 2: Determine the thickness of the laminate - measure or use drawings.

- If the part has multiple thickness due to substructure or tapered regions, measure the maximum thickness of panel (T_{max}) and the minimum thickness (T_{min}). Ref. Figure 13.
- Check your gate settings, to ensure correct coverage based on the T_{min} and T_{max} measurements. Refer to Section 4.4.

Step 3: Place the Proficiency Specimen under the X-Y glider making sure that the panel is isolated from the inspeciton table using foam blocks or similar non-coupling material. Make sure that the panel is orriented so that the black triangle marked on the panel is loaced at the upper right corner.

Caution: When scanning the specimen check that the X-Y glider is secured to the laboratory inspection table. This will prevent the X-Y glider from slipping which may cause skewed results.

Step 4: Ensure TCG curve is set to "on." Backwall signal should be around 80% amplitude using the preset TCG curve when the transducer is placed in contact with the part.

Step 5: Mark thickness changes and substructure elements by using a combination of the drawings provided and scanning across the panel. Use a water resistant crayon to mark the surface of the panel using the following structural configuration A-scan signals:

- **Edge of Stiffener** - There is no back wall signal and the location where signal loss occurs is linear and aligned with the stiffener. Ref. Figure 14.
- **Tapered Skin Region** - The back wall signal gradually shifts to the right. If TCG is on then amplitude should not decrease. Ref. Figure 13.
- **Bonded Stiffener Flange or Bonded Part** – The back wall signal decreases and an new back wall signal appears to the right of the back wall (now the bond line signal). Ref. Figure 21b.
- **Presence of backside sealant/acoustic tile** – The back wall signal decreases in amplitude. No new intermediate signal appears. There is no increase in baseline noise. Ref. Figure 15.

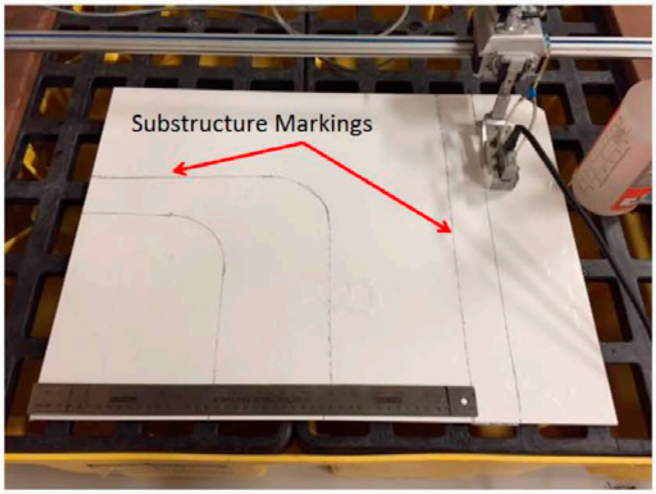


Figure 12: Example Proficiency Specimen with the Substructure Marked on the Surface of the Panel

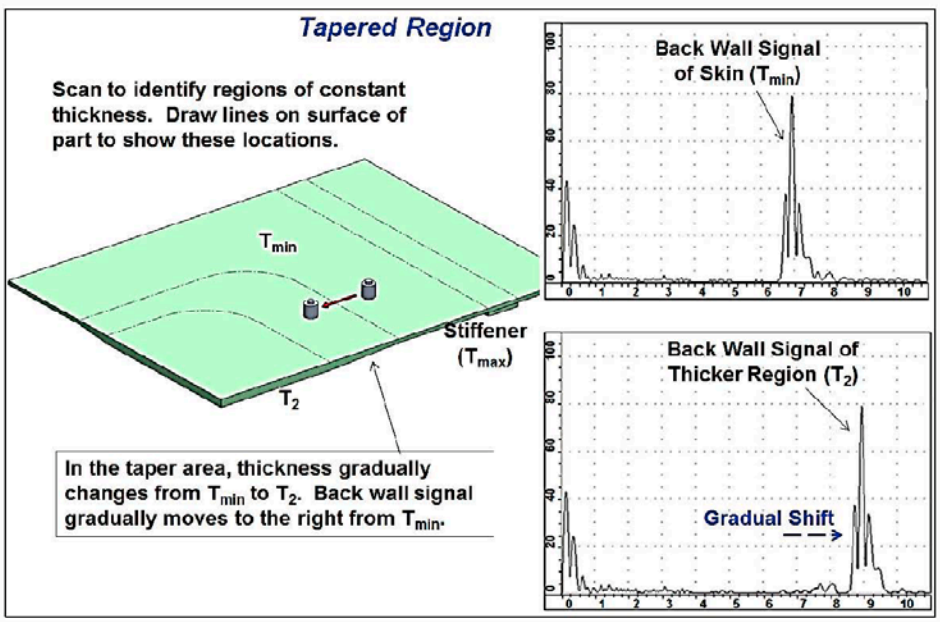


Figure 13: Identifying and Marking Tapered Region

Note regarding bonded stiffeners:Depending on bond line quality and thicknes sysmetry, gate setting may need to be changes to detect second layer flaws.

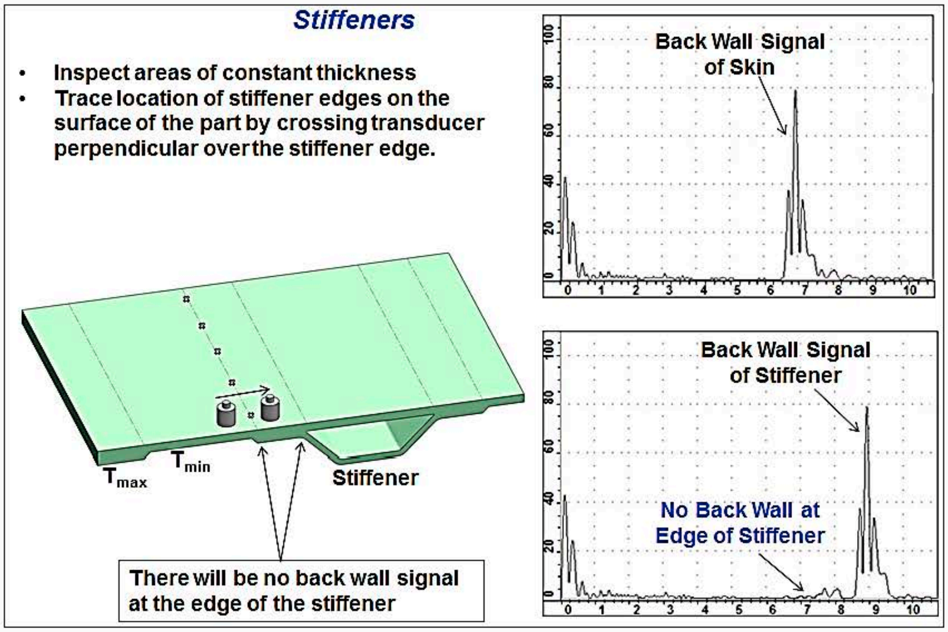


Figure 14: Identifying Co-Cured Stiffener Flanges

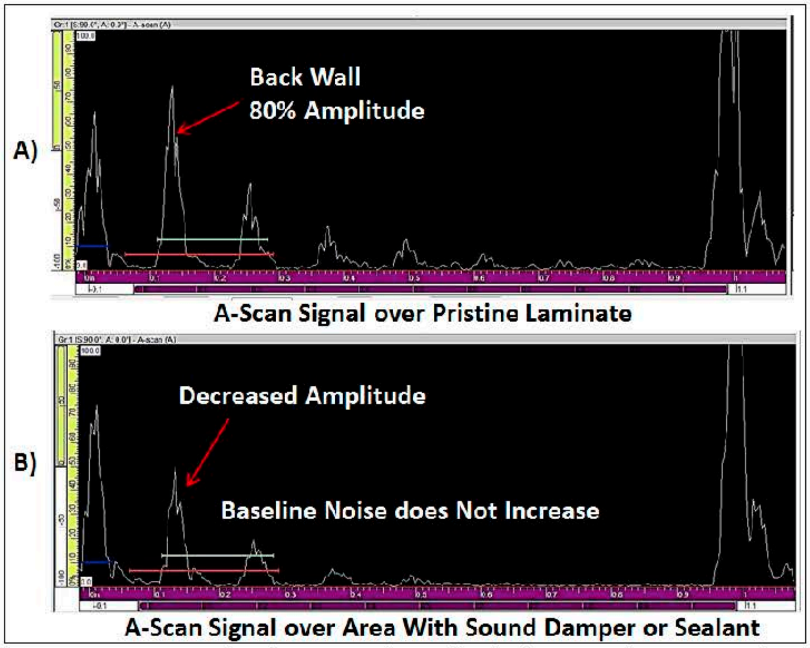


Figure 15: A-Scan Example of Decreased Amplitude due to Sealant or Sound Damper

4.6 Defect Identification

Once the panel has been scanned and the substructure has been marked on the surface of the panel, identify and mark the defects.

Step 1: Save the inspection image when you are pleased with the C-scan result.

Step 2: Examine the TOF and the amplitude C-scan results.

Step 3: Use the cursur to select points of interist on the C-scan to view A-scan reuslts. Identify and mark defect indications as follows:

Mark all flaw indications. Use reference Figure 16 through Figure 21.

- Uniform thickness – Mark indications where the ***back wall signal decreases below 6 dB or more and are greater than 0.5 inch in diamer***. Ref. Figure 20b.
- Taper Region – Mark indications where the ***back wall signal decreases below 6 dB or more and are greater than 0.5 inch in diamer***.
- Co-Cured Stiffeners - Mark incications where the ***back wall signal decreases below 6 dB and are greater than 0.5 inch in diamer***. Ref. Figure 20.
- Bonded Stiffeners – Mark indications where the ***back wall signal decreases by 6 dB from an adjacent bonded stiffener region, or the bond line signal increase by 6 dB, and the defect is greater than 0.5 inch in diamer***. Defect may also be present in the skin lamiante over the bondline and in the stiffener element. Ref Figure 21.

A-scan screen indications of defects:

- Skin Delamination - The back wall signal decreased out of view and a new intermediate signal occurs to the left of the original back wall signal. Ref. Figure 18.
- Near Surface Delamination - The back wall signal decreased out of view and no new signal occurs. The front surface signal increases. Ref. Figure 18, position 4.
- Light Concentrated Porosity - The back wall surface decreases and slightly shifts to the right – possibly slight intermediate signal and increase in baseline noise.
- High Concentrated Porosity - The back wall signal decreases out of view and baseline noise increases – possibly intermediate signal where porosity is concentrated.
- Grease Contamination - The back wall surface decreases and a slight intermediate signal appears.
- Delamination in the stiffener flange (co-cured) - The stiffener flange back wall signal goes out of view and a new signal occurs to the left of the stiffener flange back wall. Ref. Figure 20d.
- Skin-to-Stiffener Disbond - The back wall signal from the stiffener goes out of view and a new signal appears at the location of the back wall of the skin. Ref. Figure 20c.
- Delamination in the stiffener flange (bonded) - The stiffener flange back wall signal goes out of view and a new signal occurs to the left of the stiffener flange back wall. Ref. Figure 21c.

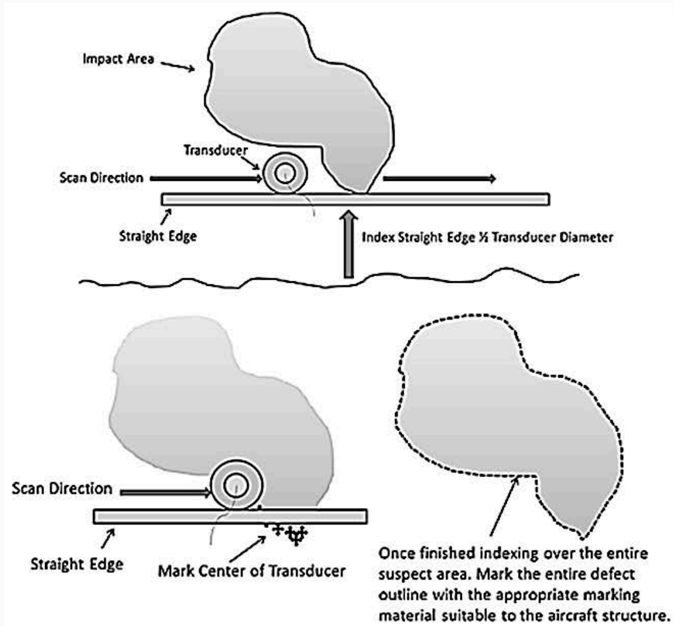


Figure 16: Mapping Damage Using Straight Edge on the Inspection Article

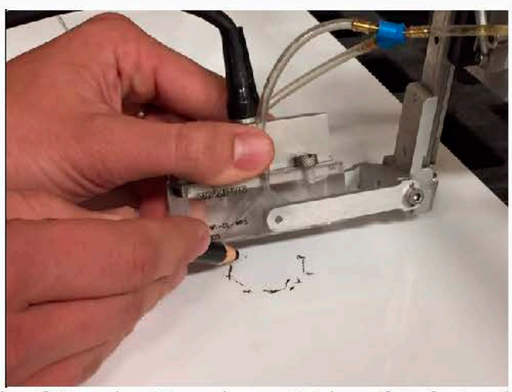


Figure 17: Example of Stopping Transducer at Edge of Defect and Marking Center of Transducer at Edge of Defect

This is a general training guide only to be used for training purposes.

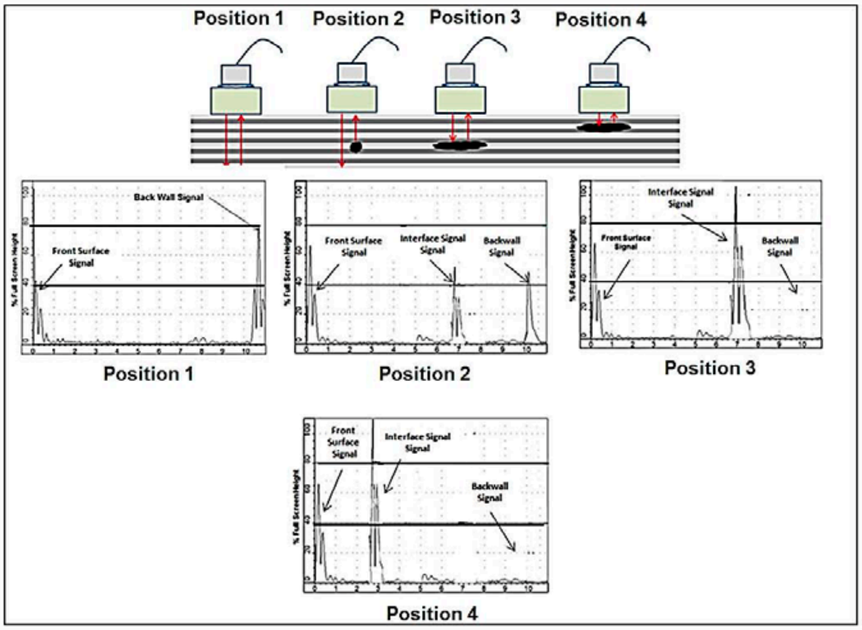
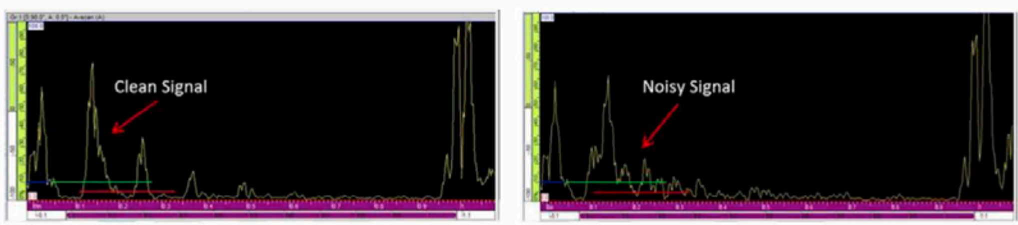


Figure 18: A-Scan Signals from Delamination in Uniform Thickness Laminate



Good Area- No Porosity

Bad Area- Porosity

Figure 19: A-Scan Image Showing Light Porosity

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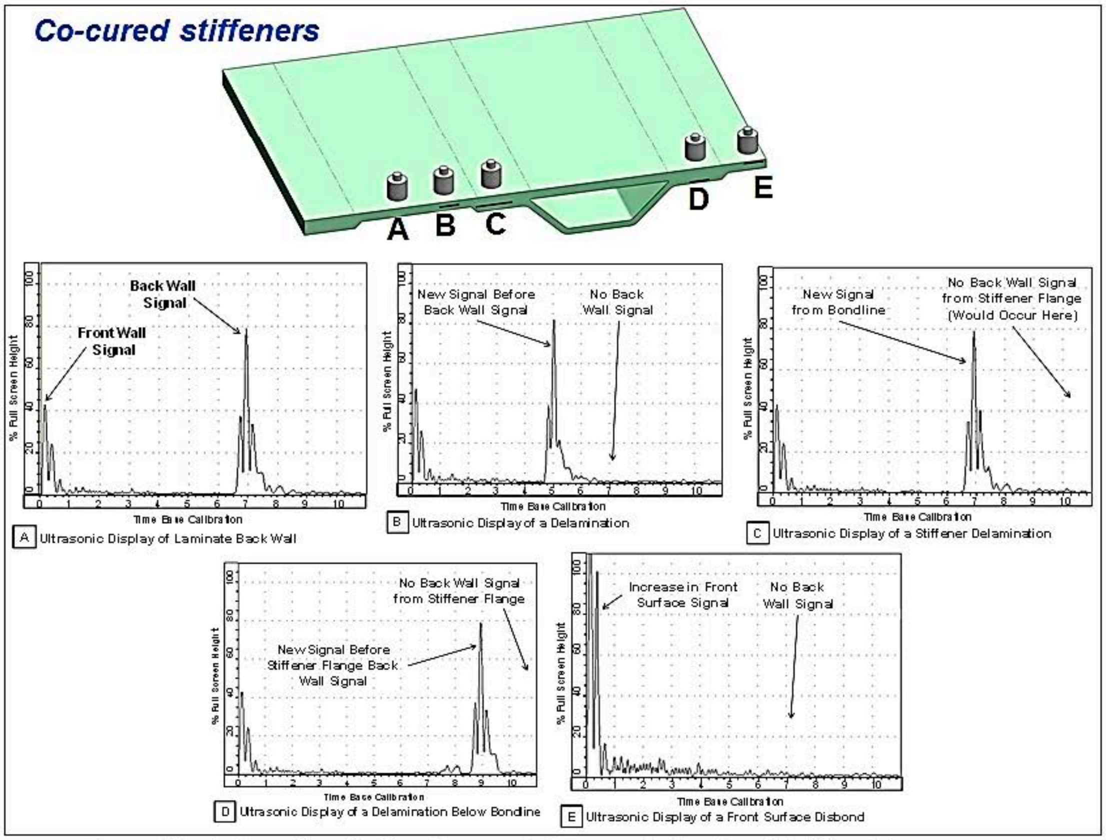

Figure 20: A-Scan Signals from Damaged Structure in Co-Cured Stiffened Areas

This is a general training guide only to be used for training purposes.

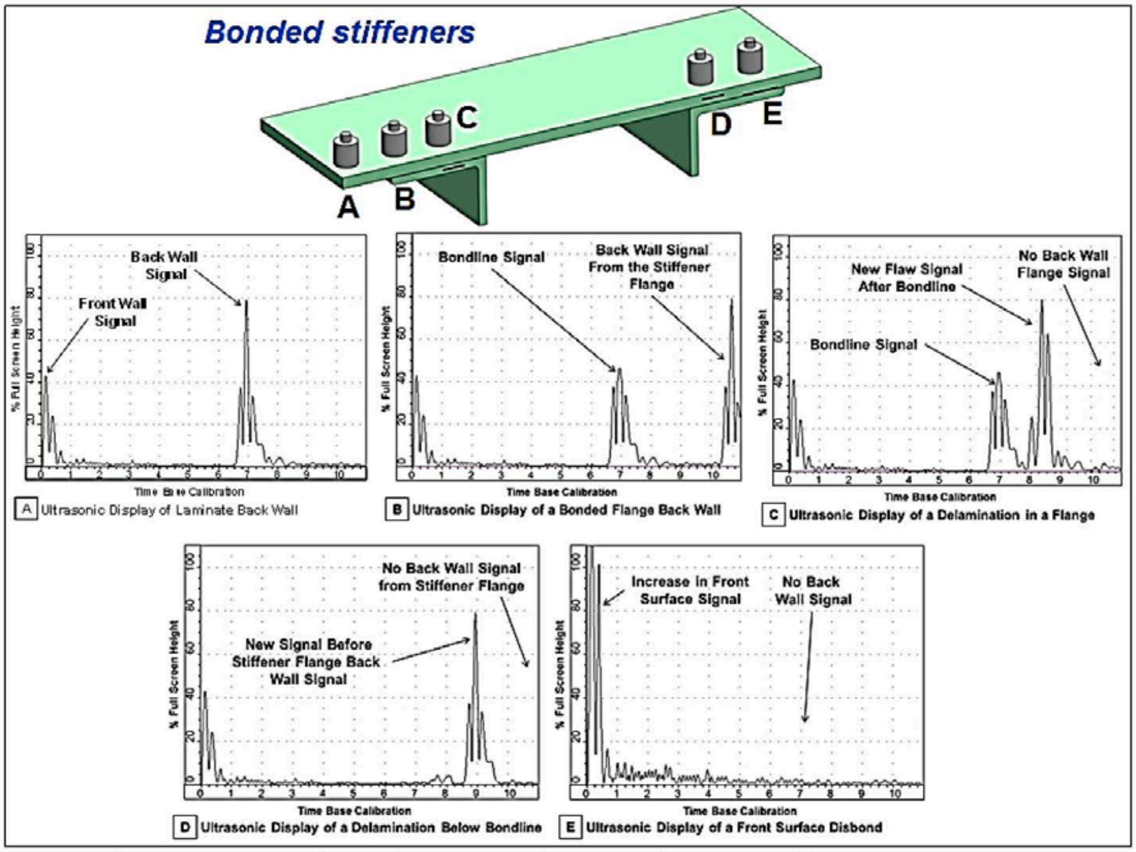

Figure 21: A-Scan Signals from Damaged Structure in Bonded Stiffener Areas

Note regarding bonded stiffeners:Depending on bond line quality and thicknes sysmetry, gate setting may need to be changed to detect second layer flaws.

This is a general training guide only to be used for training purposes.